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FINAL REPORT

Pre-Phase A Study for an Analysis of a Reusable Space Tug

VOLUME 4
SPACECRAFT CONCEPTS AND SYSTEMS DESIGN

Space Division
North American Rockwell

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MARCH 22, 1971

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FOREWORD

This volume presents spacecraft concepts and system design analyses in the Prephase A Study for an Analysis of a Reusable Space Tug. This study was conducted by the Space Division of North American Rockwell Corporation, Seal Beach, California, for the National Aeronautics and Space Administration, Manned Spacecraft Center, Houston, Texas. The effort was performed under Contract NAS9-10925. The six volumes comprising this final report include:

Volume 1.	Management Summary	SD 71-292-1
Volume 2.	Technical Summary	SD 71-292-2
Volume 3.	Mission and Operations Analysis	SD 71-292-3
Volume 4.	Spacecraft Concepts and Systems Design	SD 71-292-4
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ABSTRACT

Spacecraft configuration concepts of various diameters, propulsion capacities, modular arrangements, and mission modes were studied and compared. Resulting conceptual designs were utilized in weights, performance, operations, costs, and planning studies. Module configurations, internal and external, were explored. Conceptual vehicles responsive to NASA requirements were described, including suitable subsystems.

Candidate vehicle concepts were designed for various mission optimizations. These candidate vehicles were evaluated and three were selected for a more refined study. The three selected concepts were subjected to a more detailed design analysis in which all aspects of design were reexamined in greater detail. In addition, certain special conceptual studies were accomplished for "unusual" concepts involving unique criteria, configurations, or operational utilization. These were studies aside from the mainstream effort and were accomplished to explore variations from the baselines as well as ideas that might show unusual promise. Among these were special lunar lander configurations, ground-based concepts (rather than the primary spacebased), simplified design concepts, and alternate subsystems concepts.

The reusable space-based tug with modular design was found to be feasible as a design concept. The design is very sensitive to inert weights. The changes necessary for satisfying the requirements of a lunar lander were found to be extensive, and provisions for them in the basic tug would constitute excessive weight. Therefore, the lunar lander version of tug should be considered a block change.

ACKNOWLEDGMENTS

The following study team members made substantial contributions to this report:

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Reliability and Safety	-	L. K. Relyea
Subsystems	-	G. C. McGee

ABBREVIATIONS

The following abbreviations are used in this document:

AB	Approved bidders
ACS	Attitude control system
ACPS	Attitude control propulsion system
APU	Auxiliary propulsion unit
APCU	Auxiliary propulsion conditioning unit
CAM	Cargo module
CIS	Chemical interorbital shuttle
CLS	Chemical lunar shuttle
CM	Crew module
C/O	Checkout
CSM	Command service module (Apollo)
DET	Detail part fabrication
ECLSS	Environmental control and life support system
EO	Earth orbit
EOPD	Earth-orbital propellant depot
EOS	Earth-orbital shuttle (two-stage reusable)
EOSS	Earth-orbital space station
EPS	Electrical power subsystem
ESS	Expendable second stage

FO	Fail operational
FRR	Flight readiness review
ICD	Interface control document
IM	Intelligence module
IPP	NASA integrated program plan
IOC	Initial operational capability (date)
IRU	Inertial reference unit
LEO	Low earth orbit
LG	Landing gear kit
LM	Lunar module
LOPD	Lunar-orbital propellant depot
LSB	Lunar surface base
MK	Manipulator kit
MU	Mockup
OLF	Orbiting lunar facility
OLS	Orbiting lunar station
OMS	Orbital maneuvering system of EOS
OPD	Orbital propellant depot
PCA	Product configuration audit
PCU	Propellant conditioning unit (also APCU)
PL	Payload
PM	Propulsion module
RNS	Reusable nuclear shuttle

RST	Reusable space tug
TDRSS	Tracking and data relay satellite system
TS	Tank set
T-V	Thermal vacuum
T/W	Thrust to weight ratio
UP	Unmanned payloads

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ILLUSTRATIONS

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CONCLUSIONS

The conceptual design effort has supported the missions, operations, costs, and development studies. Conceptual designs have been made of many alternative modular and vehicle integrated concepts, and 70 drawings have been produced. From this study effort the following conclusions can be drawn:

1. Single stage, tandem two stage, and 1-1/2 stage concepts (the three baseline concepts) are viable, feasible candidates for accomplishing reusable space tug objectives.

2. Concepts 1 and 11 are more feasible than Concept 5 because the latter is not compatible with the EOS cargo bay limitations.

3. Major interfaces are those with other space program hardware systems such as the EOS, EOSS, OPD, CIS/RNS, OLF, LSB, and DSFN. The most important and sensitive of these interfaces is that of the EOS (the tandem two-stage concept is too long to fit into the EOS cargo bay).

4. A suitable set of subsystems has been identified for the tug and the operational design philosophy employed is sensitive in terms of influence on tug sizing and weight.

5. The conceptual design has permitted realistic analysis of preliminary weights for the three baseline concepts, and shows a variation of about 25 percent; these weights are very sensitive to staging modes as well as to inert weights. Advanced propulsion performance is required to provide the specific impulse and propellant inlet conditions necessary for tug performance with reasonable size and weight.

6. Docking requirements can seriously impact tug configuration design; the standard space station neuter docking hardware is too heavy to be considered for universal tug use, and centerline docking can affect or dictate engine numbers and arrangement.

7. A fully modular IM design has been found to be feasible but incurs a weight penalty for its structure which could be avoided if a nonmodular approach were acceptable as an alternative.

8. Design studies have indicated a close relationship between the engines, mission redundancy requirements, the centerline docking provisions, the adjacent O_2 tanks, and the structural load path provisions (if any) for kits such as lunar landing gear. These combined relationships have resulted in a four-engine cluster, four O_2 tanks, and a small centerline base docking port (passive) for the three baseline design concepts.

9. If lunar mission requirements are not allowed to impact the designs, a two-engine configuration results in a lighter vehicle which favors a one or two O_2 tank arrangement.

10. Without considering base docking, a single engine provides the highest possible engine performances for a given weight and therefore results in the lightest vehicle. However, the engine size is likely to be out of the range being considered on the EOS for the Orbital Maneuvering System and would thus preclude a potentially common engine development.

11. The basic tug should be designed for earth orbital mission operations. Modifications for lunar mission adaptation should be made as a block charge at a later time.

12. The tug and the EOS programs should be carefully coordinated to insure maximum commonality of hardware and technology for minimum combined program cost.

1.0 CONCEPTUAL DESIGN APPROACH

1.1 INTRODUCTION AND BACKGROUND

1.1.1 BACKGROUND

Without conceptual arrangements and their preliminary characteristics, it would not be possible to evaluate realistically the reusable space tug (RST) interfaces and operations nor should it be possible to determine a basis for relative costs and the ramifications of variations in missions, modes, and operations. Initially, the space tug was conceived to be launched either by the earth-orbital shuttle (EOS) or by a Saturn booster. The Saturn booster concepts appeared to be best suited for lunar missions because of their essentially unlimited diameter and lack of critical weight limitations. Later, it was determined that use of the Saturn booster for RST would be unlikely, and thus the influence of lunar missions on tug design was reduced. Compatibility with the EOS then became a dominant, driving consideration. Space station interfaces were determined to be important but less so than those of the EOS. Subsystems synthesis also was a necessary part of total concept synthesis and had a significant effect on concept weights.

1.1.2 OBJECTIVES

The objectives of the design concept synthesis effort were to:

1. Determine feasible concept configurations and arrangements
2. Permit realistic assessment of major interfaces and constraints
3. Enable synthesis of subsystems suitable for space tug missions
4. Permit preliminary weight analysis
5. Permit evaluation of mission modes and operations
6. Provide the basis for preliminary cost analysis and development plans

1.1.3 MISSION CONSIDERATIONS

The three general mission areas being considered are (1) low earth-orbital (2) high earth-orbital or planetary, and (3) lunar. These are characterized by low delta-V low-to-high payloads, high delta-V medium payloads, and moderate delta-V high payloads, respectively. Within these mission areas, several basic staging concepts were possible, each having a pronounced effect on configuration synthesis. The geosynchronous high orbit mission with a 10-K payload was found to be the performance driver, while the lunar landing missions were the strong configuration drivers in a multimission, universal or flexible concept (Figure 1-1). At the outset, the lunar mission configurations received major attention because of their strong influences on the multipurpose tug design arrangement concept; however, in the later phases of study, other mission areas (i.e., earth-orbital) were explored, and lunar missions penalties were not permitted to influence the configurations. This provided a basis for evaluating the multi-purpose versus limited purpose concepts and the logic for subsequent development of an evolutionary approach to a multipurpose tug.

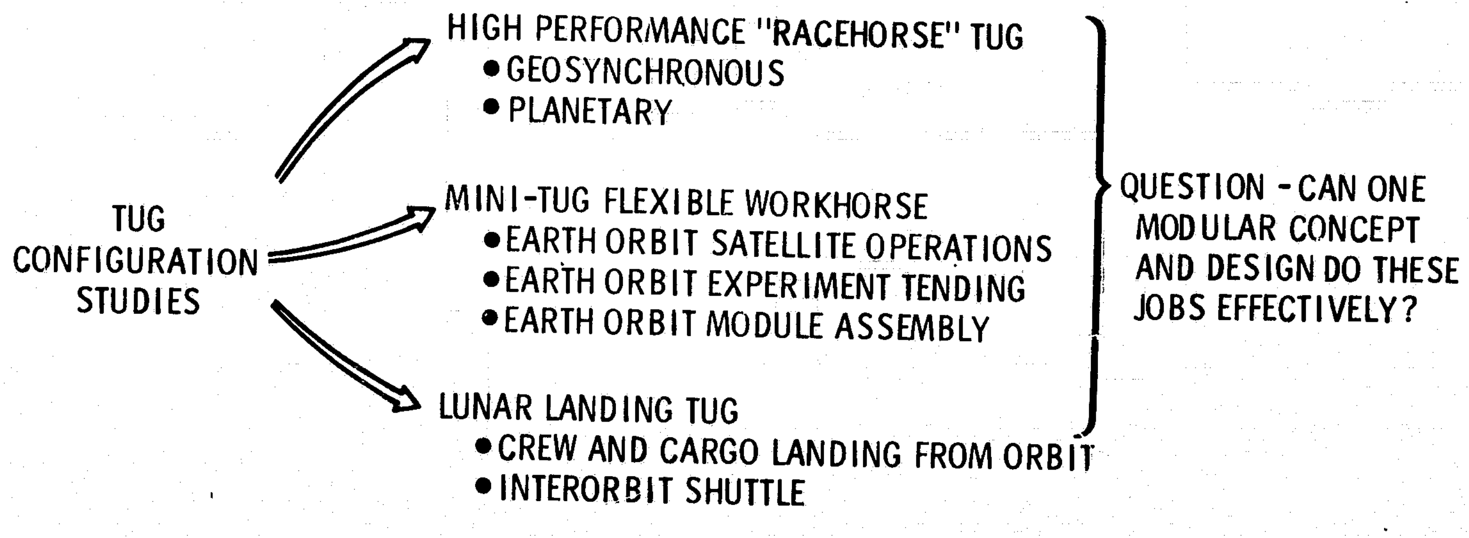


Figure 1-1. RST Mission Area Trend

1.2 CONFIGURATION DRIVERS AND CONSTRAINTS

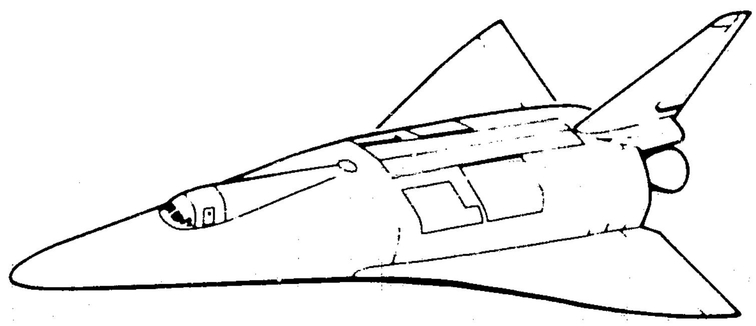
1.2.1 MAJOR IPP INTERFACES

Earth-Orbital Shuttle (EOS)

During the RST study, NR also conducted an EOS Phase A-B study for NASA-MSC. Close contact with EOS program study personnel helped to achieve a better understanding and insight into mutual characteristics affecting RST concepts. The foremost physical constraint has been the 60-foot (18.3 m) length and the 15-foot (4.6 m) diameter limits for the RST in order to be carried within the EOS cargo bay. A number of different schemes were studied for payload deployment (such as the RST and cargo modules) in and out of the EOS cargo bay. The basic approaches that appear to have the most promise are believed to be a swing-out manipulator arm at the forward end of the EOS cargo bay to which payloads, such as the tug, would be attached and to which the tug would be hard-docked, and a cherry-picker set of manipulator arms for cargo deployment and soft docking (Figure 1-2). In the latter case, the tug payload would be restrained longitudinally and laterally within the bay along its length rather than primarily at one end as with the earlier manipulator concept. These EOS approaches naturally affect the EOS-RST docking concept and structural support provisions and thus have been monitored closely in the RST study.

In some cases, the RST concept length with geosynchronous payload may exceed the cargo bay constraint. In this event, the payload would be launched separately (second EOS) and then mated to the tug in orbit. Exceeding the EOS cargo weight limit was generally not a problem, because the tug is space-based and therefore can be placed in orbit dry of propellants for subsequent orbital fueling. With ground-based concepts, the gross weight of the tug may be limited by EOS cargo weight limitations.

The tug payload was assumed to be largely independent of the EOS "mother ship" for other than structural interfaces while in the cargo bay. Exceptions may be electrical interface for status monitoring and possibly for control and fluid interfaces for propellant dump and even for emergency transfer of propellants into EOS during an abort maneuver (if the tug is ground-fueled). Naturally, the RST and EOS have a communications interface, which could include monitoring and remote control (following deployment) and active or passive communications and control from either vehicle during return rendezvous and docking maneuvers.



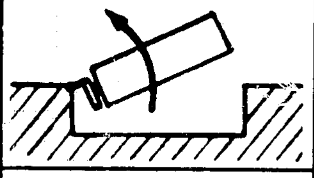
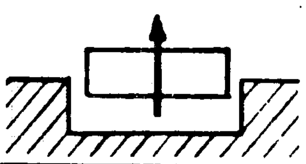
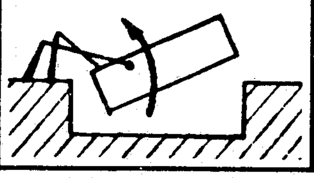
APPROACH	GEOMETRY	DESCRIPTION
ROTATION		PAYLOAD ROTATED OUT OF CARGO BAY ABOUT FORWARD END
TRANSLATION		PAYLOAD TRANSLATED UPWARD FROM CARGO BAY
ARTICULATION		PAYLOAD PLUCKED OUT OF CARGO BAY BY CHERRY-PICKER ARMS

Figure 1-2. Earth-Orbital Shuttle Cargo Deployment Alternatives

Other Operational Interfaces

The basic space-based tug concepts are assumed to be launched empty of propellants for three reasons. The first reason is that the cryogenic propellant tank insulation concept can be simpler and more effective, and the second reason is the likelihood that the RST would exceed EOS cargo weight capability if it is fully fueled prior to launch. A third reason is that extended, multiple missions from a space station base may be more economical. Loads while in the EOS bay also would be low when the tug is not fueled with propellants. With a Saturn launch, however, payload weight restrictions are not encountered, and the tug would most likely be launched fueled (Figure 1-3). In this case, therefore, the Saturn launch accelerations of approximately 5 g would establish the tug design loads.

A truly space-based RST undoubtedly would be based at an earth-orbital space station such as the EOSS, which also is under Phase B study at NR for NASA-MSC. This implies docking at the EOSS with quiescent storage between missions. Thus, compatible docking interfaces are required. It also is likely that such a space station would serve as a control center for RST operations, including remote control (when the tug is unmanned).

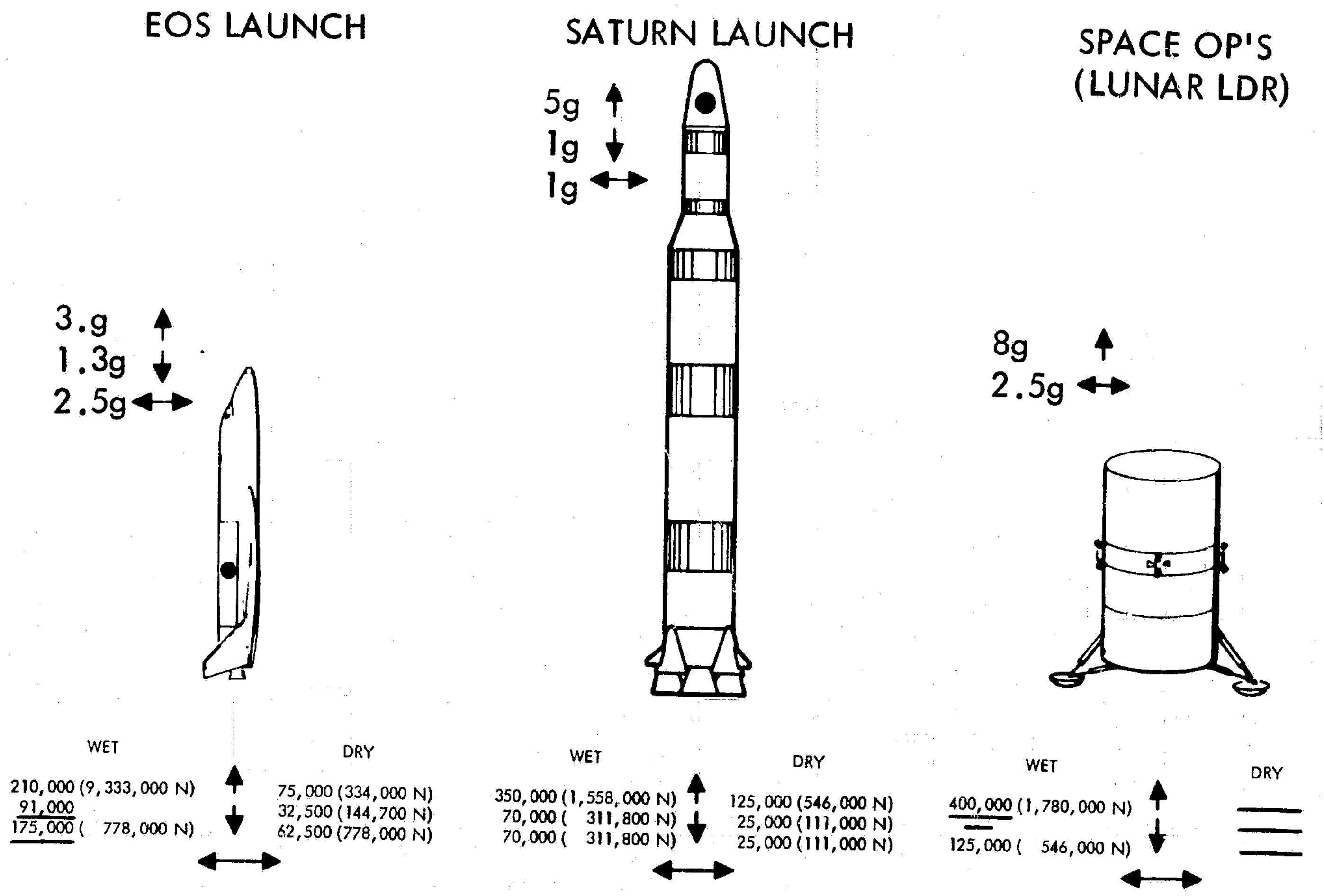


Figure 1-3. Ground-Based Versus Space-Based Loads

Another possibility would be to use a space-based tug in connection with an orbiting propellant depot (OPD) or facility such as also under current study at NR (NASA-MSFC). Docking and compatible refueling provisions would be necessary in this case. (Design concepts presented later in this report include such provisions for refueling.) These provisions also could be used when there is no separate OPD and the EOS, itself, is used in a tanker mode instead.

Should one of the large chemical injection stages (CIS) such as the S-II derivatives (under study at NR for NASA-MSFC) be used, then the RST also must be compatible with this vehicle. Similarly, the use of either a reusable nuclear shuttle (RNS) or its chemical equivalent (CLS) for earth-lunar orbit operations implies tug interfaces (both concepts also are under study at NR for NASA-MSFC). At the lunar end of cislunar missions, the RST might interface operationally with an orbiting lunar station (OLS) or lunar surface base (LSB). These two lunar concepts are being studied at NR for NASA-MSC and NASA-MSFC, respectively.

The RST has been identified as having the most interfaces of all IPP elements[1] (Figure 1-4). The approach was to consider the physical and operational interfaces in each case and to assess their influence on RST design and operations. Because nearly all of these elements are in a state of flux or are only loosely defined at this time, it is extremely important that a flexible approach be taken in the immediate future with generous allowances for alternatives. Therefore, it is especially important to maintain cognizance of the interrelationships of these IPP elements; the study was benefited in this respect because of NR's broad involvement in IPP studies and concepts. In addition to the size and weight constraints of the EOS cargo bay, some other particularly important interfaces include a universal rendezvous and mechanical docking system, refueling system, communications interfaces, and an integrated cargo handling concept between space systems.

1.2.2 EARTH ORBITAL VERSUS LUNAR MISSION REQUIREMENTS

The differences in requirements for earth orbital and lunar missions are such that they do constitute major differences. Low earth orbit missions generally do not require high performance from the space tug, but instead they are characterized by payload variety and flexibility, high mission frequency, and small to moderate ΔV increments. They may be either manned

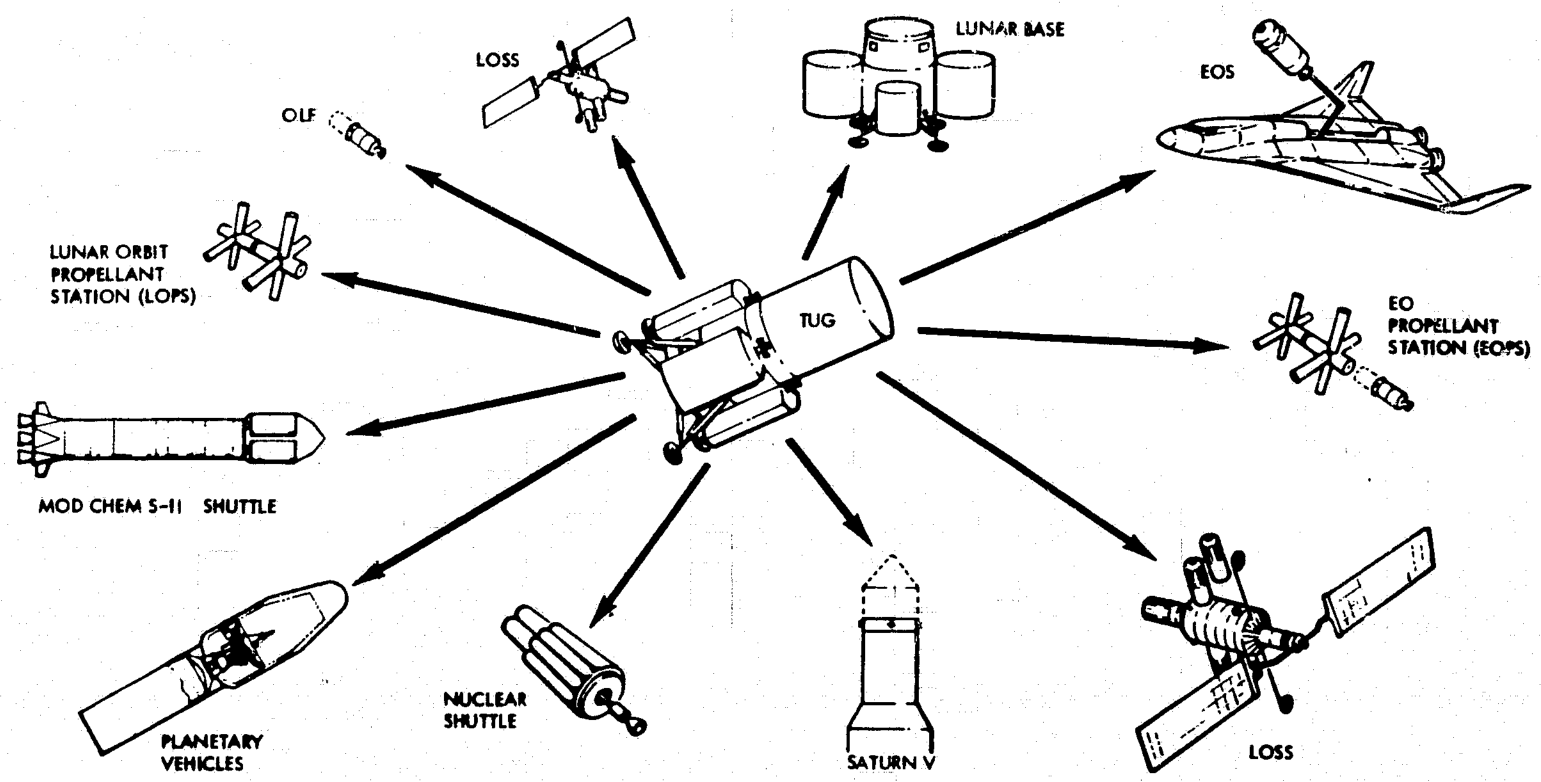


Figure 1-4. Docking Interfaces

[1]VonTiesenhausen, G.:

or unmanned with remote monitoring and control being provided from another major IPP element such as the EOS or EOSS. Lunar missions provide unique driving requirements on the configuration in several respects (Figure 1-5):

1. Addition of lunar landing gear
2. Requirement for deep throttling of engine(s)
3. Need to provide low vehicle cg for landing stability and light landing gear
4. Need for each crew and cargo access to and from the lunar surface
5. Visibility for a manned landing during descent
6. Critical mission phases during descent and ascent, which tend to require multiple engines
7. Tendency to require scar weights on the basic RST
8. Assembly of kits peculiar to lunar missions
9. Longer mission durations and remote basing (lunar orbit)

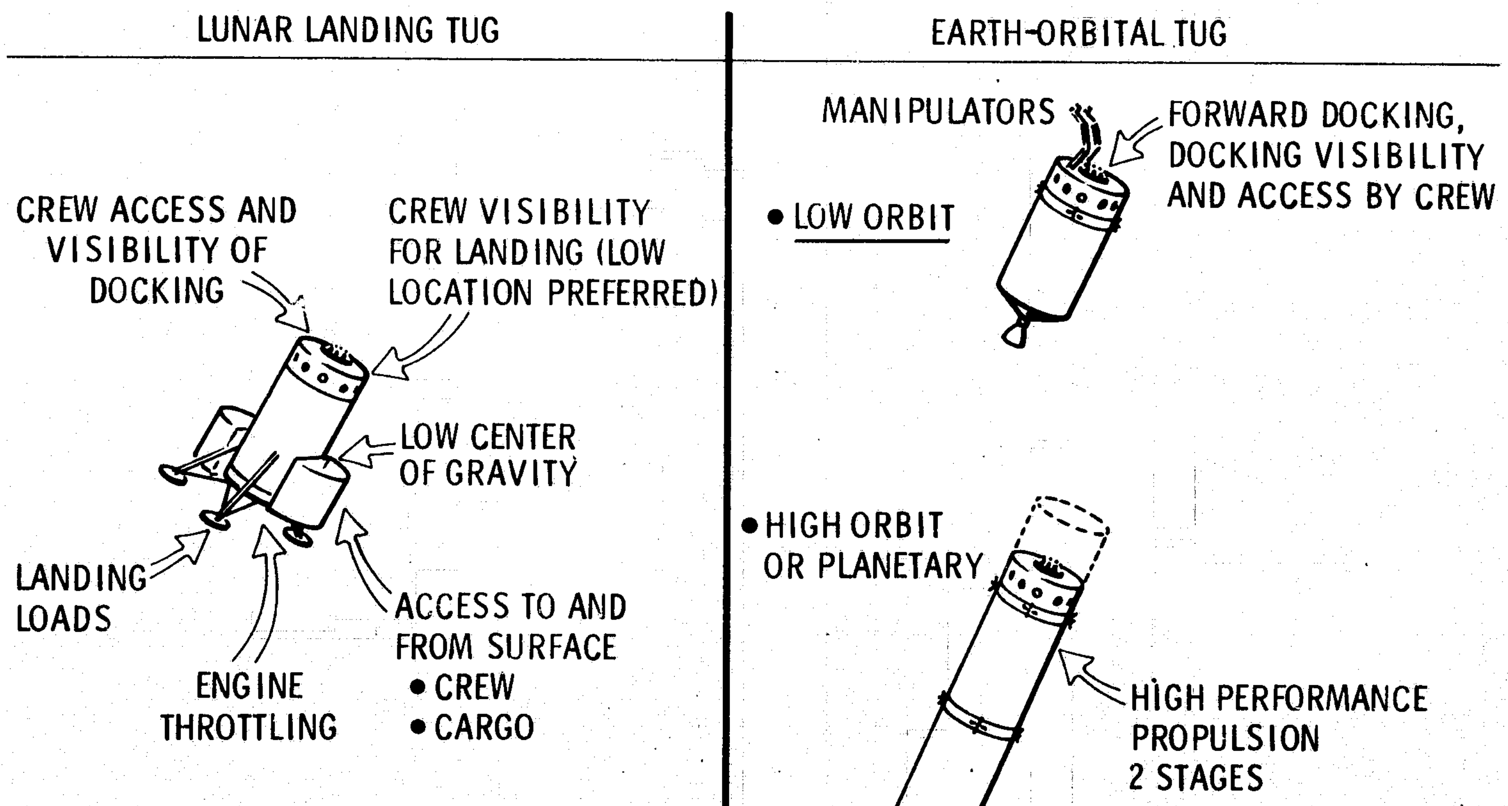


Figure 1-5. Concept Design Drivers

The necessary characteristics are believed to be achieved best by utilizing landing gear in kit form, which can be assembled to the vehicle in earth or lunar orbit. Other kits needed for the lunar landing mission only are extra horizontally oriented radiators for the lunar surface stay, extra consumables, a possible larger antenna (depending upon the relay systems availability in lunar orbit), and cargo pods.

Proportional engine throttling is required for good dynamic control during lunar touchdown maneuvers. Landing gear becomes long and unduly heavy if the vehicle cg is not kept low by locating the heavier modules and components low in the landing configuration. This and the need for easy unloading access on the surface tends to favor balanced outboard (laterally displaced) cargo pods or modules. Likewise, surface access and visibility considerations (as well as low cg) tend to favor having the crew module on the bottom of the stack. During the final landing retro and touchdown maneuvers or during lunar takeoff, engine failure could be disastrous; thus the tendency is to favor multiple engine configurations. On the other hand, the difficulty of achieving the important geosynchronous 10-K payload insertion mission (which establishes RST propellant capacity) attests to the need to avoid scar weights for lunar missions.

In some earth-orbital missions such as payload transfer to and from EOSS-EOS, double-ended docking on the tug would permit alternately maneuvering the fresh and the spent cargo payloads, thus facilitating cargo module exchange; although alternate ways of accomplishing this mission may be available. Docking at the aft (engine) end of the tug also may be desired for interfacing with the EOS cargo bay manipulator. (However, this concept may not be the final choice for EOS cargo handling.) These aft docking requirements tend to preclude a single-engine configuration, unless an unusual and larger-diameter concentric docking mechanism were permissible.

The geosynchronous orbit 10-K payload placement mission establishes the vehicle propellant sizing as the most demanding primary mission because of the very high delta V encountered. These demands require that inert weights for other missions are to be avoided generally in order that vehicle propellant sizing does not become excessive. Likewise, high performance from engines is demanded. This tends to favor a single large engine as compared to multiple smaller ones, because engine technology cycle balance limitations do not permit the necessary high chamber pressure for high I_{SP} at high mixture ratios in smaller engines. Multiple engines also are heavier but are shorter and, moreover possibly can be nested better in and around tanks to yield a shorter vehicle.

In low earth orbit, payloads vary from very small to very large and massive (up to 1/4 million pounds - 136,000 Kg), such as a space station module or RNS. Generally, delta-V requirements are small and sometimes can be handled by the RCS engines alone. Manned earth-orbital missions require the CM to be forward (opposite end from engines) to provide visibility for rendezvous and docking operations and for pressurized docking transfer of crew. For manned lunar landers, the CM should be on the bottom of the stack to facilitate landing vision and to provide for easy surface egress and access. Such considerations are shown in Figure 1-5.

RCS jets should not be located close to the RST payload docking end, if large diameter payloads are to be moved because of potential jet impingement. The wide range of possible tug payloads and configurations makes longitudinal cg a wide variable, and this makes ACPS-RCS jet placement (with a minimum number of jets for adequate redundancy) more difficult.

1.2.3 OTHER FUNCTIONAL DESIGN CONSTRAINTS AND DRIVERS

The approach to concept configuration design considers the effects of such factors as relationships between functions, modular philosophy, operations interfaces, and physical constraints.

Modules

The tug vehicle subdivides into convenient modules according to considerations of manufacture, test, mission operations, technologies involved, alternate uses, and relationships between functions and modules. These are the propulsion module (PM), intelligence module (IM), crew module (CM), and cargo module (CAM), plus special mission-peculiar kits, which can be utilized with the other tug modules (Figure 1-6).

PM

The PM function involves primarily propulsion technology for propellant and gas tanks, pressurization, thermal control, venting, refueling and dumping, controls and actuators, and engines. The technologies involved are principally those of the major engine manufacturers, such as Aerojet, Pratt & Whitney, and Rocketdyne, except that the tankage, related structure, and thermal control are most commonly provided by the major space system developers. The PM is a convenient unit to manufacture and test and can be used singly or in multiples, parallel, or tandem, where required. Because it is used on nearly all types of missions regardless of the others and because it is critical to basic vehicle performance, the PM is one of the most easily identifiable basic units or modules.

EARTH ORBIT MISSIONS

PM

- Maximum performance in geosync mission
- Single engine preferred for efficiency
- Short configuration to save space

CM

- 6-man crew transport
- 2-man working module
- Pressurized crew transfer
- Visibility of payload docking operations - forward location

IM

- Subsystems, "brains"
- Manned or unmanned provisions
- Remote or manual control

CAM

- Easy loading at space station
- Transferable to another vehicle
- Pressurizable, double docking

KITS

- Extra provisions for manned flight
- Manipulators for repair and retrieval
- Neuter docking adapters - either end of any module

LUNAR MISSIONS

PM

- Modified high performance - minimum propellant logistics
- Minimum scar provisions for landing missions
- Multiple engines desired for redundancy

CM

- Provisions for 4 men, 28+ days stay
- Pressurized crew transfer
- Lunar landing visibility, surface access - low in vehicle stack

IM

- Subsystems, "brains"
- Adaptable to manned or unmanned operations
- Additions for lunar

CAM

- Single module can be left on surface
- Easy loading at LOSS
- Easy unloading on surface
- Little effect on vehicle c.g.

KITS

- Extra subsystems required for manned flight
- Landing gear - minimum tare weight on vehicle
- Extra cooling radiators

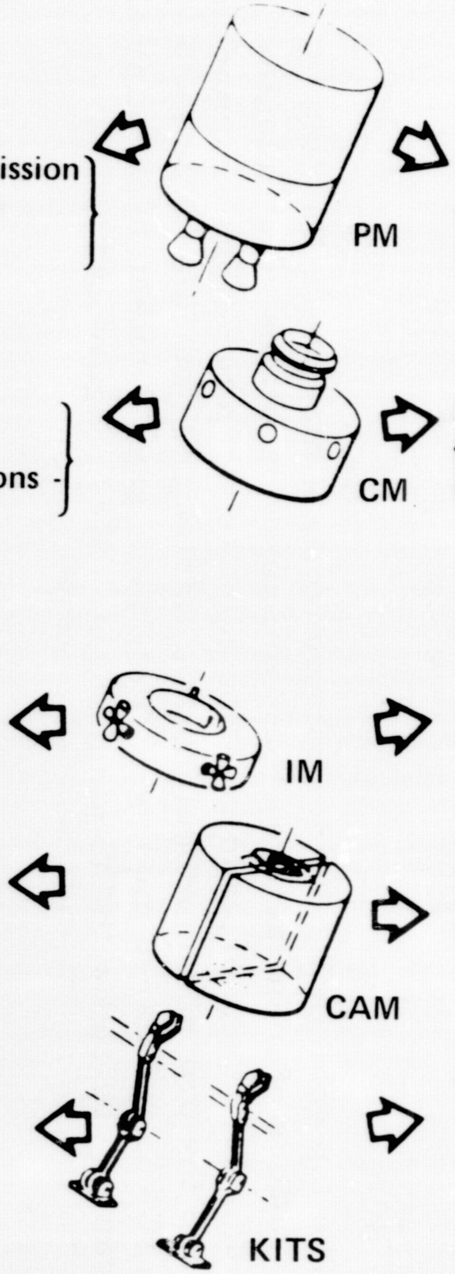


Figure 1-6. Desired Vehicle Module Characteristics

The PM physical design constraints are indicated in Figure 1-7. The length and diameter constraints permit the PM to be transported to orbit by EOS with payload attached if possible to avoid orbital assembly operations. Provisions on the upper end of the PM provide for the always-present IM support attachments; being a multifluid and electrical interface, the attachment is best accomplished on the ground where clean room conditions and excellent checkout facilities are available. Except for design constraints, any number of engines could be used. These constraints include desirable centerline docking, nesting of engines between, around, and amont propellant tanks for minimum length, and the potential for utilizing a common engine development with the EOS orbital maneuvering system. The aft docking port is convenient for handling multiple cargo module exchanges in low earth orbit, docking to a manipulator facility in EOS cargo bay, or for permitting convenient docking to an orbital propellant facility or depot (OPD).

IM

The IM is considered to be a convenient accumulation of the "brains" and minor "muscles" (G&N, telecom, power, attitude control propulsion system, and ACPS). The equipment involved is primarily electronic in nature but includes electromechanical and fluid systems as well. The main reason for the existence of the IM as an entity is the flexibility in application it affords for variable uses and configurations.

An alternate philosophy, which may be preferable where diversity of applications are not considered as important, is to utilize a semimodular IM approach. Here, the functions and equipment that are closely related in terms of technology or interaction are located in the PM, and the more strictly electronic items only are located in a separate area or partial IM. Examples of the PM-related equipment are the ACPS and electrical power equipment. In the electronics category are the avionics equipment, such as G&N, telecom and data. This alternative IM approach appears to be potentially superior (although at the possible sacrifice of autonomous operations flexibility) in natural separation of items for manufacturing, test, and checkout and refurbishment interfaces or interrelationships. For the reusable space tug study, the full modular IM flexible approach has been adopted as a baseline for the vehicle concepts, but the semimodular IM also is considered a viable candidate.

IM design constraints are shown in Figure 1-8 for a fully modular concept. Typical equipment is shown, including RCS engine clusters. Analysis of the requirements for redundancy resulted in the pentad cluster indicated, and these clusters are retracted within the maximum diameter by hinging or rotating the doors on which they are mounted.

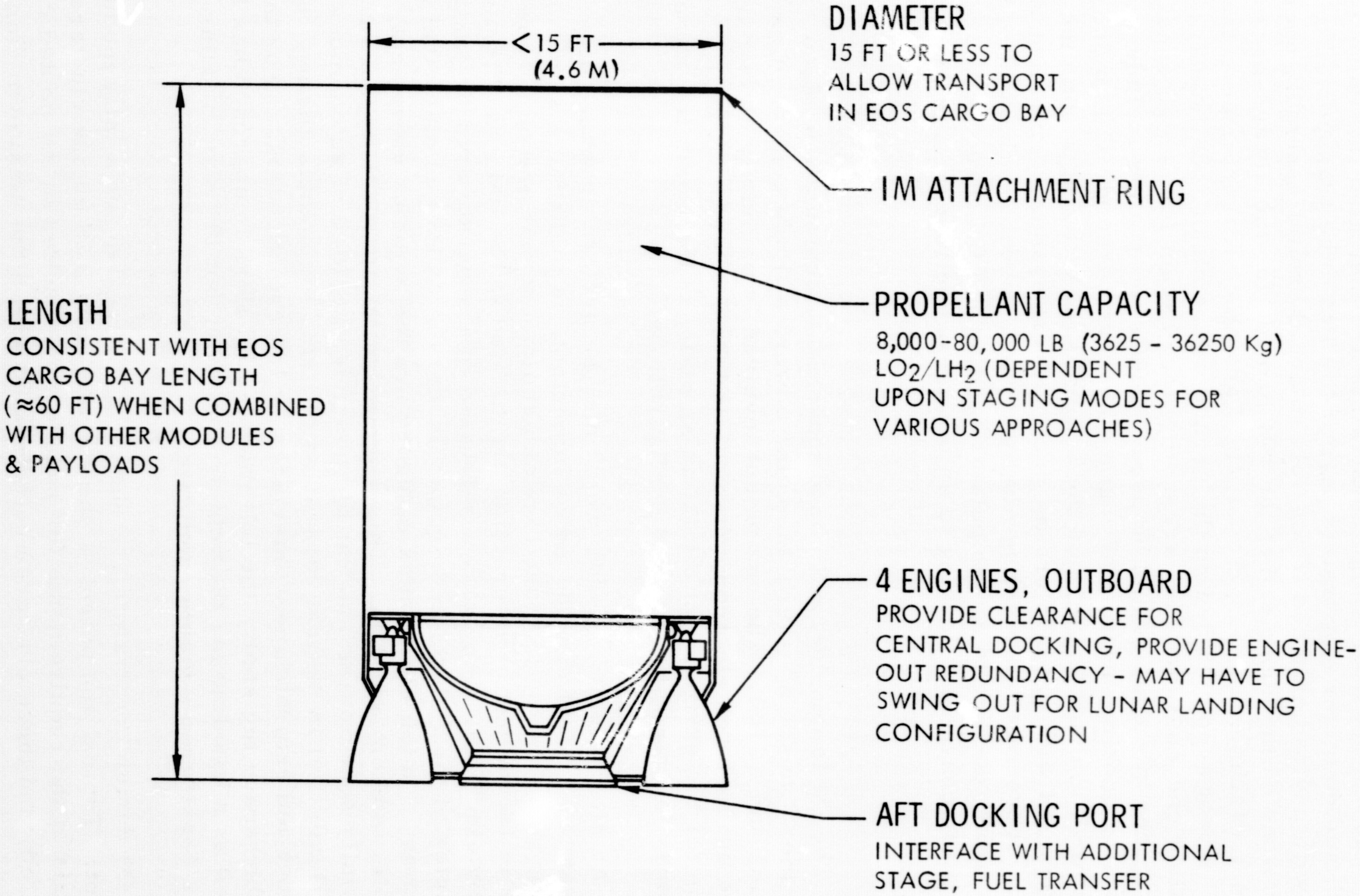


Figure 1-7. Propulsion Module Design Constraints

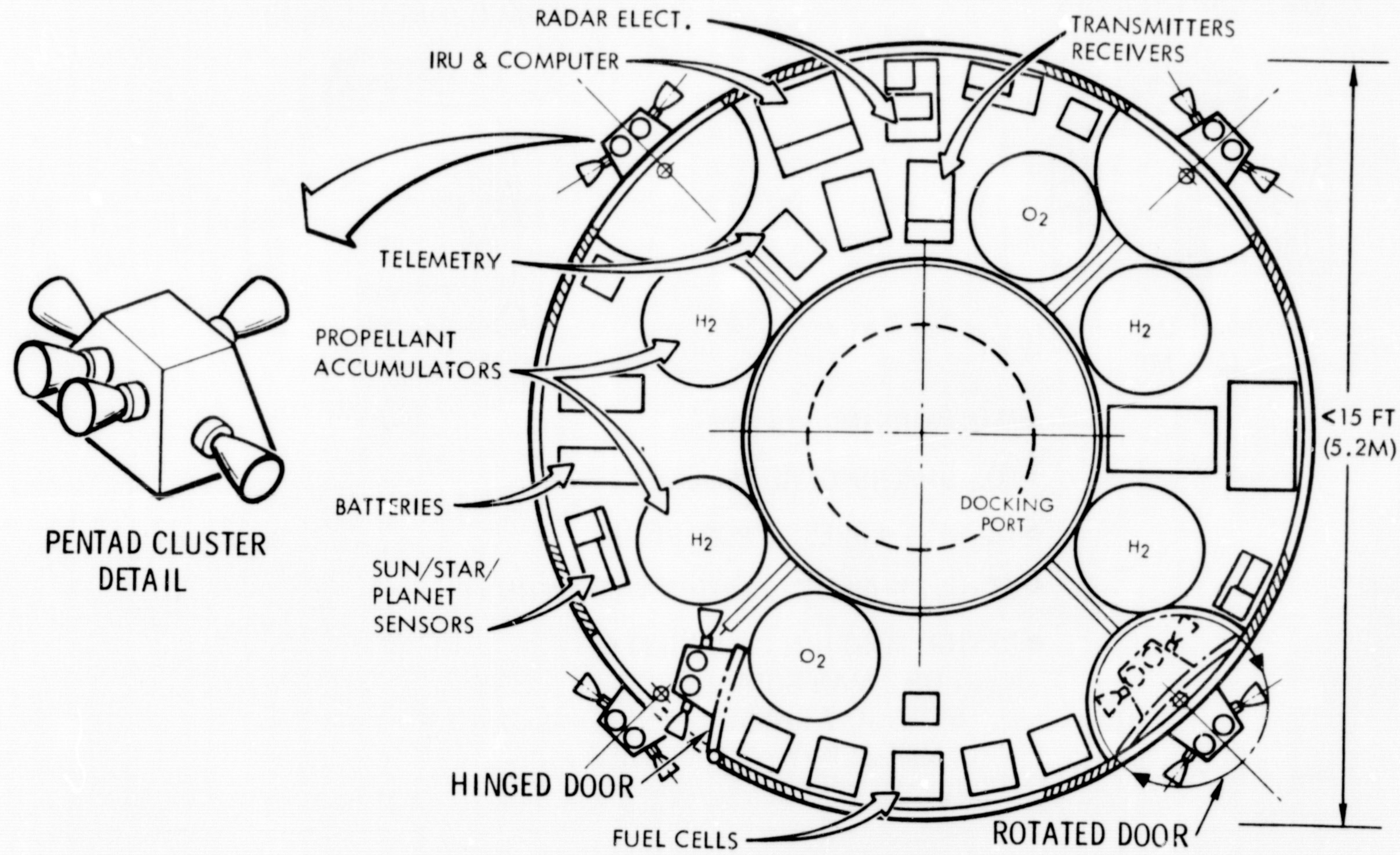


Figure 1-8. Intelligence Module Design Constraints

CM

Tug missions may be either manned or unmanned; therefore, the CM is a separate optional module. Because of weight sensitivity of the tug in certain high delta-V missions, it is necessary to put the additional subsystems capability for manned missions in the CM or add them to the IM, which is basically constituted for unmanned missions. This approach minimizes the scar weight carried on unmanned missions.

Figure 1-9 shows crew module (CM) design constraints. The 15-foot-diameter (4.6 m) module with a length of approximately 8 feet (2.4 m) has more than adequate space volume per man (in accordance with accepted standards) for tug mission conditions. The CM is basically a 7-man, 7-day mission crew transport for up to 6 men (the latter figure could be doubled for emergency rescue missions). Provisions are shown for either vertical or horizontal orientation, requiring pressurized docking for shirtsleeve crew transfer plus an emergency escape hatch. The horizontal configuration evolves with a 12-foot (3.7 m) diameter and does not stack as well nor

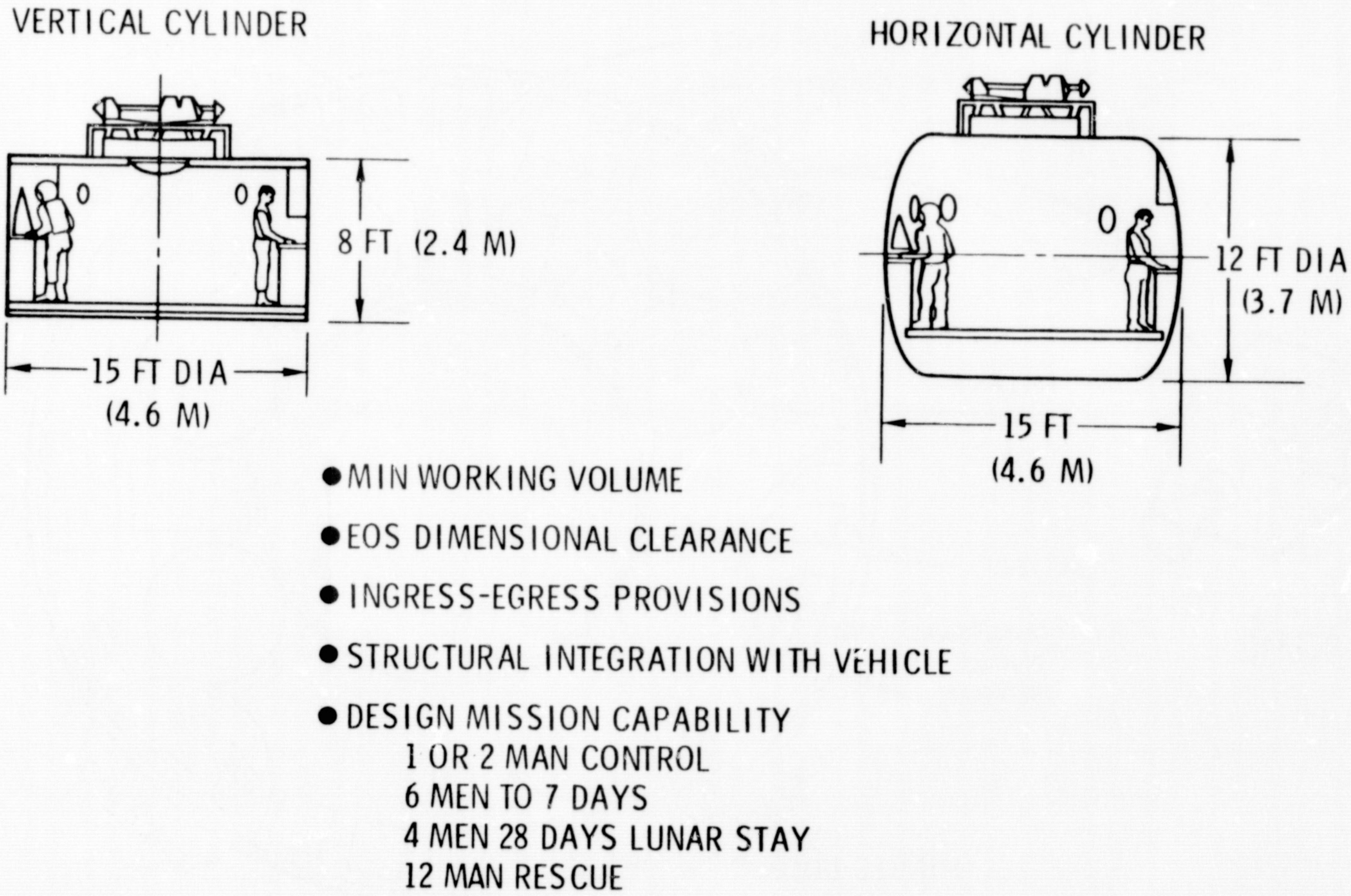


Figure 1-9. Crew Module Design Constraints

integrate as easily with other cylindrical modules. The same basic module can be simply converted to serve as a temporary lunar station (or earth and lunar orbital station) module for lunar landing missions. The design of this module is such that it could also be utilized in a number of other ways and in alternate system applications to minimize the overall costs with separate but similar developments. Examples of its use would be as an interim space experiment module, orbital station modular element, OPD control module, or use of the CM shell as a pressurized cargo module.

A minimum crew module for manned control only would be a two-man module (1 pilot capable of control). Considering the number of times it would be needed, the cost of an extra module development and the logistics scheduling problems with two types of CM's (2- and 6-man), the use of the 6-man CM off-loaded is a better approach.

CAM

Requirements for a cargo module (CAM) appear primarily in the lunar mission area. However, the EOSS program studies have included the use of cargo modules in conjunction with the space station resupply and experiment functions and in other experiment-oriented programs utilizing

EOS (SOAR). On the shuttle study, transfer of cargo modules via tug to the EOSS and return of spent "garbage-can" cargo modules to the EOS for earth return has been considered.

For landing 10-K (4.5 Kg) payloads on early lunar 28-day-stay manned missions, the use of hemipods, which are halves (or quarters) of the full modular diameter permitted in delivery to orbit by EOS, are attractive for landing mission support or experiment supplies. When separated in earth or lunar orbit, the halves can be attached low on opposite sides of the tug lander to provide low cg for minimum landing gear weight and easy surface access to the cargo. These modules or pods could be left on the surface for some subsequent base use or conversion. The two hemipods could be bolted together if a full cylindrical module were required for earth-orbital use.

1.3 OVERALL CONCEPTUAL APPROACH

1.3.1 CONCEPTUAL DESIGN LOGIC

The logic of the study conceptual design effort is shown in Figure 1-10. Based on the preliminary requirements analysis of mission operations and estimates of performance weights, rough requirements for propellant quantities were established. From these data preliminary conceptual layouts explored the many configurational possibilities for modules, spacecraft, and interfaces with other hardware elements. This activity was conducted in conjunction with operations studies and systems studies in a continual feedback and refinement loop. The candidate configuration concepts that evolved for various design optimization assumptions were evaluated, and the three most favorable concepts chosen at the study midterm point. Following this, additional refinement iterations of mission operations, interfaces, system performance, weights, systems and subsystems characteristics, and reliability considerations were conducted while the three baseline concept designs were being refined. Data for the refined concepts then were utilized in the costs and development plans analyses.

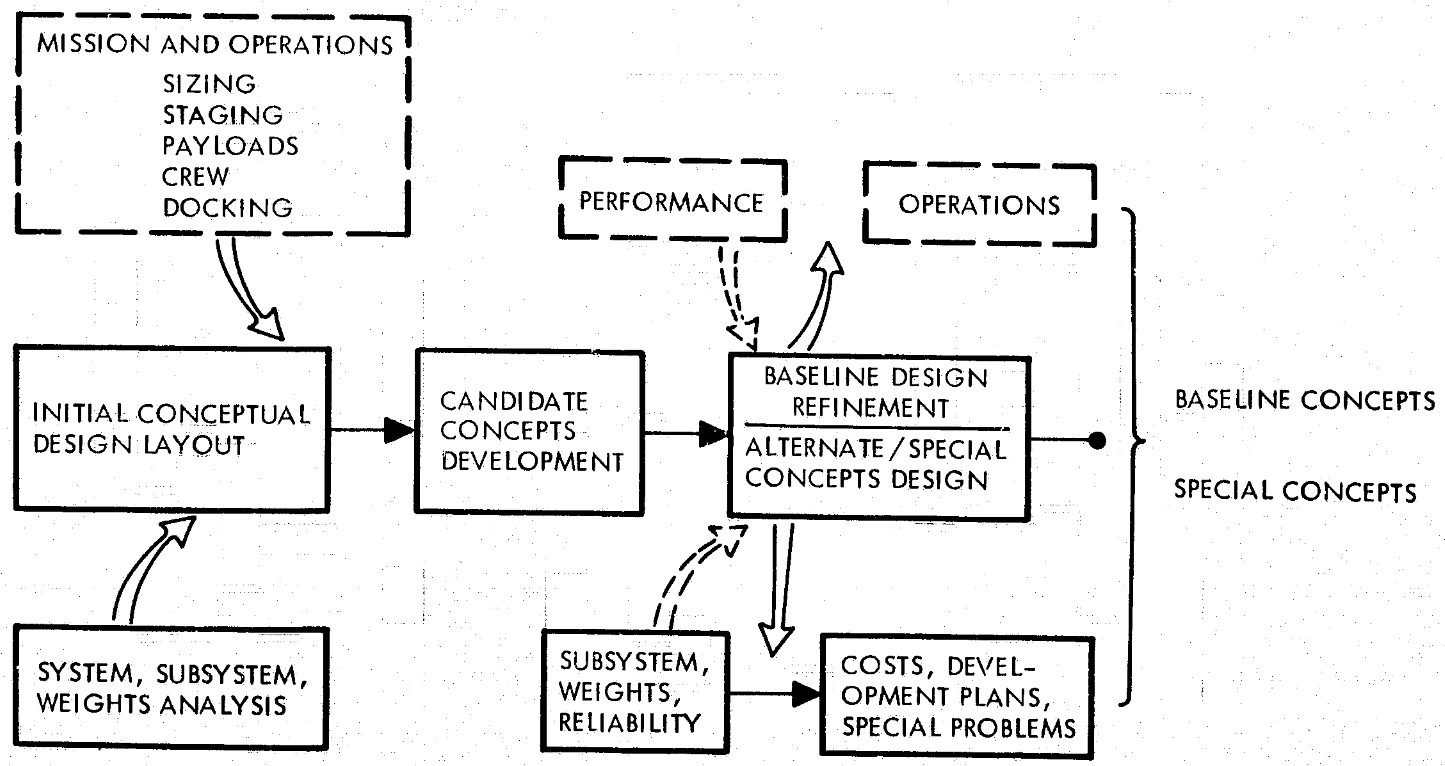


Figure 1-10. Conceptual Design Logic 99

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1.3.2 MODULE SIZING

Initially in the study, the basic modular diameter was considered "open" and still to be determined. Factors in determining module shape and dimensions were the triple considerations of launch vehicle compatibility, minimum inert weights, and general reasonableness of the sizes and shapes or volumes that result in regard to usage, convenience, assembly, and interfaces (Figure 1-11). PM diameters of 22 feet (6.7 m), 15 feet (4.57 m), and 12 feet (3.66 m) were explored (Figure 1-12). The 22-foot diameter concept would be basically compatible with a Saturn booster. The 15-foot diameter concept is basically compatible with the EOS cargo bay clearance limit, although it should be noted that in EOSS Program studies, the PM 14-foot (4.27 m) diameter has been considered to be the outside diameter limit for EOS cargo and experiment modules. The 12-foot diameter PM was explored to provide a range for parametric evaluation of the diameter effects. Weight relationships are shown in Figure 1-13 for the basic element, the PM. The 15-foot diameter module is seen to be near optimum. In addition, it was found during the study that EOS launch is probably the only viable approach, and this would heavily favor the 15-foot diameter concept. (A brief study indicated that a larger diameter tug carried outside the mold-line of the EOS is not a total impossibility in theory but in practice could be a most difficult task aerodynamically and structurally. A diameter smaller than 15 feet (4.57 m) would produce a lighter module, but the EOS cargo bay space would not be used effectively and this would tend to lengthen the "bird" beyond the allowable limit of 60 feet (18.3 m) for both tug and payload.

Having determined the base diameter, the approach was then to conceive the shortest PM configuration consistent with keeping weight low so that propellant and overall weights could be kept low. In general, the additional constraints or features required and desired, such as multiple engines, aft docking and the weight minimization, were accorded more priority than absolute minimum length because of the high leverage of inert weight on required propellant, up to 7.5 pounds (3.4 Kg) of propellant for each 1 pound (0.45 Kg) of inert weight. These and other options studied related to PM concept sizing are shown in Figure 1-14. A large recoverable single-stage tug would provide simplicity, large propellant reserve on many missions, a temporary orbital propellant storage facility, and flexibility for accommodating unusually large delta-V or payload missions by tandem staging. However, some negative factors must be realized (in parentheses in Figure 1-14). The single-stage concept, as optimized for the high-energy geosynchronous 10-K payload mission, is large and heavy and somewhat oversize for lunar missions where costly propellant resupply may be aggravated. An expandable stage would be small and therefore would be a useful size for low earth-orbital missions. However, negative factors are

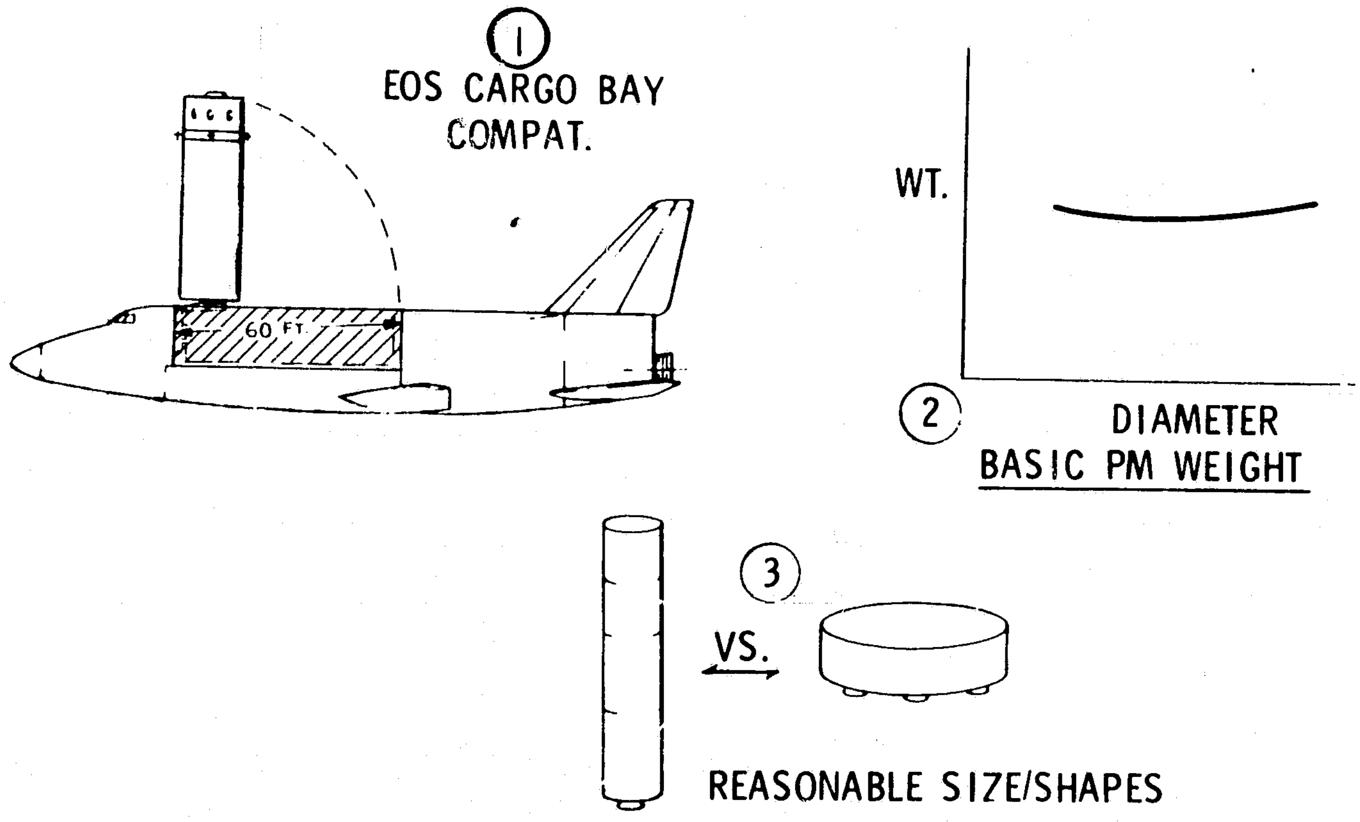


Figure 1-11. Main Considerations in Choice of Concept Dimensions

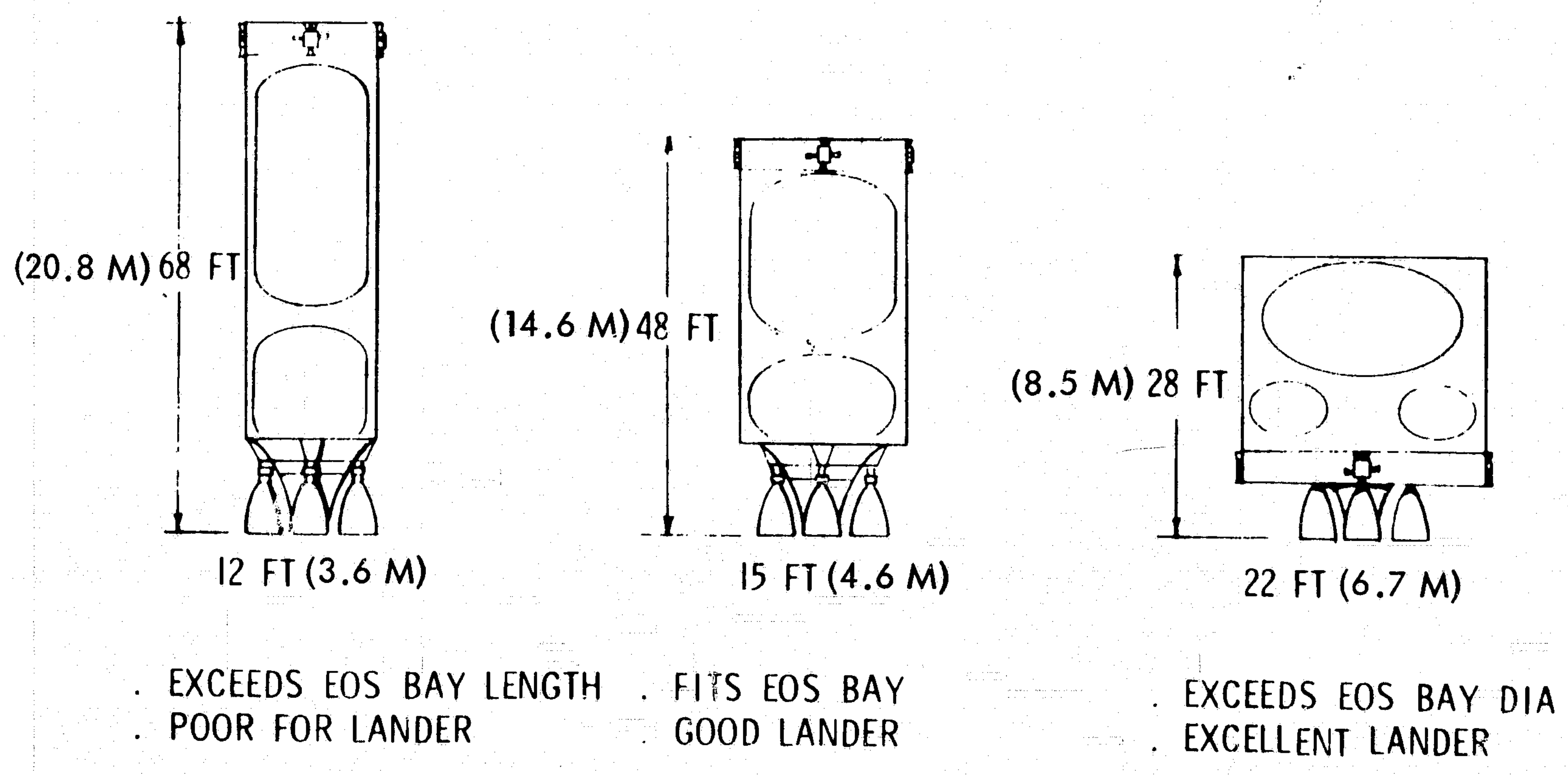


Figure 1.12. Propulsion Module Geometry Factors, 80,000-Pound (46.3 Kg) Propulsion Weight Shown

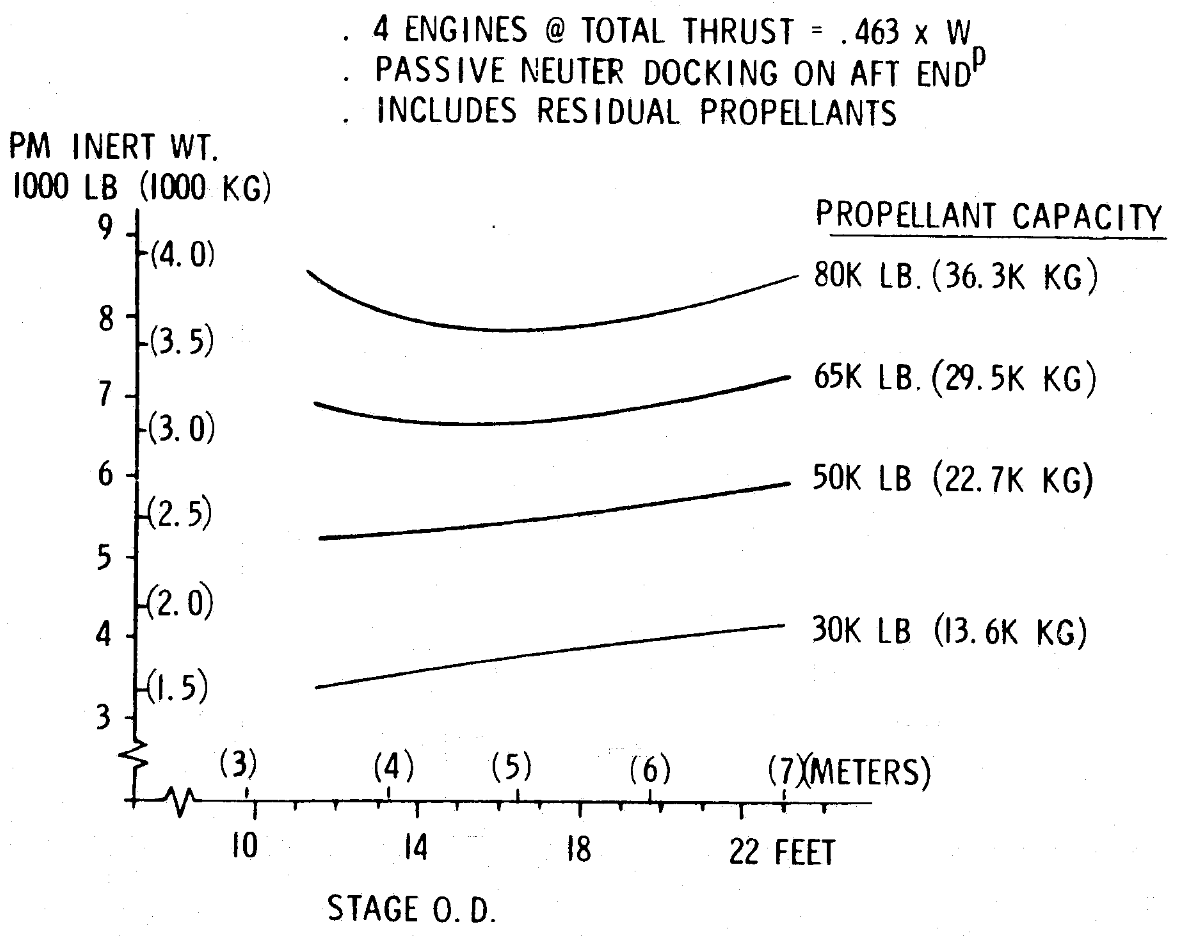
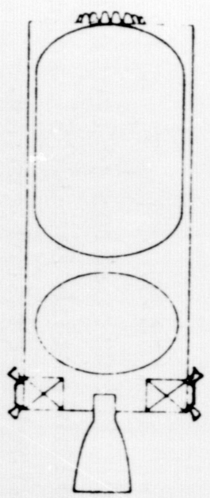


Figure 1-13. Propulsion Module Weight Trends

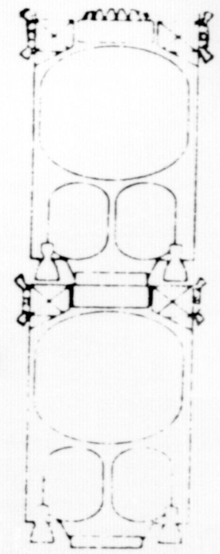
that expending the stage may tend to be costly, and the small size and expendable compromises for cost do not permit the degree of mission flexibility and reserve as does a large stage. As a lunar lander, it is too small and would tend to require more complex clustering of two PM's. The tandem-staged concept provides high performance and a basic versatile PM size, but it requires two IM's and does not necessarily fit lunar-lander sizing concepts well. The expendable tank set concept appears to have many virtues as a result of being sized for expenditure of the tank set on a geo-synchronous (high-energy) mission. The total gross weight is relatively low compared to a single stage although more complex, and the small PM fits very well into many low earth-orbit missions. The total concept appears to be approximately the right size for lunar landing.

The IM sizing approach was to investigate the design of a fully modular concept, a submodular concept in which the subsystems could be separated as units from a major assembly module, and the dispersed or integrated concept (components integrated within PM wherever convenient) as shown in Figure 1-15. The module concept would have a



LARGE SINGLE-STAGE RECOVERABLE
- SIMPLE
- PROPEL RESERVE POTENTIAL
- FLEXIBLE POTENTIAL
- (LARGE DIMENSIONS)
- (HEAVY)
- (BIG FOR L.E.O. MISSIONS)
- (LARGE L. LANDER)

2-STAGE
- HI PERF.
- FLEXIBLE SMALL SIZES
- SMALL LEO CONFIG.
- (REQ 2 IM'S)
- (NOT RIGHT FOR L. LANDER)



SINGLE-STAGE EXPENDABLE
- SMALL, COMPACT
- GOOD FOR L.E.O.
- (COSTLY)
- (LESS VERSATILE)
- (TOO SMALL L. LANDER)

EXPENDABLE TANK SET
- SMALL LAUNCH CONFIG.
- LOW LAUNCH WT.
- SMALL PM SIZE FOR LEO
- FLEXIBLE TANK SIZE APPLICATIONS
- GOOD L. LANDER

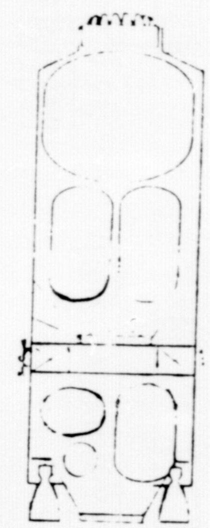


Figure 1-14. Propulsion Module Concept Sizing Options and Virtues

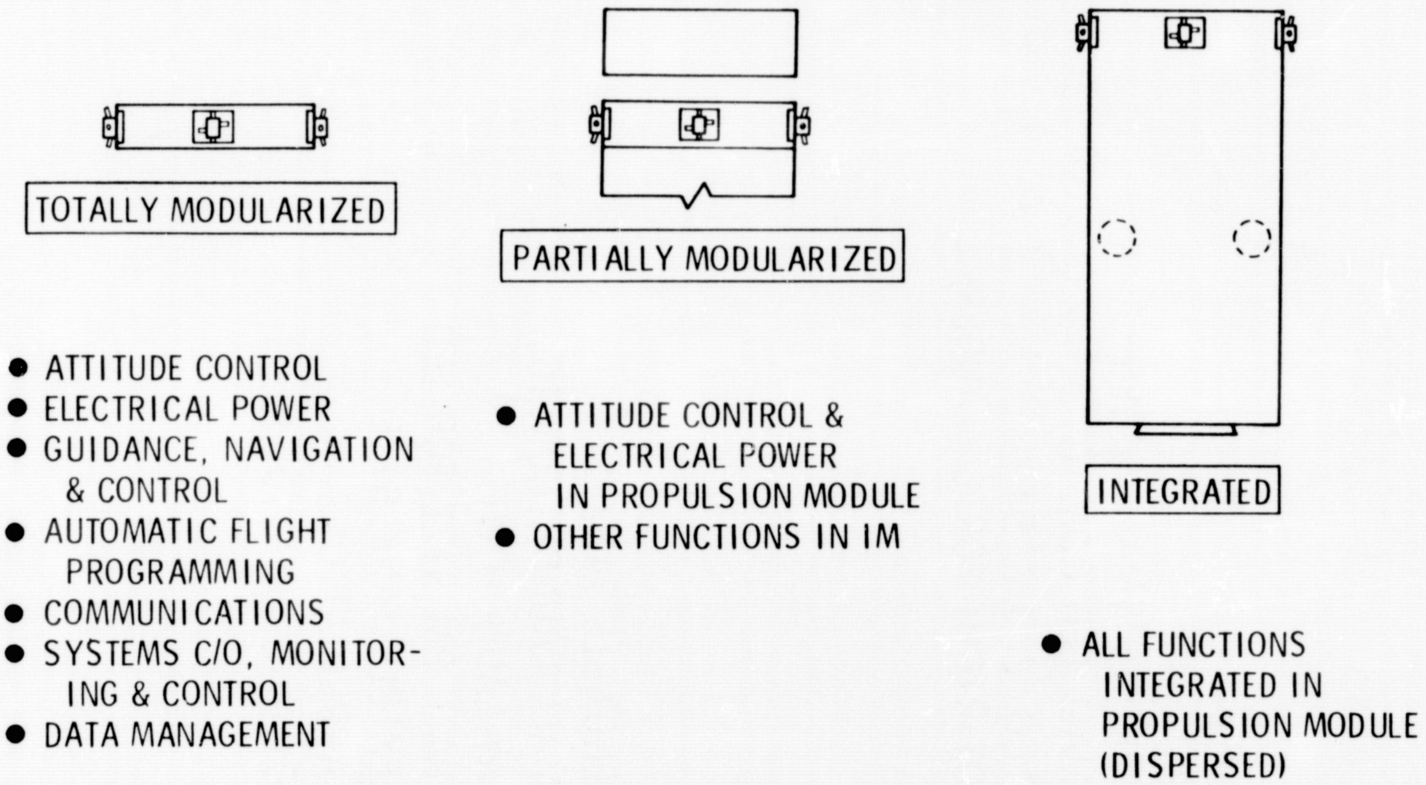


Figure 1-15. Intelligence Module Concepts

15-foot (4.6 m) diameter to permit stacking with other compatible modules in various assemblies for different missions. The module depth would be as required to incorporate necessary subsystem functions. The submodular approach could actually be applied to either of the other two approaches; packages of subsystems could be utilized anywhere in the vehicle or together in an IM framework. In actuality, the modular IM may well evolve with submodules for convenience in manufacture, test, checkout, repair, and alternate application usage. Still another approach is to accumulate the electronic and avionics type equipment together in one package and the electromechanical and fluid-related subsystems (fuel-cell electric power, attitude control, etc.) within the PM close to the common propellant supply. In this way, the propulsion-related components and the electronic-type components could be separately manufactured, qualified, checked out, and refurbished along common related technology lines.

Likewise, crew modules of 12-, 15-, and 22-foot (3.7, 4.6, and 6.7 m) diameter were examined for weight-volume relationships (Figure 1-16). Again, the 15-foot (4.6 m) diameter compatible with the EOS cargo bay was found to be a good compromise size in a single vertical cylinder and provides

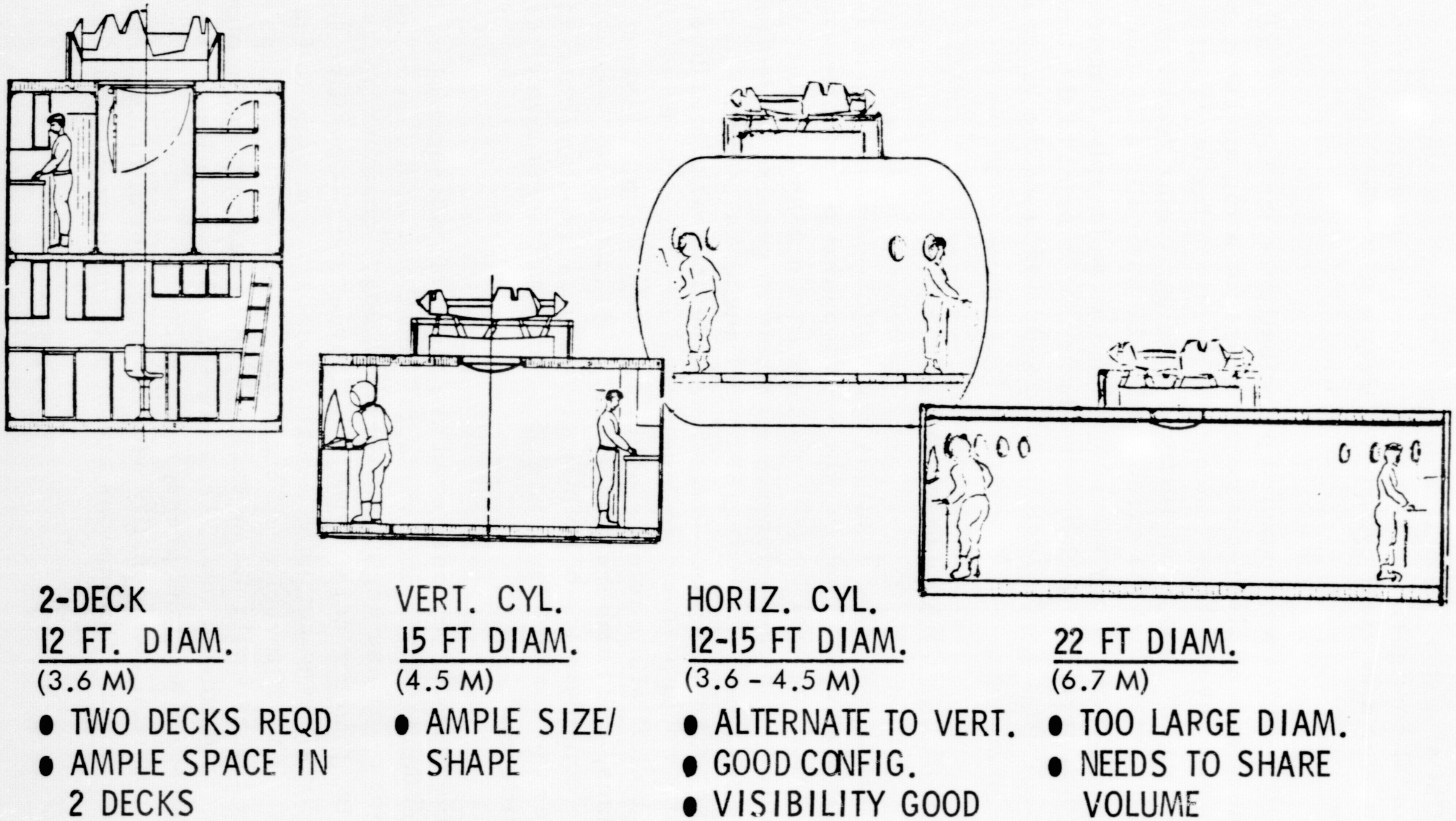


Figure 1-16. Alternate Crew Module Sizing Arrangements

good workspace, adequate crew volume per man, and a reasonable shape for modular assembly stacking (Figure 1-16). Horizontal cylinder crew modules (CM's) also were evaluated and compared to vertical cylinders. The horizontal configuration was investigated to determine its relative advantages over a vertical cylindrical segment for the lunar surface shelter application.

Cargo modules were studied principally for the lunar landing mission, because there was no clear-cut requirement for a tug module for use in other missions beyond those concepts currently being investigated in the EOS, EOSS, SOAR, and RAM studies. The particular requirements of the lunar lander dictate that the cargo module be basically a 15-foot (4. 6 m) cylinder for EOS cargo bay transport, which can be split longitudinally into halves (or quarters) for subsequent attachment in orbit to the lander PM. In this way, a low cg and excellent access to and from the lunar surface is assured.

1. 3. 3 DESIGN ISSUES AND CONSIDERATIONS

PM issues and considerations are summarized in Figure 1-17. In the structural design of the vehicle, the relationships of the meteoroid protection provision, the structural load-bearing shell, the tank structure and supports, and the multilayer high-performance insulation require tradeoff iterations. Truss-type structure with integral propellant tanks as well as continuous skin-stringer structure are candidates. One approach is to determine the concepts that would perform adequately and have acceptable cryogenic propellant boiloff with minimum overall concept weight. Various tankage shapes, sizes, and numbers can be analyzed, and the tankage arrangement can include consideration of both longitudinal cg location (important for lunar landers or for determining RCS jet moment arms under certain failure mode conditions).

Engine types to be considered are those of conventional bell-type nozzles, including retractable variants, and those with varying degrees of advancement regarding engine nozzle technology, chamber pressure, area ratio effects, prime requirements, chilldown, and net positive suction pressure (NPSH) requirements. Engine numbers from one to four were considered. Thurst levels were examined in relation to concept weights so that gravity losses would not be excessive while attempting to minimize weight by keeping thrust low. These factors all have a large affect on engine dimensions and tank pressurization design. Due consideration also is given to potential commonality with the EOS orbital maneuvering system design requirement. Dimensions are important to overall configuration

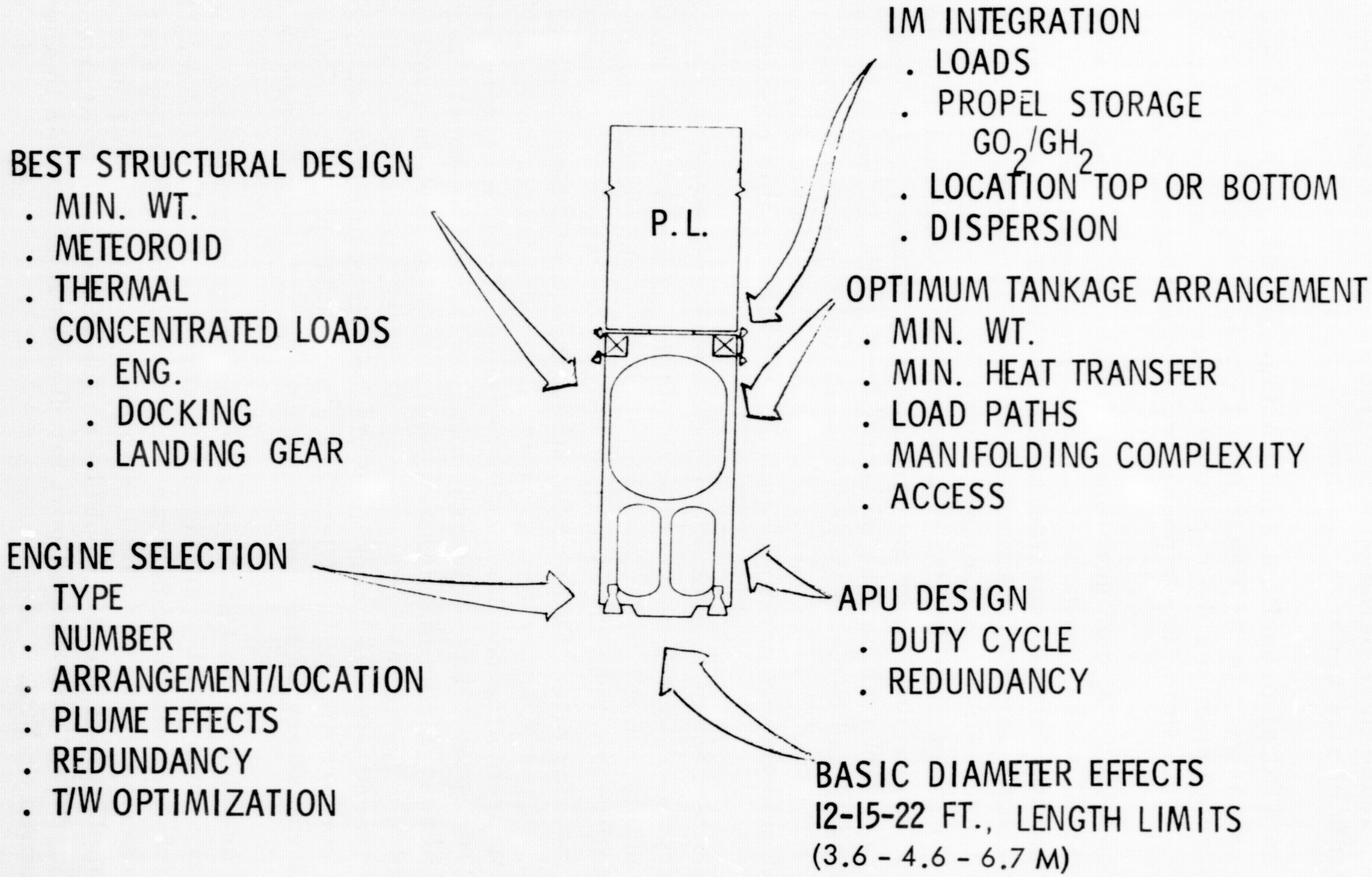


Figure 1-17. PM Issues and Considerations

optimum design. Structural loads, docking provisions, adjacent tankage configuration, and engine type and numbers comprise an interacting set of variables found to be paramount in establishing functional designs within constraints of size and with minimum weight.

The APU or propellant conditioning unit (PCU) concept has been studied extensively in the Phase A and Phase B EOS studies for NASA. The analysis optimizes the combined factors of reserve and redundant capacity, cycle time, usage peaks, vehicle location, etc.

Conceived PM configurations relate to the interfacing requirements for the IM, the other fundamental unit comprising a basic RST. If the IM is completely modular, considerations are the structural interface for its support and the functional interfaces for fluids, gases, and electrical leads (Figure 1-18). As discussed previously, some subsystem elements may be located within the PM to promote short propellant feed lines, common grouping, accessibility, or procurement convenience. The gaseous propellant storage tanks which may be located within the IM must be related to the PCU capacity and cycle response to peak and sustained demands. These and other IM trades and considerations are indicated in Figure 1-19. (Internal subsystems details are discussed later in this report.)

Having chosen 15-foot (4.6 m) diameter, the depth required is as necessary to enclose the subsystems and the gaseous O_2/H_2 auxiliary propellant supply, on the order of 3 feet (0.9 m). Subsystem volumes required may result in a hole in the center of the module which can incorporate and support docking kits. Future RST study phases will be concerned with the type and degree of subsystem grouping into submodules which appears useful for promoting checkout and servicing. For a long-term space-based concept, the possibility of pressurizing the IM will be considered so that manned access during maintenance operations (or even in a mission) could be provided. Locations for antennas, RCS jets, and other bulky equipment must be traded with the weight penalties which result from each set of tradeoff compromises. RCS jets are a particular problem because of their potential variations in size, shape, number, and interference or orientation.

In addition to the CM basic size and shape alternatives, a number of additional issues will be considered (Figure 1-20). The CM may have provisions for accepting landing gear loads so that the resulting scar weights are not carried by the PM in all missions and modes. For manned Earth-orbital applications the CM would be on top of the vehicle to promote visibility, with a pressurized port for direct crew transfer to and from a space station or EOS.

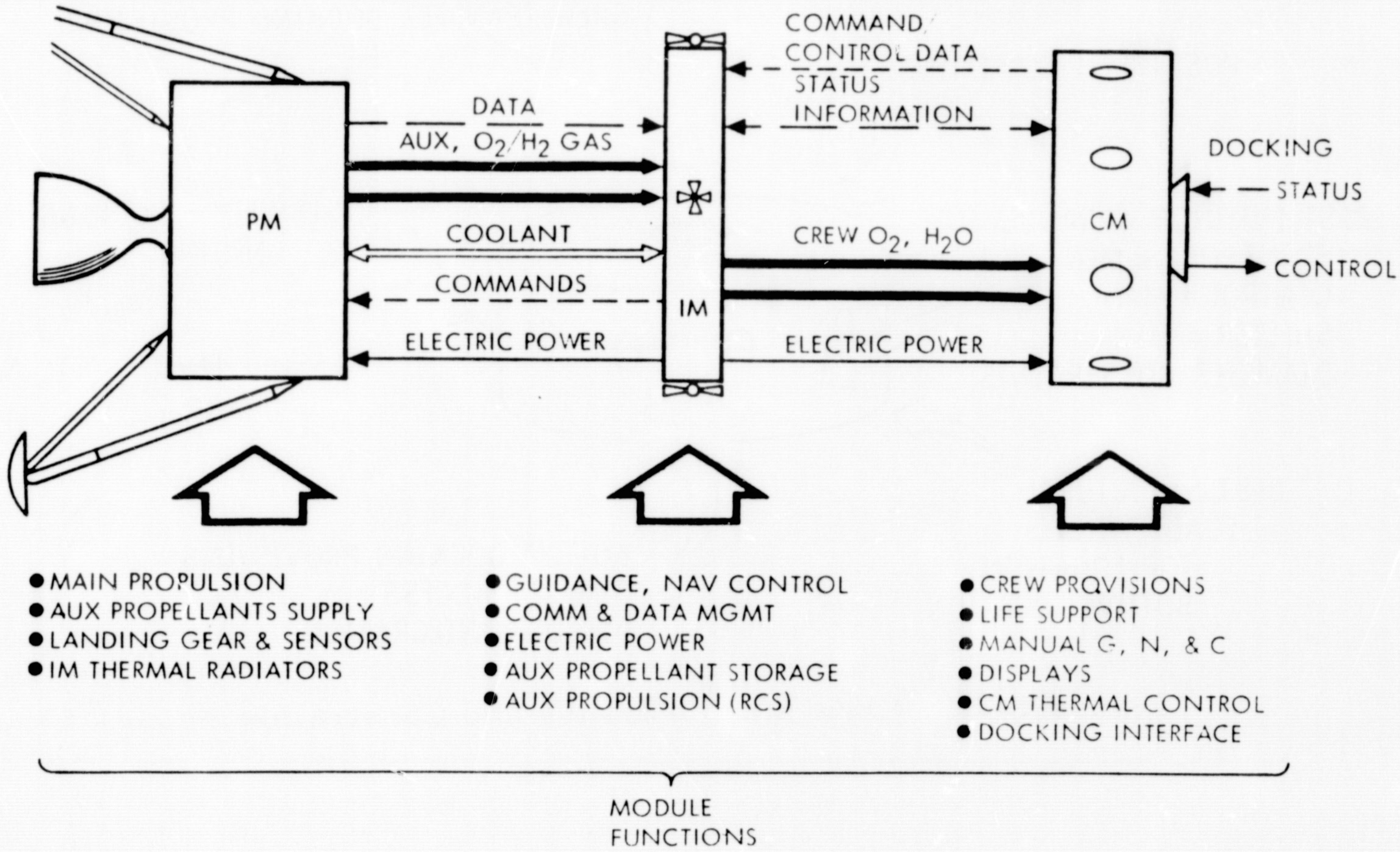


Figure 1-18. Module Functions and Interfaces

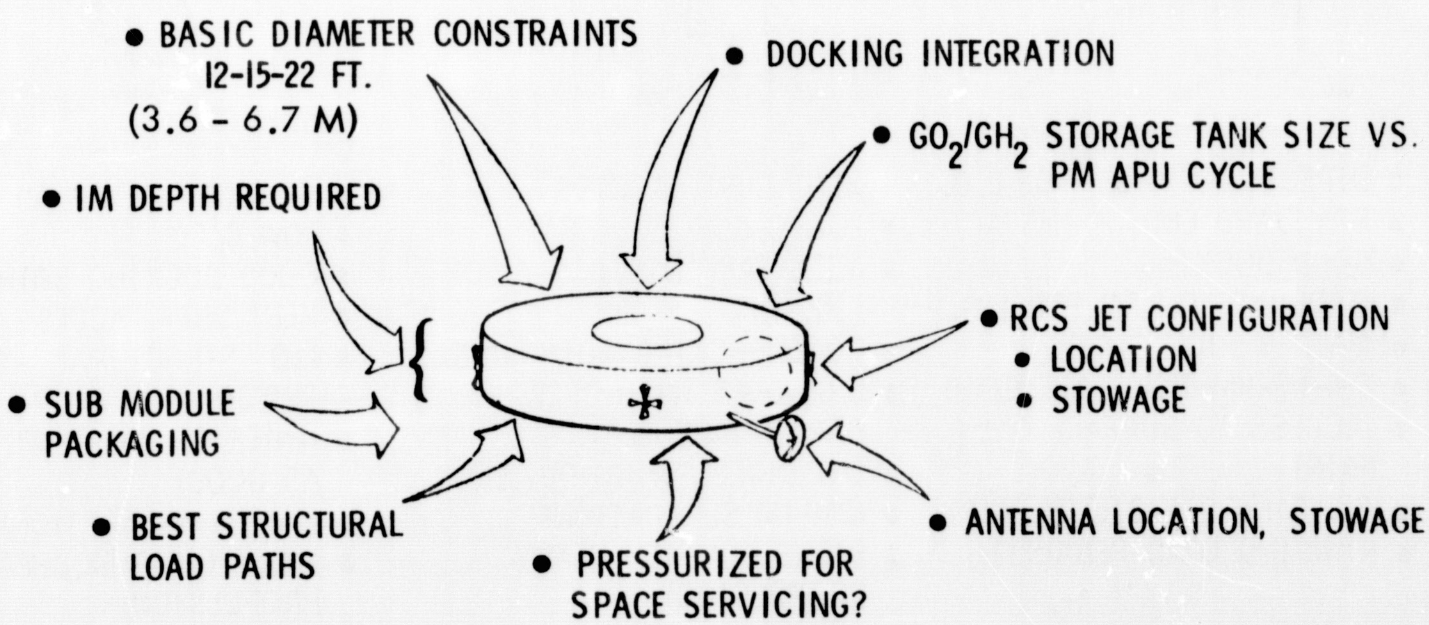


Figure 1-19. IM Tradeoff Studies

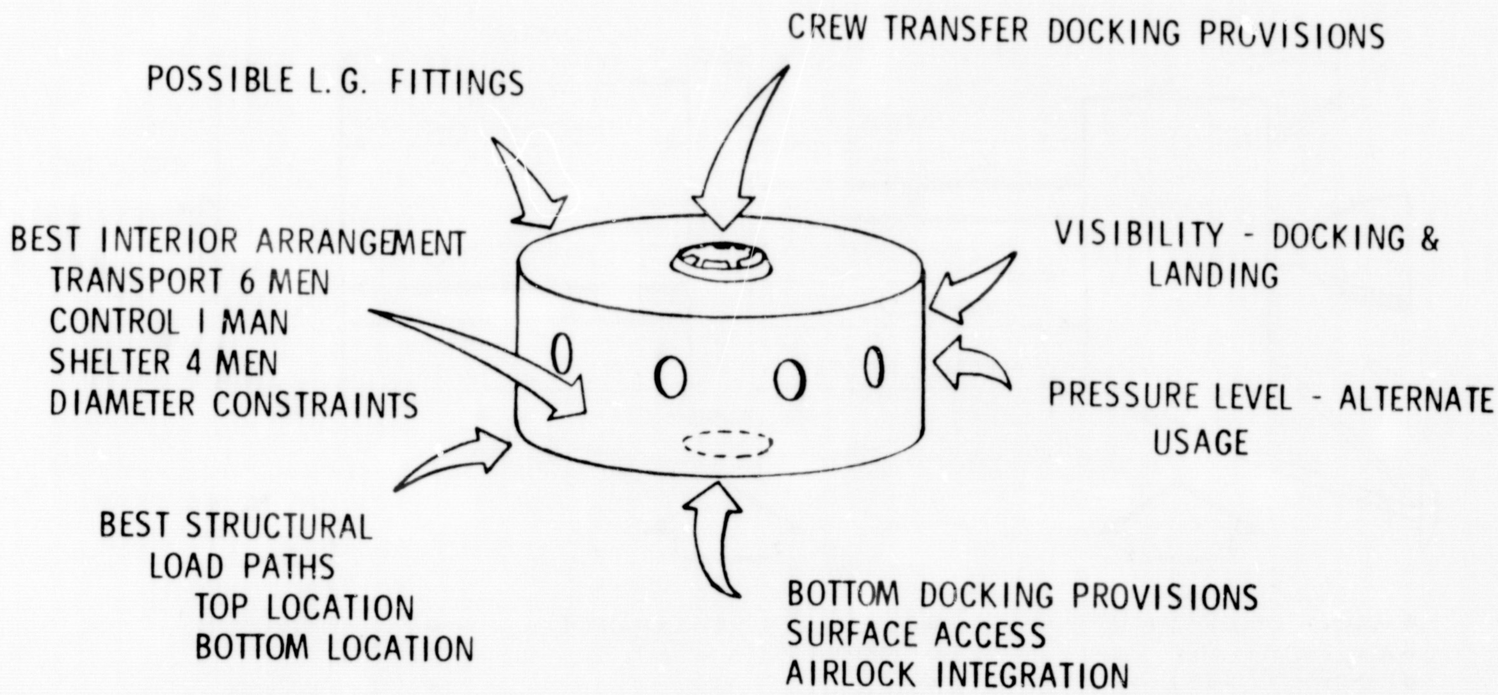
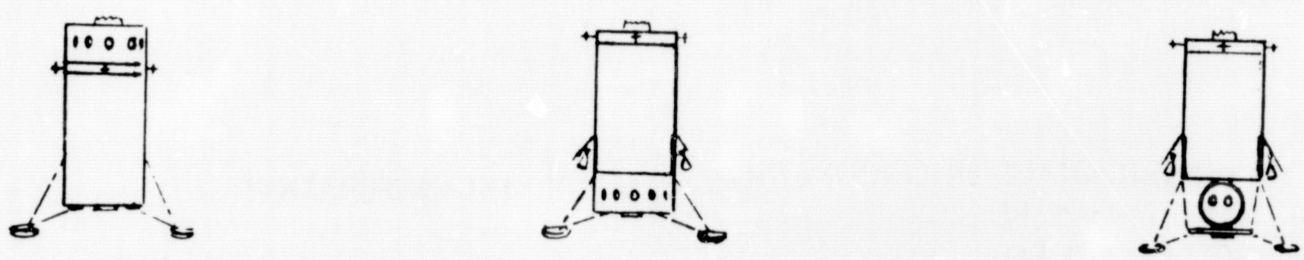


Figure 1-20. Crew Module Issues and Constraints



TOP-MOUNTED CM

- DIRECT EQUIVALENT OF E.O.
- STANDARD ENGINE INSTAL.
- GOOD DOCKING
- (TARE WTS ON PM FOR L.G.)
- (FAIRLY HIGH C.G.)
- CM-IM-PM INTERFACE GOOD
- (ACCESS TO SURFACE ONLY FAIR)
- (LANDING VISIBILITY FAIR)
- (LANDING LOAD PENALTY)

BOTTOM-MOUNTED CM

- XLNT SURF. ACCESS
- LOW C.G.
- MIN. TARE WTS IN PM FOR L.G. KIT
- GOOD DOCKING BOTH ENDS
- L.G. GOES WITH CREW (OR CARGO) MODULE
- (REQUIRES SWING-OUT ENG)
- (CM-IM SEPARATION)
- (MORE COMPLEX HEAVY PROPULSION)

BOTTOM HORIZ CM

- SLIGHTLY BETTER VISIB.
- LOW C.G.
- GOOD DOCKING BOTH ENDS
- XLNT SURF. ACCESS
- L.G. STRUCTURAL INTEGRATION POOR
- (INTERIOR ARRGT NOT AS GOOD)
- (CM-IM SEPARATION)
- (MORE COMPLEX, HEAVY PROPULSION)

Figure 1-21. Lunar Lander Crew Module Integration

In a lunar lander, the CM would be located on the underside of the vehicle to promote landing visibility and to insure easy access and egress to and from the surface during a 28-day stay. Locating this heavy mass on the bottom rather than on the top of the lander stack insures that high inertial loads are not imposed on the PM during landings (Figure 1-21).

The pressure level within the CM could be 5 psia (3.5 N/CM^2) for the lunar surface mission with an O_2-N_2 mixture.

In Earth-orbital missions either a mixed or pure O_2 atmosphere could be utilized at 5 to 10 psia (3.5 to 6.9 N/CM^2). To insure compatibility when docked to a space station in Earth or lunar orbit, a 14.7 psia (10 N/CM^2) pressure level can be used. The underside CM location implies docking provisions on its bottom end to provide for pressurized crew loading or unloading at an orbital station. An airlock is required for lunar surface operations and could incorporate pump down provisions to counterbalance the internal atmosphere losses for missions of extended duration. Internal provisions can include a 2-man control station position (which one man could operate). Controls and displays would be common to a degree with those of the EOSS. Visibility for Earth-orbital and for lunar landing missions is needed. The lunar lander would probably require special inclined windows to promote observation of surface conditions during touchdown. As noted previously, horizontally-oriented versions of the CM can promise advantages, particularly for the manned lunar landing mission. However, integrating such a shape into a vehicle stack does pose a problem in regard to interfaces and load paths, and this consideration must be traded against the potential gains as a shape for a lunar shelter.

In considering cargo modules (CAM), the mission considerations must be traded with the problems of transport, attachment to a tug, loading, unloading, and operations required. Space station logistics studies conducted separately were centered on cargo modules, which are of 14 to 15 feet (4.3 to 4.6 m) in diameter for transport in the EOS cargo bay. Use of these modules with the tug appears feasible rather than requiring a special module as part of a tug program. For lunar landing missions, the constraints on lateral and vertical cg position considerations of access from the surface tend to require special modules or pods, such as full cylinders, halves, or quarters of a cylinder less than 15 feet (4.6 m) in diameter. These modules can be assembled to the outside of the lander as opposed to clusters of PM's about a central cargo core module. General vehicle complexity in these cases is an important consideration; another consideration is whether or not the CAM is the driver in establishing the total vehicle concept as in the case of multiple PM's clustered around central cargo.

Kits for special purpose functions can be considered as mission-oriented equipment in some cases or as add-on features to a basically simple tug. The lunar landing missions in particular require a number of features not necessarily present in other mission applications. Landing gear must be attached in such a way as to distribute concentrated loads from struts into areas of existing heavy structure if possible. A plug-on kit concept might be attached to the CM aft docking port and to existing structure while possibly utilizing another tug for orbital assembly operations. Extra radiator area, horizontally oriented, can be attached at the top of the vehicle and additional antennas can be used if desired to communicate to Earth directly at high data rates.

The approach for some special mission equipment, such as search and acquisition radar, particular docking configurations, manipulators or special maintenance and retrieval grapplers, is to bolt on these items to a basic tug (with provisions) or the special equipment can be incorporated into submodules. The latter can be docked to a standard tug with little if any special tug preparation by utilizing the regular docking ports.

A major tradeoff consideration then is whether to (or how much to) permit lunar mission special requirements to affect the design and particularly scar weights of the basic tug. Adequate provisions in the original design would make the later transition to a lunar lander configuration easier, but this consideration must be traded against the cost; heavier inert weights would seriously impact the propellant sizing for the high-energy geosynchronous mission. An alternative approach that may be viable considering the uncertainty in lunar mission planning is the one selected herein to avoid such scar provisions and to accommodate lunar landing mission requirements by a block change in the RST program, somewhat analogous to the Block I and Block II Apollo CSM change. Among the changed items could be the engine modifications for lander throttling (about 8 to 1 or 10 to 1), possible change to swing-out type engines to clear a low mounting of the CM, addition of cargo pods and provisions, lunar landing gear kit, extra lunar radiator kit, and changes to the CM arrangement for 4-man, 28-day shelter use.

2.0 CONFIGURATION AND CONCEPTS ANALYSIS

2.1 SCOPE OF ACTIVITY

2.1.1 CONCEPTUAL DEVELOPMENT STUDIES

Initial Parametric Conceptual Design

As noted in the Conceptual Design logic in the preceding section, the initial activity involved the broad exploration of basic modules, multipurpose space-based vehicle arrangements, equipment, and the important interfacing systems. Preliminary subsystem characteristics and estimates of PM sizing (propellant quantities required) were used to permit early exploratory configuration concept layouts. Different concepts used varying propellant quantities as a result of different assumptions as to mission and payload design objectives or basis for optimization. Since inert and system weight were changing rapidly during this period, propellant quantities were considered parametric within a range. These quantities of O_2/H_2 varied from as little as 25,000 pounds (11,300 kilograms) for an expendable design case to 80,000 pounds (36,300 kilograms) for a fully modularized, single-stage, autonomous, space-based vehicle. Docking, refueling, staging, tankage, engines, equipment arrangement, etc. were explored in these early parametric configuration design studies. Drawings believed to indicate design progress and feasibility are included in this report; some indicate design approaches that probably should not be followed in future studies.

Candidate Concept Designs

As insight was gained into the various possibilities and alternatives, and as results of system and subsystem trade studies were considered, candidate multipurpose design concepts were developed. The concepts were optimized for various primary mission conditions (but also usable for other missions) and they were prepared for the concept selection at the study midterm point (Table 2-1). Design activities are summarized in Table 2-2. In the table a synopsis of all significant design effort is presented, including the design refinement and alternatives studies described in a later section of this report. Those marked with an asterisk are the drawings included in this report.

Table 2-1. Summary of Candidate Multipurpose Approaches

Case	PM Maximum Capacity 1000 lb	Staging Modes*			
		Geosynchronous Mission	Lunar Mission		
			Mode A	Mode B	Mode C and D
1	80 (36,300 kg)	A-1**	A-1	A-1	A-1
2	52 (23,600 kg)	D-1, B-3	A-1**	-	-
3	45 (20,400 kg)	D-1**	C-1, D-1	A-1	A-1
4	41 (18,600 kg)	B-3	-	A-1**	-
5	36 (16,300 kg)	B-3**	C-1, D-1	C-1, D-1	A-1**
6	31 (14,100 kg)	C-4**	B-3**, C-1, D-1	C-1, D-1	C-1, D-1
7	27 (12,200 kg)	A-2**	C-1**, D-1**	C-1, D-1	C-1, D-1
8	23 (10,800 kg)	B-5	-	B-3**	-
9	21 (9,500 kg)	B-5	-	C-1**, D-1**	B-3**, C-1**, D-1**
10	15 (6,800 kg)	B-5**	C-1***	C-1**	C-1***
11	48/8.8 (21,800/4 kg)	E-4**	E-1		

*(A-1) - Single-stage, recovered
(A-2) - Single-stage, expended
(B-3) - Two-stage, both recovered
(B-5) - Two-stage, second expended
(C-1) - Two-propulsion modules, one IM, parallel operation
(C-4) - Two-stage, second stage recovered, IM on second stage only
(D-1) - One propulsion module, one tank set, one IM operating as single stage
**Mission(s) from which concept originated
***Also has tank sets on each propulsion module

Table 2-2. Synopsis of Design Efforts

Item No.	Dwg No.	Design Objectives	Results	Conclusions, Recommendations, and Comments
1	5383-1	Develop representative candidate concepts for lunar shuttle.	Three sheets showing possible 22-ft-dia concepts.	Assumed 22-ft dia necessary. Concept No. 10 appears best. (60,000 pounds of propellants assumed)
2	5383-2	Develop 15-ft lunar shuttle configurations.	Two sheets of possible tankage module arrangements.	Concept No. 3 best; 15-ft dia appears feasible.
3	5383-3	Work out cargo mix for -1 Concept 10.	Typical mix shown for 10,000-lb payload.	Proved that volume adequate for 10,000-lb discretionary payload.
4	5383-4	Study lateral disposal of modules.	Arrangements shown that could be candidates.	One with two PM's and central stack of crew and cargo modules appears reasonable.
5	5383-5	Prepare matrix of tug applications.	Small drawings of multiple concepts in 21 missions modes.	Large number of possibilities to be used in operational planning studies.
6	5383-6	Develop configurations for maximum geosynchronous performance.	Three concepts shown—different tankage, 15-ft dia.	Three concepts showed possibilities and fit within EOS, including payload for USAF.
7	5383-7	Study layouts for IM arrangement.	Integrated and modular concepts shown, 15, 22-ft diameters.	15-ft dia appears to use space better.
8	5383-8	Show lunar lander with cargo pods.	Quad pods, low CM shown for lander.	Good possiblity for cargo access, low c.g.
9	5383-9	Illustrate expendable geosynchronous concept.	Compact, 15-ft, 22,000-lb propellant concept shown.	
10	5383-10	Illustrate 15-ft manned lander.	Four-man crew shelter, quad pods shown.	Looks good as a candidate concept, building on previous configurations.

Table 2-2. Synopsis of Design Efforts (Cont)

Item No.	Dwg No.	Design Objectives	Results	Conclusions, Recommendations, and Comments
11	5383-11	Conceive LOSS resupply configurations.	Cargo transfer to tug from LOSS shown.	Cargo pods appear to integrate well with LOSS storage and handling provisions.
12	5383-12	Develop ideas for EOS cargo bay integration.	Several possibilities identified for tug in EOS.	Tug mounting and crew transfer feasibility shown.
13	5383-13	Lay out 6-man crew module for Earth orbit.	15-ft-dia crew module arrangement developed.	Four-man shelter module can be reconfigured to good six-man version for short durations.
14	5383-14	Develop two-man minimum Earth orbit tug.	"Mini-tug" concept described.	Practical adaptation of big tug parts using two-man module same size as air lock on 6-man version.
15	5383-15	Develop arrangement of 15-ft lunar lander.	Feasible concept layout made.	15-ft-dia concept appears feasible, four-man with 10,000-lb cargo pods.
16	5383-16	Show summary of other concepts.	Used as illustration of candidates (in proposal).	Feasible arrangements of "families" illustrated.
17	5383-17	Illustrate lunar program build-up.	Physical relationship of systems shown (in proposal).	Graphic picture of operations will permit time-line analysis.
18	5383-18	General arrangement—60,000-lb tug with four engines.	Lunar and orbital configurations developed—engine position change.	Good configuration concept if permissible to move engines between lunar landing and orbital missions.
19	5383-19	Same with forward cargo module.	Showed feasibility of alignment for EOS cargo bay.	Cargo modules can be carried in line in EOS, deployed in space while connected.
20	5383-20	General arrangement—side-mounted mission module	Showed possibility of lateral crew, cargo modules.	C.g. shift accommodation bad, other characteristics good.

Table 2-2. Synopsis of Design Efforts (Cont)

Item No.	Dwg No.	Design Objectives	Results	Conclusions, Recommendations, and Comments
21	5383-21	General arrangement—horizontal lunar lander.	Illustrated problems with engines, c.g. shift.	Engine complexity and sensitivity to c.g. shift detract from other good qualities.
22	5383-22, 5383-22A	Concept sketches—alternative PM arrangement.	Showed possible candidates with various tankage, PM's.	Small PM's provide diverse possibilities.
23	5383-23	Alternate structural/engine arrangement.	Engine clearances, load paths explored.	Two engines on centerline do not clear CM well.
24	5383-24A	Docking arrangements with multiple PM's.	Feasibility of arrangements shown with horizontal crew module.	Require dual EOS launches, space assembly.
25	5383-25	Configurations translating PM's.	Feasibility study of multiple PM's with dual orientation.	Furnished insight on possibilities for EOS launch and deployment of vehicles.
26	5383-26	Concept matrix chart.	Summary—comments on unusual concepts.	Orderly descriptive material on design studies to date, particularly "unusual" ones.
27	5383-27	Alternate geosynchronous concepts.	Basic two-stage configuration data to fit within EOS, staged, etc.	Supplied understanding of geometry limitations for high-energy mission configuration.
28	5383-28	Alternate geosynchronous tank set concepts.	Explored PM + tank set configurations to fit in EOS.	Showed feasibility of tank set configuration within geometry constraints.
29	2283-1	Identify concept code and list those studied.	Identifies concept coding and those studied to date.	Basic identification chart for concepts studied.
30	2283-2	Provide general arrangement of geosynchronous lunar concept.	Shows concept optimized for geosynchronous adapted for lunar landing.	Geosynchronous optimized requires auxiliary equipment for crew lunar surface access from top of stack.

Space Division
North American Rockwell

Table 2-2. Synopsis of Design Efforts (Cont)

Item No.	Dwg No.	Design Objectives	Results	Conclusions, Recommendations, and Comments
31	2283-3	Compare geosynchronous 22-ft concept versus 15-ft version.	Shows equivalent 22-ft-dia concepts to compare with 15-ft version.	Comparison can be made on directly similar or optimum equivalent basis, 12, 15, 22 feet, to determine effect of diameter on arrangement and concept weight.
32	5383-29	Explore staging length and structure.	Developed interstage philosophy, structural details, engine configuration for minimum length.	Four high P_c engines appear necessary for minimum overall length, short interstage, good ℄ docking.
33	2283-4	Develop lunar lander configuration with 15-ft CM on 22-ft PM, top-mounted.	Good stack, low silhouette, fair visibility.	Concept is feasible, provides good lander configuration.
34	2283-5	Show evolution of concepts based on geosynchronous mission design.	Three evolutionary concepts shown, with fourth having lunar mission feedback.	Lunar mission requirements may have repercussions on original geosynchronous concept, which established basic sizing.
35	2283-6	Develop single-stage lander based on geosynchronous sizing (80,000 pounds).	Design arrangement shown with swing-out engines.	Design permits double-ended docking freedom by eliminating usual engine conflict with crew module and docking port on bottom.
36	2283-7	Illustrate typical tug for use on chemical injection stage studies.	80,000-lb single stage illustrated.	Use as envelope for tug.
37	5383-30	Illustrate Air Force mission vehicles.	Geosynchronous mission vehicle arrangements illustrated; 1-stage, 1-1/2-stage, 2-stage.	Compact arrangements with four engines look good, fit EOS bay.
38	2283-8	Layout basic dimensions of 12-ft-dia modules for 20,000 to 80,000 pounds.	Sizing of 12-ft concepts illustrated.	Feasibility of high-capacity module appears questionable with this diameter.

Table 2-2. Synopsis of Design Efforts (Cont)

Item No.	Dwg No.	Design Objectives	Results	Conclusions, Recommendations, and Comments
39	2283-9	Develop optimum size top CM lunar lander configurations for 12-ft-dia modules.	Single and dual deck CM's shown, 1500- and 900-psia engines, neuter and Apollo docking drogues.	Lander concepts relatively tall and heavy with top-mounted CM; poor lander configuration.
40	2283-10	Explore configuration of 12-ft-dia modules with two paralleled PM's for lunar lander.	Paralleled PM's configuration shown with central two-dock CM over CAM and with cargo pods.	Has functional advantages, low c.g., good ground access, etc., but complex and probably expensive (two PM's, two IM's).
41	2283-11	Refine configuration of basic top CM concept, 15-ft dia, 80,000 pounds and 60,000 pounds of propellant.	Refined configurations developed from 2283-2 configuration with latest RCS, refueling, engines provisions.	"Baseline" concept, forward CM, for 60,000 and 80,000 pounds of propellants
42	2283-12	Refine tug concept with swing-out engines, double-ended docking.	Design refined in 15-ft dia for 80,000 pounds of propellant, (dimensions for 60,000 pounds also noted), latest fueling, RCS, engine size, etc.	Concept appears feasible but swing-out engines incur some weight and complexity penalty; slight I_{sp} losses.
43	2283-13	Explore crew module design in 12/15/22-ft diameters.	Configurations shown with vertical cylinder modules, horizontal cylinder; for 6 men/7 days, 4 men/28 days (lunar), 12-man rescue.	15-ft diameter adequate for all three applications; 12-ft diameter requires two decks; 22-ft diameter too large for CM unless volume is shared.
44	2283-14	Lay out coolant radiator location possibilities for the various vehicle concepts.	Radiator locations shown for 12-, 15-, 22-ft-dia vehicles, Earth orbital and lunar landers.	(Not completed).
45	2283-15	Lay out 22-ft-dia crew module arrangement.	Commodious arrangement shown.	Space could be shared with cargo; might be good for semi-permanent base module.

Table 2-2. Synopsis of Design Efforts (Cont)

Item No.	Dwg No.	Design Objectives	Results	Conclusions, Recommendations, and Comments
46	2283-16	Indicate feasibility of putting some of IM functions (propulsion-related) in PM.	Layouts with alternate locations of ACS/RCS and fuel cells in PM; electronics at top of PM.	Possible to divide IM functions to place propulsion-related items in PM at cost of some module independence. Need check for heat source effects in PM.
47	2283-17B	Define configuration of Concept 3, optimized tank set (D-1) for geosynchronous mission.	Layout developed with alternative configuration for other missions such as low Earth orbit lunar lander.	Lander is very tall for Mode A; good concept for Earth orbit.
48	2283-18B	Define configuration for Concept 4, optimized for NASA Mode B lander.	Layout developed, alternate mission configurations shown.	Mode B lander looks reasonable (top CM shown but not preferred). Would require tank set (similar to Concept 3) for Mode A.
49	2283-19	Define configuration for Concept 5, optimized for two-stage (B-3) geosynchronous mission.	Layout developed with mission alternatives.	Compact vehicle for prime mission. Rather tall lander in tank set (D-1) mode. Would be very small in B-3 mode (2-staged) lander.
50	2283-20	Define configurations for Concept 6, optimized for two-stage (B-3) lander and/or geosynchronous with first stage expended.	Layout and alternatives described.	B-3 (two-stage) lander looks good but operational complexity may not be justified. Small Earth orbital stage is desirable.
51	2283-21	Develop configuration for 15-ft-dia 4-man/28-day lunar shelter.	Layout shows arrangement for 15-ft-dia vertical cylinder CM for lunar surface mission.	Differences from six-man crew transport are small and can be converted either way easily (12-man rescue also feasible).
52	2283-22	Define configuration for Concept 11, optimized for expendable tank set geosynchronous mode.	Layout shows primary and alternate mission mode configurations.	Small PM provides excellent low Earth orbital vehicle. Lander fairly tall with tank set (recovered, D-1 mode).

Table 2-2. Synopsis of Design Efforts (Cont)

Item No.	Dwg No.	Design Objectives	Results	Conclusions, Recommendations, and Comments
53	2283-23	Explore feasibility and features of horizontal PM in lunar lander configuration.	Layout developed for central CM and CAMs at each end, four engines.	More practical design than original concept; better balance, higher reliability, more flexibility. Remaining problems: structural design, buried CM, engine-RCS coupling.
54	2283-24	Lay out tug interfaces with EOSS cargo bay.	Tug installation concept developed for transport to Earth orbit in shuttle.	Tug concepts using their cargo manipulator (one of several EOSS study concepts) appear feasible. Sliding pin joint aft supports shown.
55	2283-25	Define configuration of lunar lander with horizontal cylinder CM.	Layout shown for lander and for alternate missions.	Horizontal CM appears feasible. Somewhat more difficult to integrate; shape mismatches. May be slightly better as CM, but difference not significant at this point.
56	2283-26	Describe "mini-tug" configuration and interface with EOS/EOSS for cargo transfer.	Mini-tug (CM-PM-IM + skirt module) shown; docking interfaces, deployment, alternate transfer modes studied by EOS.	"Mini-tug" looks feasible, uses extra 600 pounds of cryogenic propellants in skirt module with Apollo cryogenic tanks.
57	2283-27	Describe EOS stowage and deployment of lunar landing gear kit.	Landing gear attached to a docking spider; stowed folded, deployed and extended from EOS to dock with base of tug; attach upper legs.	Feasible approach assuming tug has bottom docking provision; requires minimum EVA assembly. Deployment per current baseline EOS cargo deployment mechanism subject to change.
58	2283-28	Layout configurations suitable for unmanned lunar landers with large space station modules as payloads, utilizing Concept 1 80,000-lb PM's.	Dual PM's shown for very large 33-ft-dia station module lander. Also shown are dual PM's and single PM for landing clusters of 15-ft-dia lunar base modules.	Configuration uses platform for landing gear and load distribution. Recovery of unmanned PM's (tugs) possible. Eight lunar base modules (from NASA-MSFC study) or tug CM's can be landed with two 80,000-lb PM's or four with one PM. A four-deck station core module can be landed.

Table 2-2. Synopsis of Design Efforts (Cont)

Item No.	Dwg No.	Design Objectives	Results	Conclusions, Recommendations, and Comments
59	2283-29	Layouts for unmanned large module landers with Concept 5 (37, 000-lb) PM's.	Configuration shown for Concept 5 PM's landing 33-ft-dia and 15-ft-dia base modules.	Lands a two-deck station core module with two PM's of 37, 000 pounds each.
60	2283-30	Layouts for unmanned lunar landers using Concept 11 PM's	Landing configuration for four-deck space station core module, multiple base modules, including integration with landing gear kit.	Landing gear kit can be integrated with support docking structure for PM's or crew module payloads for facile space assembly.
61	2283-31	Alternate Concept layouts—single-stage 60, 000-lb propellants.	Layout incorporate integrated IM functions, dual and single engines, ground or space-based.	Concepts appropriate for reduced requirements; tankage (LOX) arranged for simplicity with one or two engines. Shows potential for reduced weight from baseline configuration.
62	2283-32	Baseline concept layout for Concept 1	Refined layout of basic single-stage geosynchronous-optimized concept for space-based, multi-purpose use	(Supersedes Item No. 41)
63	2283-33	Baseline layout for Concept 5	Refined layout of basic concept for two-stage vehicle.	(Supersedes Item No. 49)
64	2283-34	Baseline layout Concept 11	Refined layout for 1-1/2-stage concept with expended tankage with geosynchronous payload.	(Supersedes Item No. 52)
65	2283-35	Define modular IM configuration for use on three selected concepts.	Modular IM shown with equipment trays and retractable ACS modules. G&N sensors defined.	Sufficient volume in baseline approach for growth. Swing-out ACS modules feasible. Good equipment accessibility. Modular approach feasible.

Table 2-2. Synopsis of Design Efforts (Cont)

Item No.	Dwg No.	Design Objectives	Results	Conclusions, Recommendations, and Comments
66	2283-36	Develop layout of Concept 1A, a lightweight geosynchronous mission design.	Defined layout of special single-stage geosynchronous-optimized concept for ground-based semi-autonomous operation; lightweight subsystems and two engines, nonretractable RCS.	Lightweight Racehorse, integrated structure feasible. Multiple (LOX) tanks and dual engines attractive, but cost some length. No retraction of ACS nozzle assemblies for EOS cargo bay stowage required.
67	2283-37	Develop layout of tug interfaces with EOS cargo bay.	Tug installation concept developed for transport to Earth orbit in shuttle; baseline tug concepts checked for fit.	Tug concepts utilizing EOS manipulator and roller and ratchet end supports feasible. Concepts 1 and 11 accommodated in cargo bay. Concept 5 requires two launches.
68	2283-38	Prepare layout of ground-based expendable concept 7A using EOS OMS tankage and engine.	Defines ground-based single stage geosynchronous-optimized concept for single purpose use with multiple OMS tank concept.	Principle of EOS OMS tanks configurationally feasible. Multiple tanks (LH_2) present operational and insulation difficulties. Heavy construction because of multiple tanks. Subject to considerable change as EOS design evolves.
69	2283-41	Develop layout of ground-based expendable concept 7B using O_2/H_2	Defines ground based single-stage expendable geosynchronous-optimized concept for single purpose use; simple design.	Integrated structural approach feasible. Single tanks and engine give lightweight vehicle. ACS pods not required to retract to fit within EOS cargo bag. Short stage of 27,000 pounds of propellant.
70	2283-42	Develop layout of ground-based expended storable propellant concept.	Define ground-based single stage geosynchronous-optimized concept for single purpose use; 46,000 pounds of storable propellants; simple design.	Integrated structural approach feasible. Lightweight and short stage. Small size attractive and possibly cost competitive with cryogenic stage. Alternate multiple tanks shorter but nonintegrated structure. Within EOS cargo bay envelope without retraction of ACS pods.

2.1.2 SUPPORTING SYSTEMS AND TECHNOLOGY STUDIES

The scope of the systems and technology studies supporting the conceptual design effort (and subsequent refinement studies) is summarized in Table 2-3 with results enclosed as Appendixes A through F. These studies do not include the subsystems, which are described in a separate volume of this report, but do include the following:

Appendix A: Integrated Structures - design of structure in which the related design of meteoroid protection and insulation is considered.

Appendix B: Reliability - Evaluation of system design for mission success and crew safety; single-point failure analysis and justification

Appendix C: Weight - Parametric relationships for modules and concepts

Appendix D: Environmental Protection - Evaluation of meteoroid protection in space operations

Appendix E: Propulsion - Engine optimization and propulsion system analysis

Appendix F: Cryogenic Storage.

Table 2-3. Conceptual Design Analyses/Studies

- Integrated structures
  - Insulation
  - Meteoroid protection
  - Structural loads
- Reliability
  - Mission success
  - Crew safety
  - Single-point failures/justification
- Weights
  - Parametric designs
  - Modules
  - Vehicle concepts - baseline
  - Alternate/special concepts
- Conceptual designs
  - Parametric sizing studies
  - Module arrangements
    - PM
    - IM
    - CM
  - Candidate configurations
  - Three baseline refined designs
  - Alternate/special concepts
    - Ground-based
    - Lightweight
    - Horizontal lander

2.2 CONCEPT SYNTHESIS AND DEVELOPMENT

2.2.1 EARLY 15-FOOT (4.57-METER) DIAMETER CONFIGURATIONS

Basic Configuration

A large number of layouts were prepared in the exploration of tug configurations. They eventually led to development of the midterm candidate designs. The ones described in this section are considered particularly interesting or significant in tug concept evolution. In the configuration shown in Figure 2-1 emphasis is placed on the Earth and lunar orbital operations. The vehicle is configured with a forward CM and an aft IM, which surrounds the two main engines. Two RL-10-sized engines are used for the main propulsion, and a propellant tank configuration consisting of a single LH_2 tank and three LOX tanks has been used. The IM is mounted aft of the LOX section of the PM. The helium pressurization tanks for the main propulsion system are located near the LH_2 and LOX tank forward bulkheads, respectively. They have therefore been removed from the central area. In place of the helium tanks in the central area of the IM, a slotted area of clearance has been provided for the forward portion of the RL-10 engines. The slotted area allows more room within the IM for ACS equipment and electronics than the circular area previously provided when accommodating helium tanks.

Emphasis on the orbital operations mode established the requirement for the forward-located CM. This allows the crew to view payloads that may be attached to the forward end. The CM uses a cylindrical air lock about the longitudinal axis. The air lock inside diameter is 50 inches (1.3 meters) and terminates at the forward end in a pressure hatch. At the forward end of the CM is a neuter cone docking system. This system makes the tug compatible with space station docking and allows easy egress from the CM through the docking system. The CM offers a maximum of usable volume with the centrally located cylindrical air lock. The module is constructed of honeycomb sidewalls and bulkheads. Counter-top work stations with overhead storage provisions are typical throughout the module. The control station is similar to that of lunar module, with the standup position and overhead viewing window ports for docking and payload observation.

The concept shown features a single LH_2 tank and three LOX tanks. The LOX tanks are cloverleaf arranged, with the LH_2 transfer line passing through the center of the cluster. The LOX tanks transfer to a manifold, which feeds the engines. The PM structure consists of a honeycomb cylinder with

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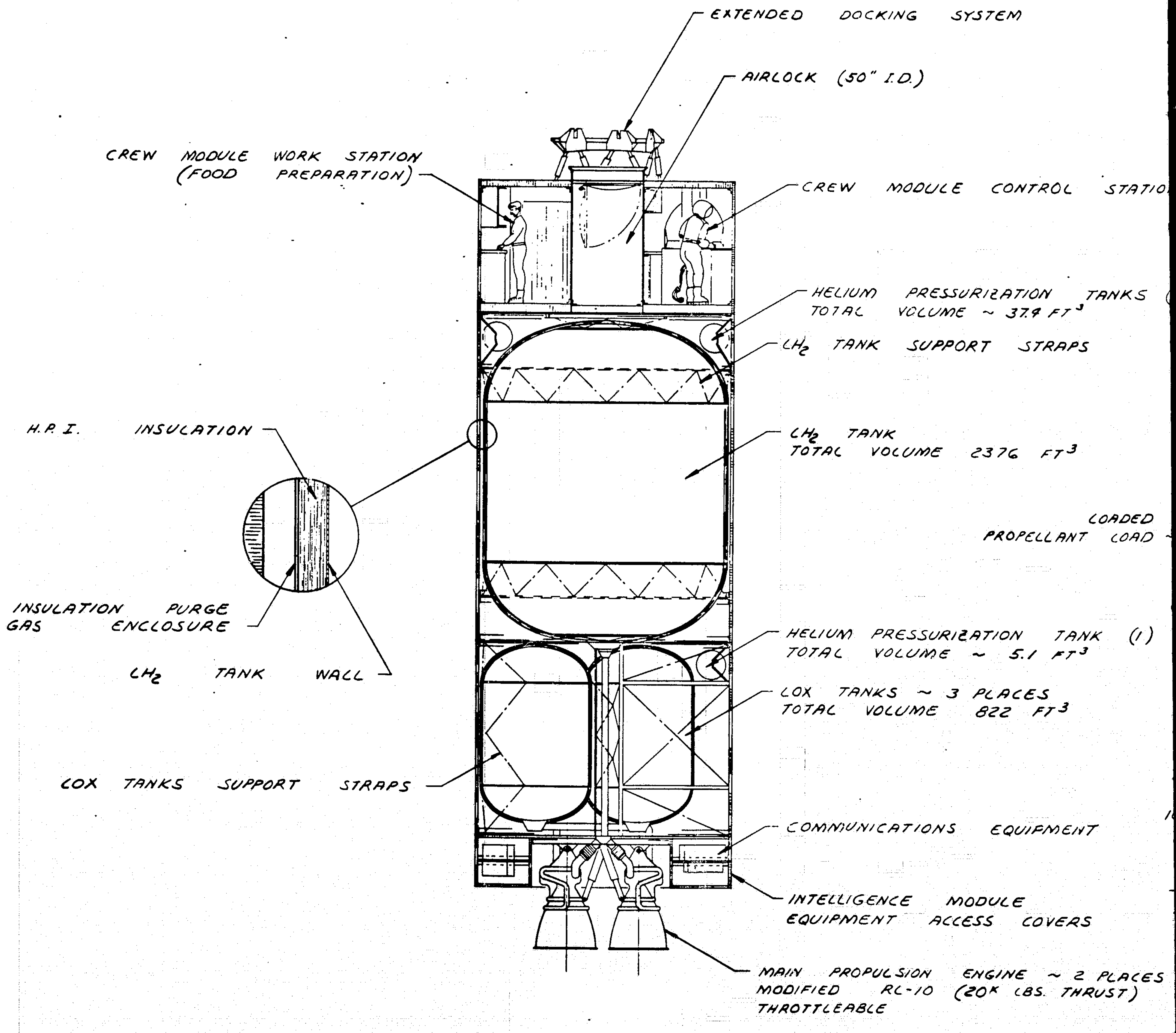

SECTION A - A

ORBITAL OPERATIONS CONFIGU

TOTAL PROPELLANT LOAD 60,000

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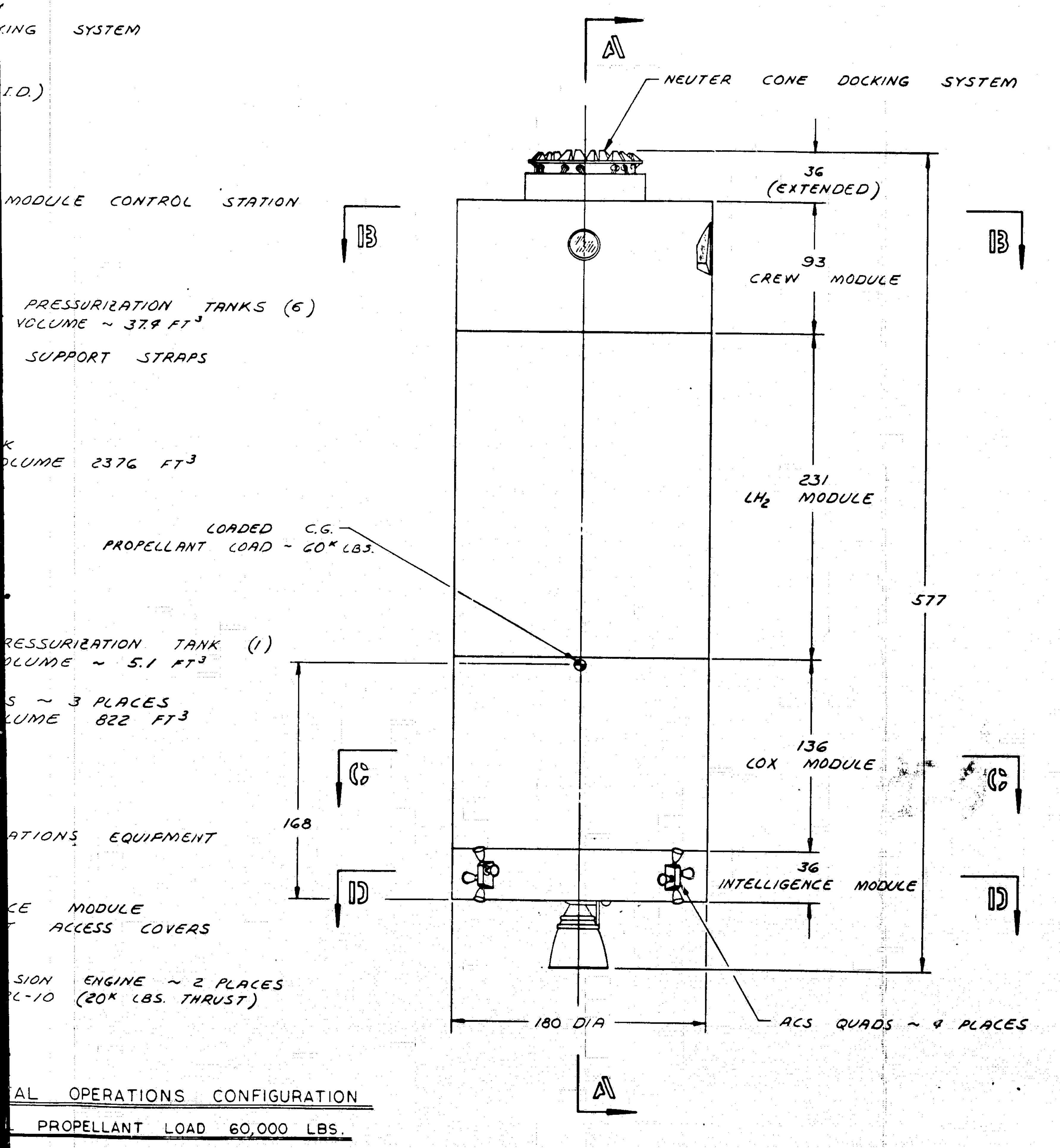

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Space Division
North American Rockwell

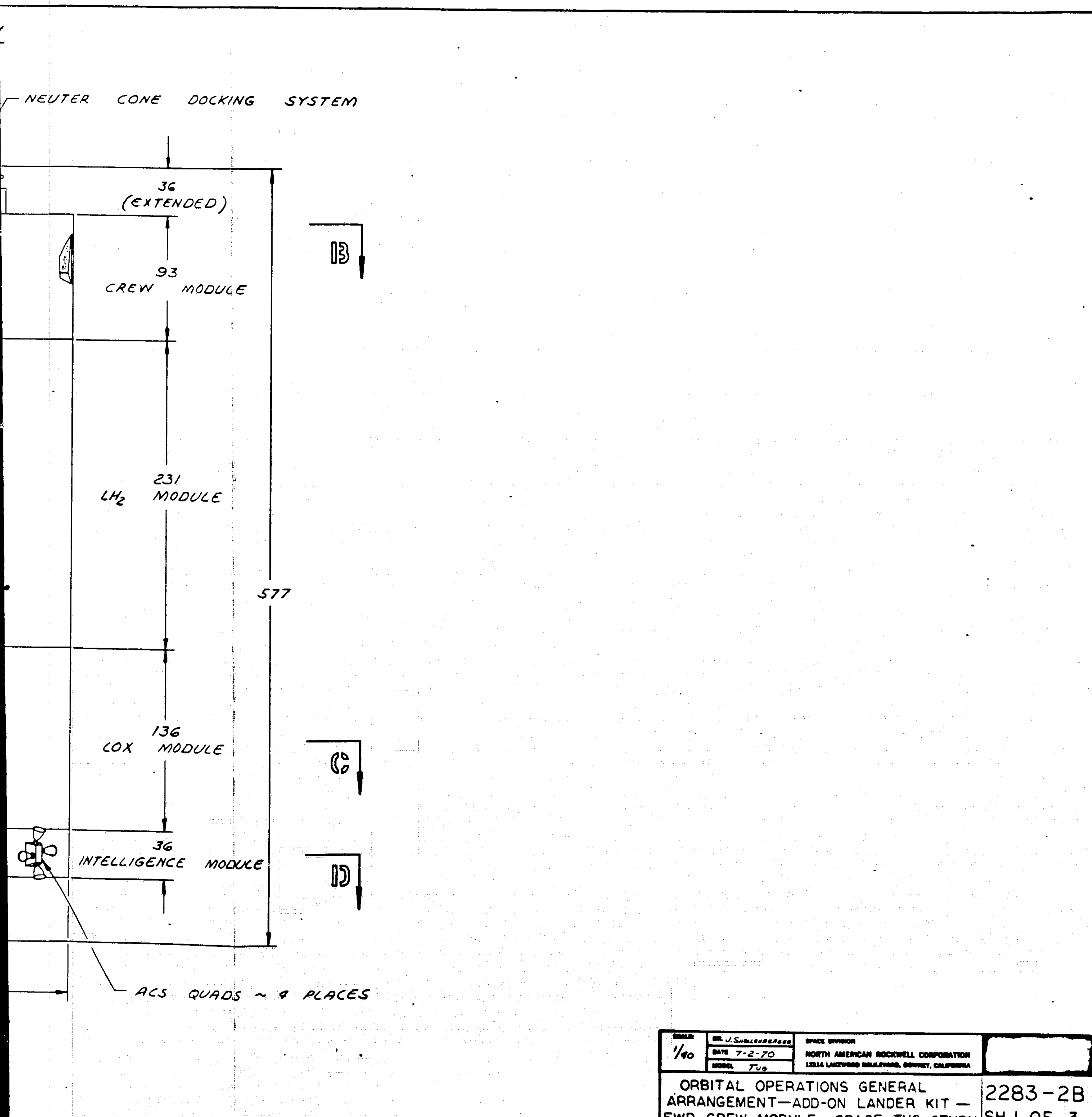

Figure 2-1. Add-On Lander Kit - Forward Crew Module Orbital Operations General Arrangement (Sheet 1 of 3)

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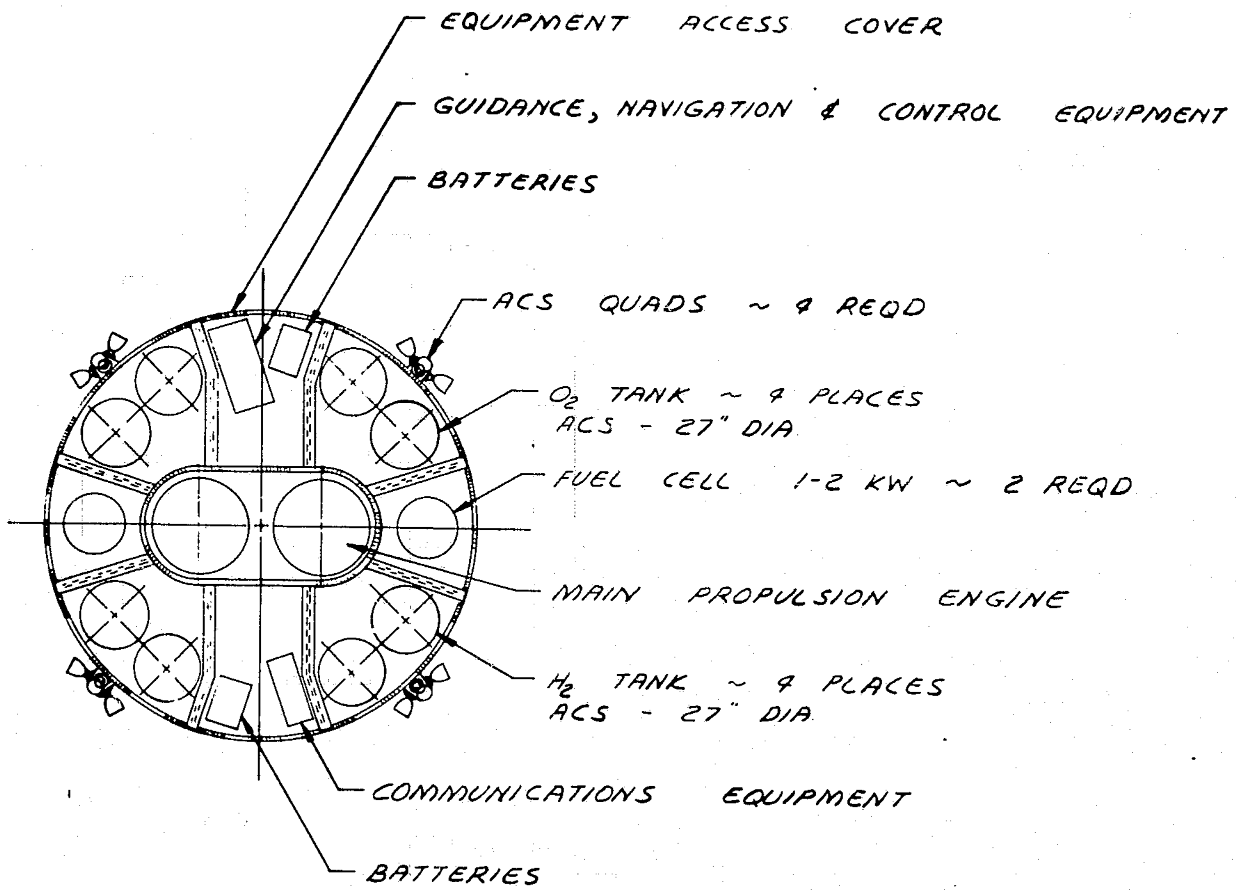
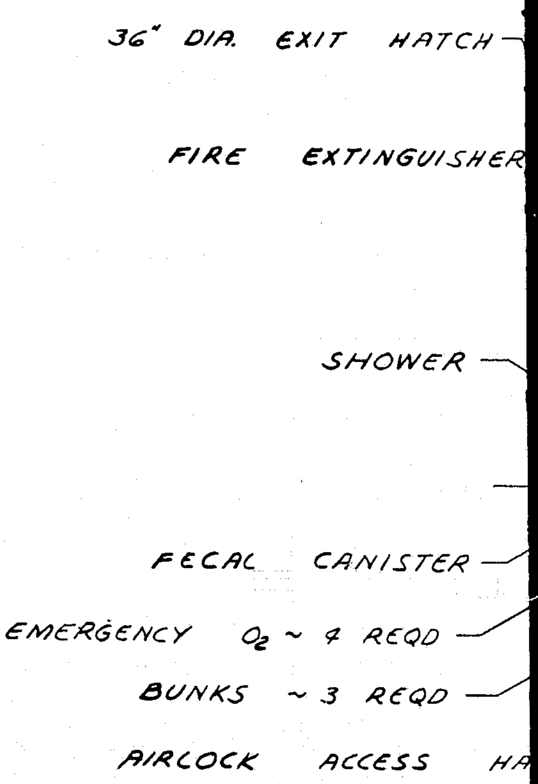
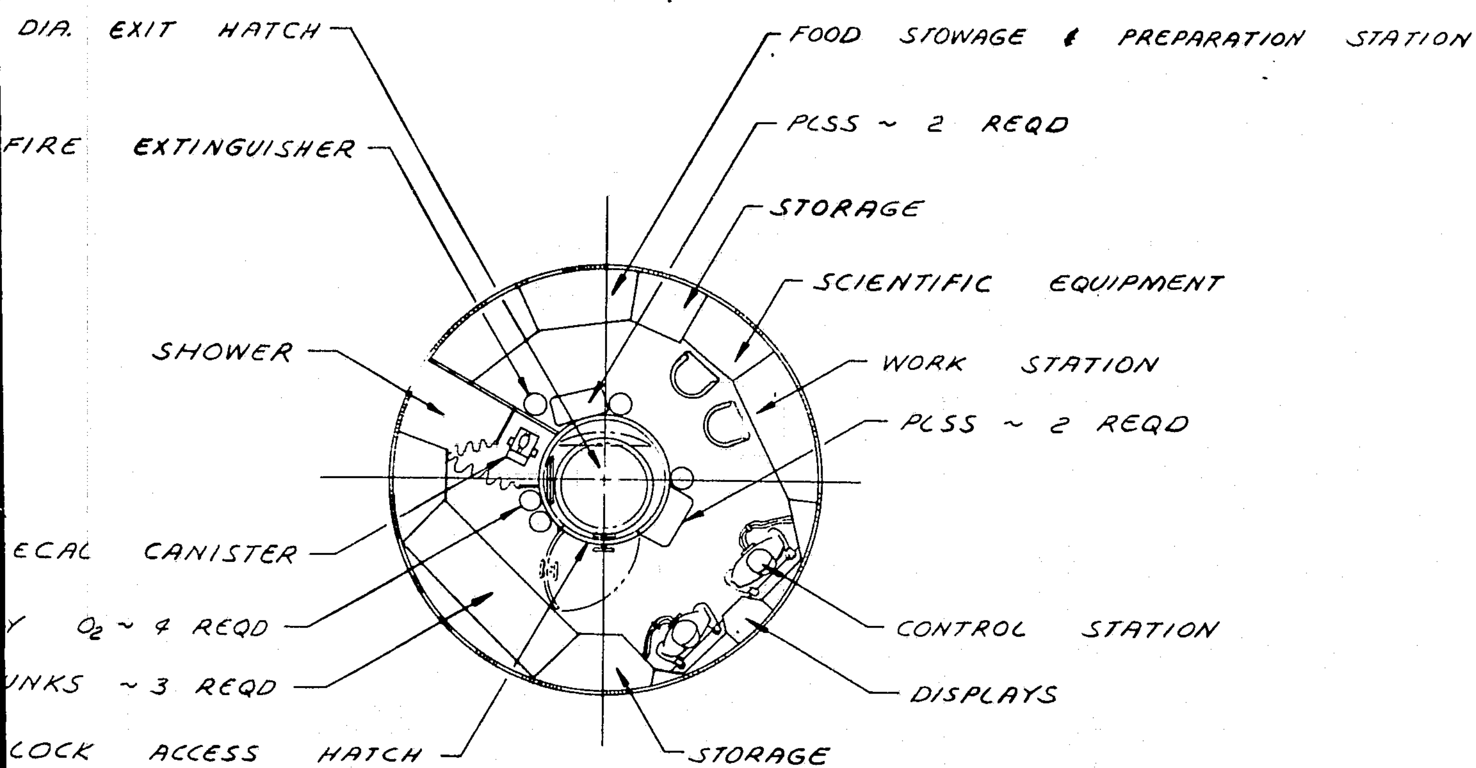


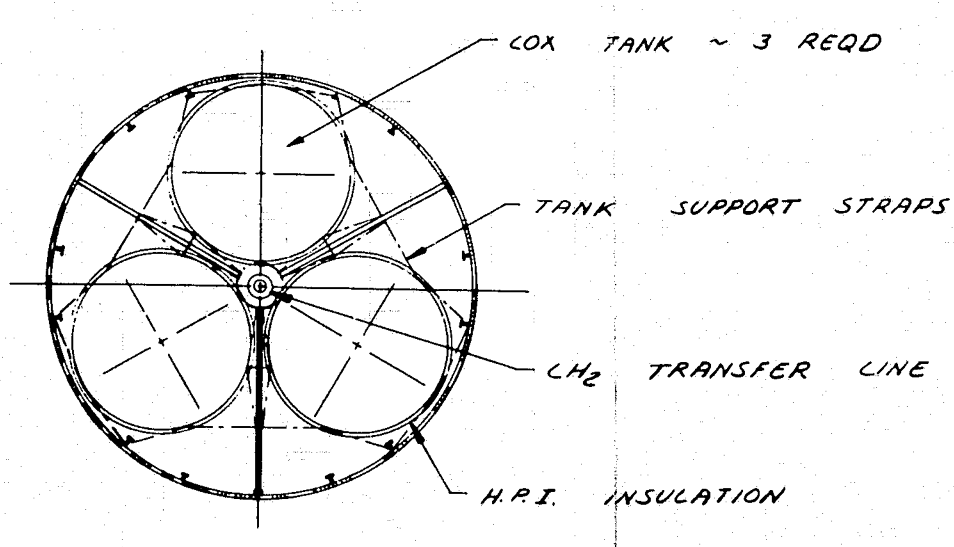
SECTION D - D

INTELLIGENCE MODULE ARRANGEMENT





SECTION B - B
CREW MODULE ARRANGEMENT



SECTION C - C
LOX TANK ARRANGEMENT

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SHEET 2 OF 3

Figure 2-1. Add-On Lander Kit - Forward Crew Module Orbital Operations General Arrangement (Sheet 2 of 3)

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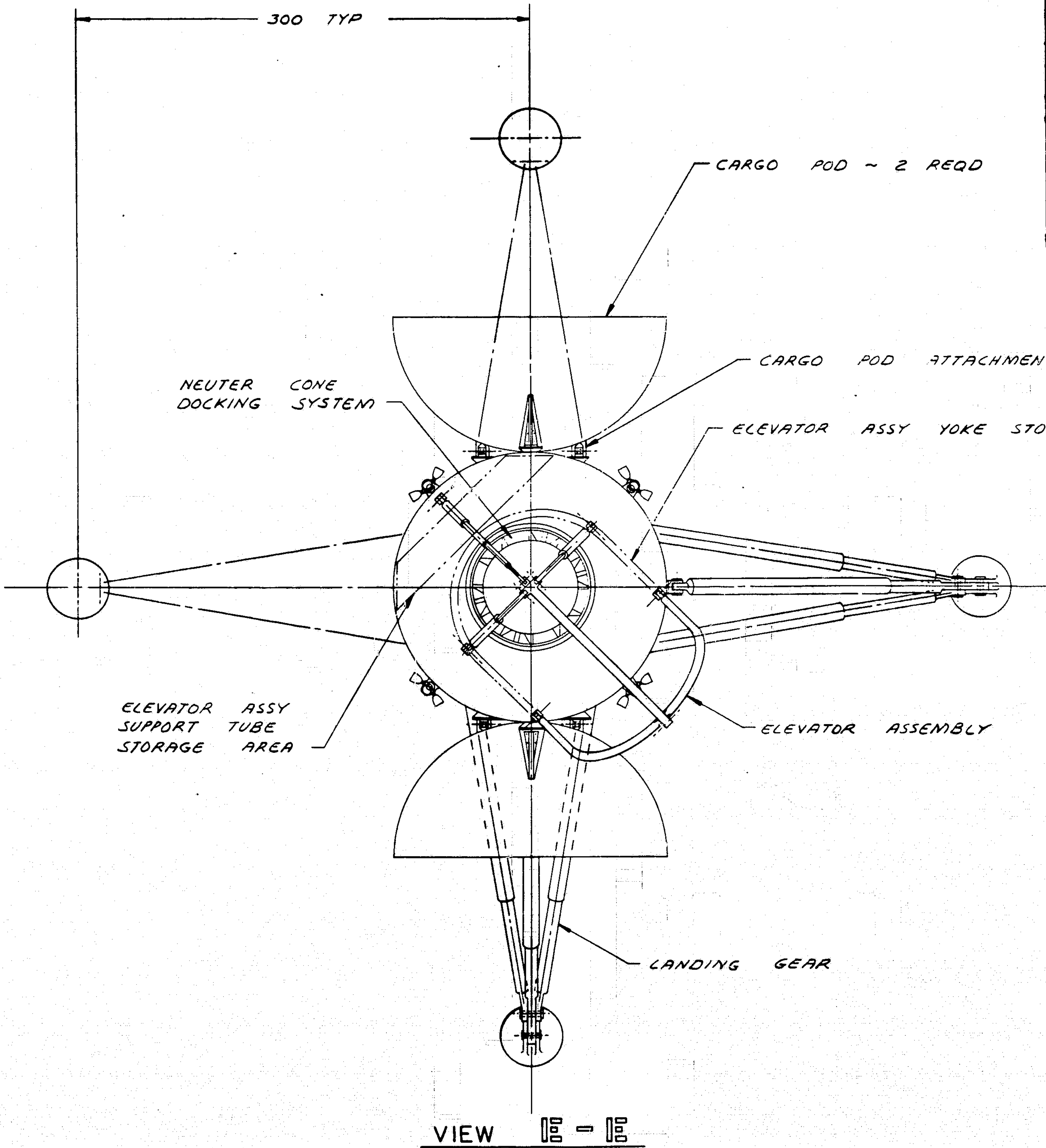

VIEW E - E

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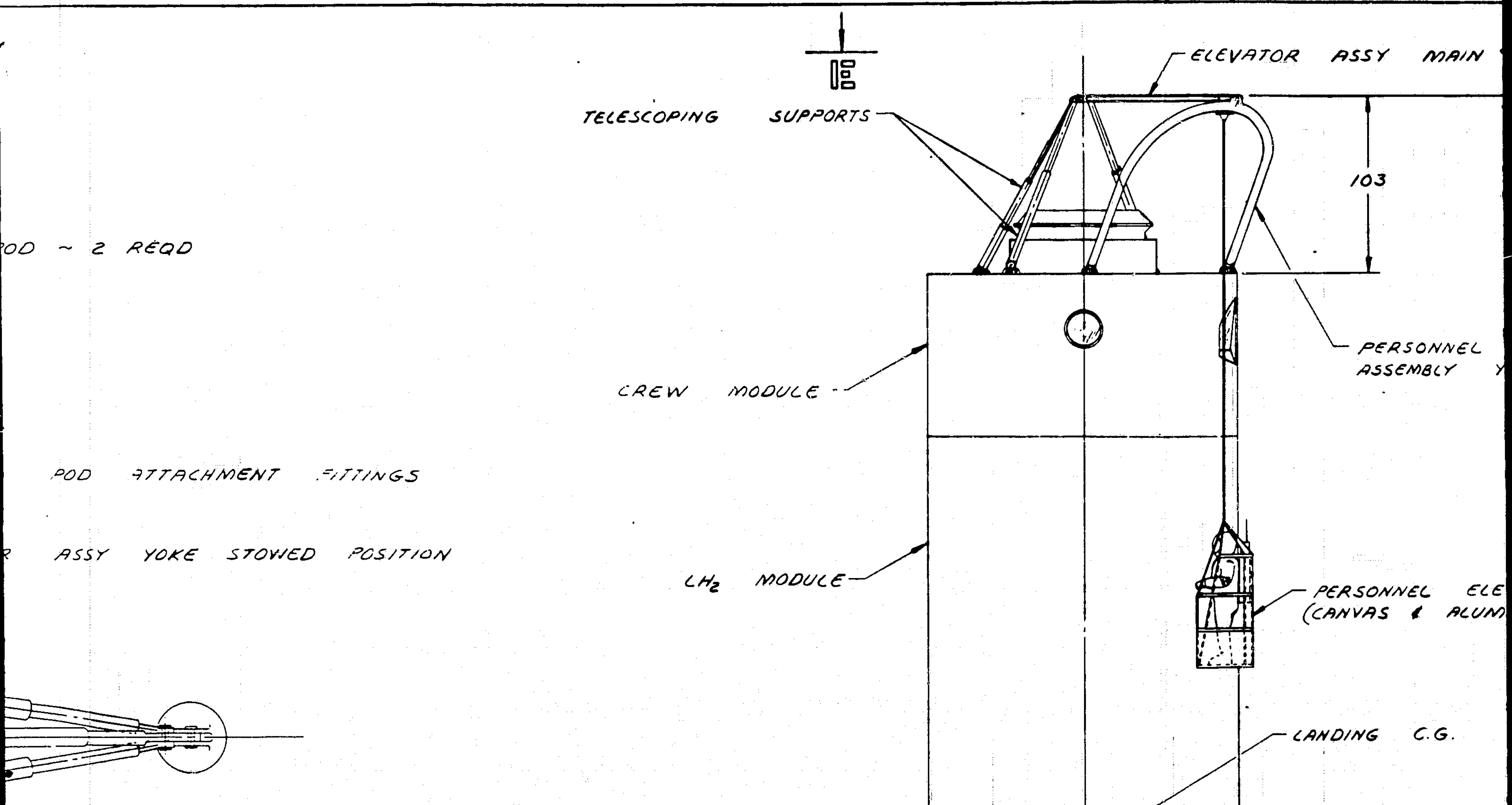

Figure 2-1. Add-O
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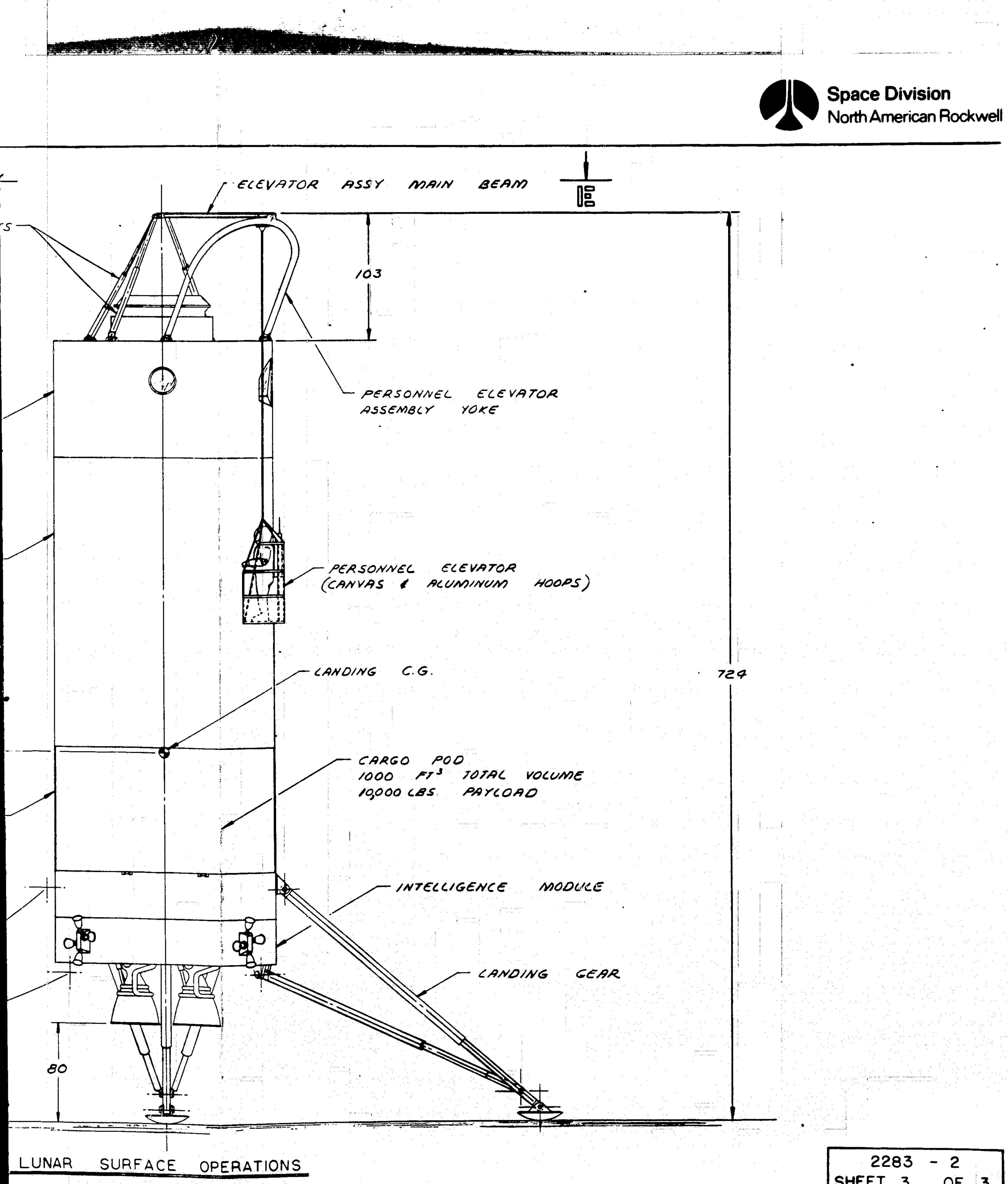


Figure 2-1. Add-On Lander Kit - Forward Crew Module Orbital Operations General Arrangement (Sheet 3 of 3)

stiffening longerons and girth rings. The hydrogen tank is supported from the girth rings. The oxygen tanks are supported from three radial beam-like structures in the LOX section.

The IM is built up of eight sections. Four of the sections are RCS quads with tankage and all associated hardware. Each section is removable from the main structure to facilitate manufacture, assembly, test, and checkout. The RCS tanks and associated valves, filters, transducers, etc., are mounted on a honeycomb shelf that attaches to the beams. A removable honeycomb panel on the outside of each section completes the structure. Surrounding the two engines is an oblong cylindrical honeycomb structure to which the beams and shelves are attached. Two of the remaining sections house the electrical power fuel cells. The remaining two sections accommodate the other subsystem equipment. The communications, guidance and navigation, electrical power and other subsystem equipment is located on the shelves in these sections.

Lunar Adaptation

For the lunar landing operation, an add-on landing gear kit has been provided. (Prior exploratory layouts revealed that stowage and deployment of folded gear constitutes a formidable problem). The landing gear attaches to the bottom of the IM and to a girth ring on the LOX section of the PM. The overall stacking arrangement of th vehicle has not been changed to accommodate the landing mode. The landing gear is such that at touchdown the main propulsion engine nozzles are at least two exit diameters from the lunar surface. Cargo pods (hemipods) are attached to the vehicle just forward of the landing gear and directly over part of it. The pods accommodate 10,000 pounds (4,536 kilograms) of cargo and offer a volume of 1000 cubic feet (28.3 cubic meters). The CM is oriented such that, when the landing gear is attached, the crew control station is directly over one gear between the cargo pods. This position offers maximum visibility for the crew during landing. The crew is lowered to the lunar surface by a personnel elevator assembly. The structural members of the elevator (yoke, main support beam, and telescoping braces) are stowed on the forward surface of the crew module and are easily assembled through use of pivots, trunnions, and pip pins. The elevator is constructed of canvas and aluminum rings with an aluminum-stiffened plate floor and is stored on the forward surface of the crew module. A winch assembly is provided for lowering and raising the elevator. This arrangement of landing gear and top CM with elevator and hemipod cargo modules provides minimum compromise of the original configuration of the vehicle, which is based on earth orbital operations. Lunar surface operations are accomplished by add-on kits to the orbital configuration.

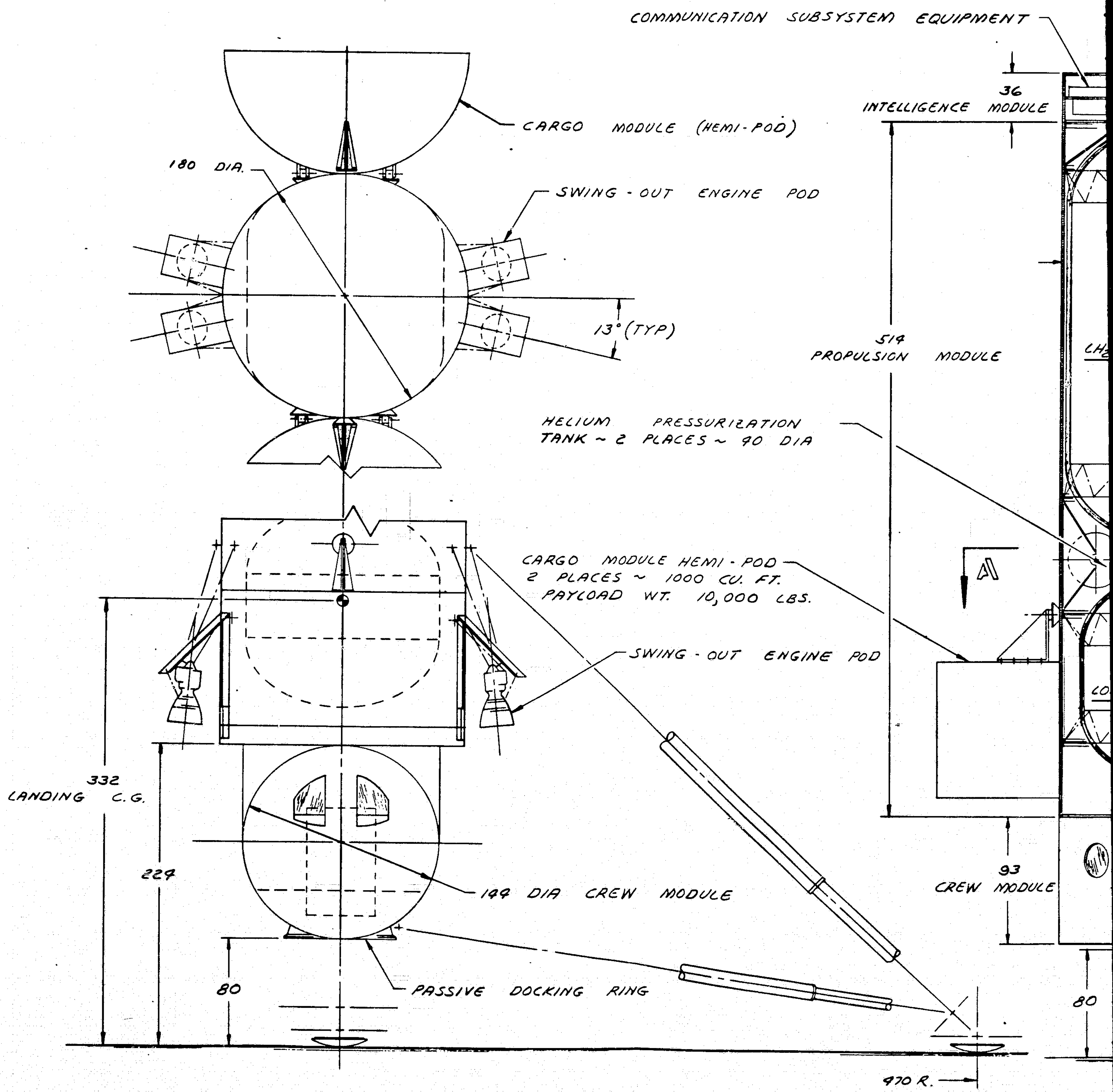
Lunar-Oriented Design

The configuration shown in Figure 2-2 is based on using four 10,000-pound-thrust (44,482-N) engines, which swing out from the basic vehicle body. The configurations shown were developed about the lunar landing mission mode optimization. This tended to favor multiple engines and the size was right to match closely the engine size needed at that time by the EOS orbital maneuvering system (OMS), permitting a possible common engine development. This drawing depicts two configurations for the lunar lander. The first uses a standard vertical crew module (CM). The second uses a horizontally mounted CM of a smaller diameter (12 feet) (3.66 meters) but of the same volume (1300 cubic feet) (36.8 cubic meters) as the 15-foot-diameter (4.57-meter) vertical module.

The swing-out engine concept accommodates several desirable features at one time, which was not possible with other previous configurations studied. First, the CM is located aft (bottom) to provide excellent crew visibility upon landing, easy egress to the lunar surface by the crew, and a minimum of landing gear scar weight on the PM since the landing gear can be partly attached to the CM.

The configuration shown accommodates 80,000 pounds (36,287 kilograms) of usable propellant at a mixture ratio of 6 to 1. A single LH_2 tank and a single LOX tank are used although four LOX tanks were investigated and found to lengthen the configuration. To maintain the shortest possible configuration, the four-LOX-tank version was not selected. The IM was mounted on the forward end with an active neuter cone docking system buried within the central hole in this module. Burying the docking system also contributed to the shortest possible configuration. The pressurization tanks for the main propellant tanks are located between the LH_2 and LOX tanks.

The first configuration shown on the drawing uses a vertical CM identical to that used on previous configurations (internal arrangement). This module is a honeycomb pressure vessel with a passive docking ring at the aft end. Crew egress to the lunar surface when landed or to the space station when in orbit is through this docking ring. Directly above the CM is the PM. It is constructed of a honeycomb cylinder with circumferential stiffeners from which the LH_2 and LOX tanks are supported. Mounted atop the PM is the toroidal IM. The IM is made up of honeycomb bulkheads and a central shelf that is closed out by an inner cylinder and outer panels of honeycomb. The IM houses all of the subsystem equipment and the ACS system, including the four sets of ACS engine quads. A neuter cone docking system is attached to the IM and is partly submerged in the central portion of the torus. A conical skin-stringer structure between the PM outer shell and the aft surface of the inner cylinder of the IM helps support the docking system.



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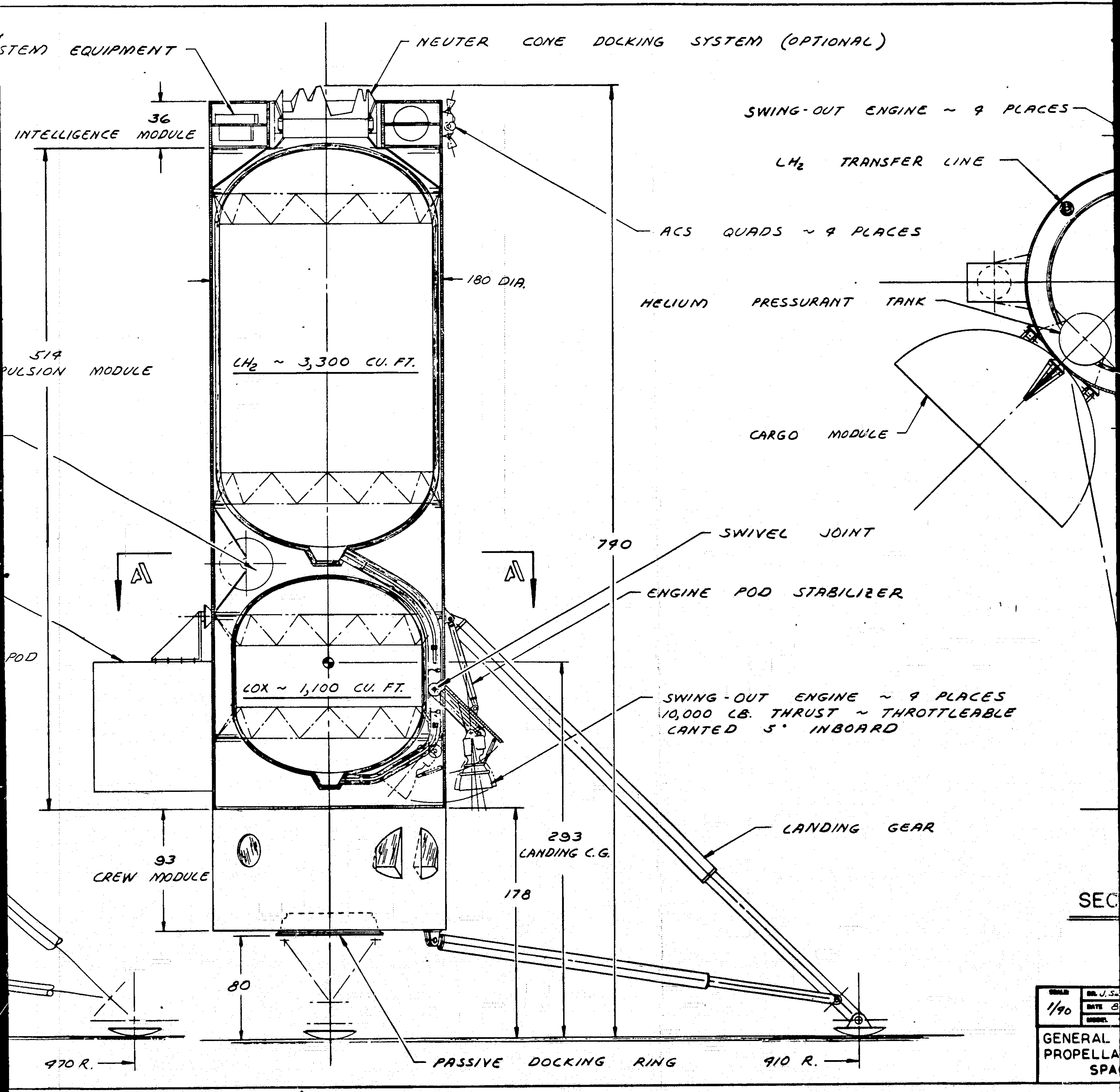

Figure 2-2. 80K Lb Propellant - Swingout E
(Sheet 1 of 2).

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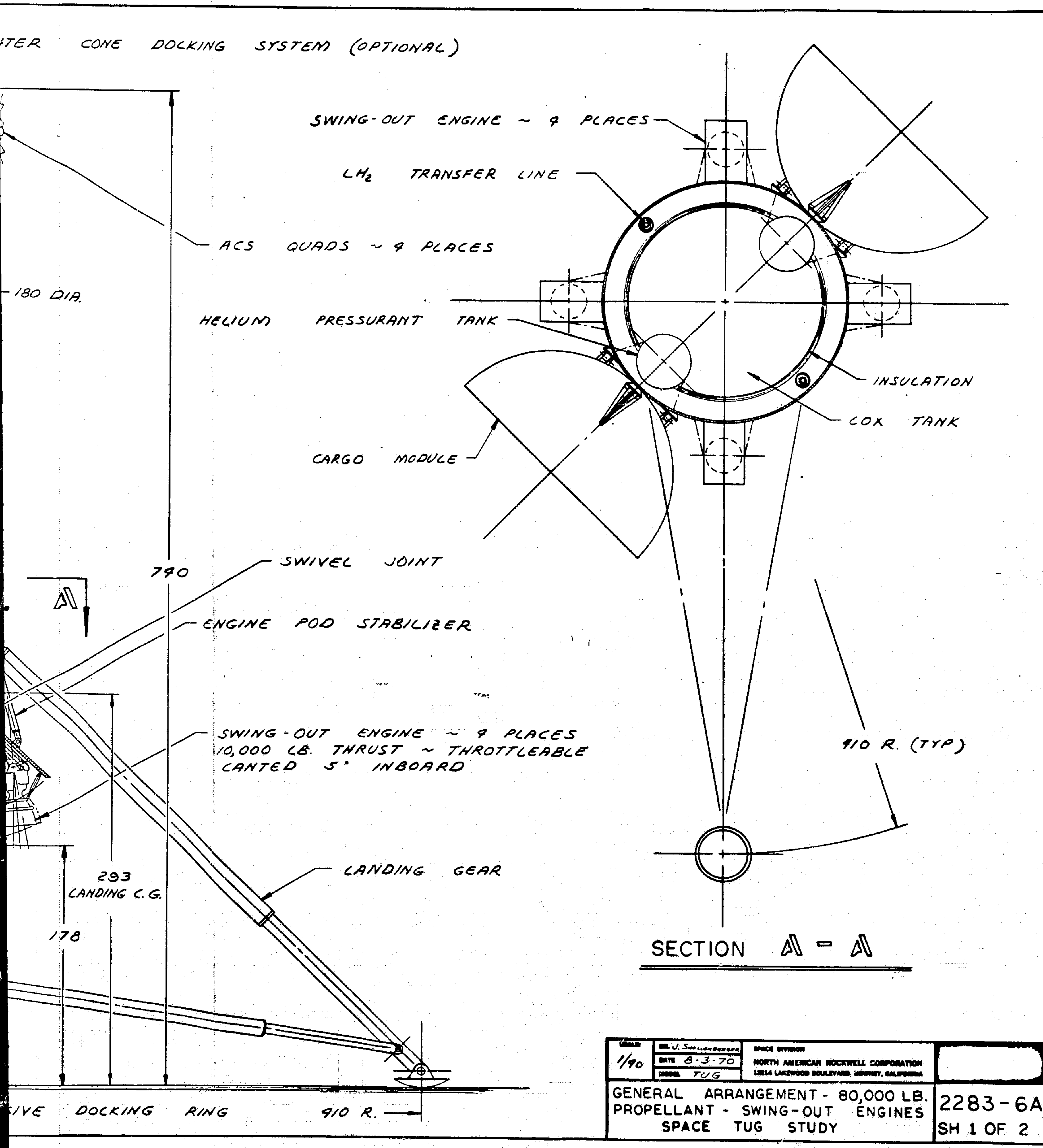
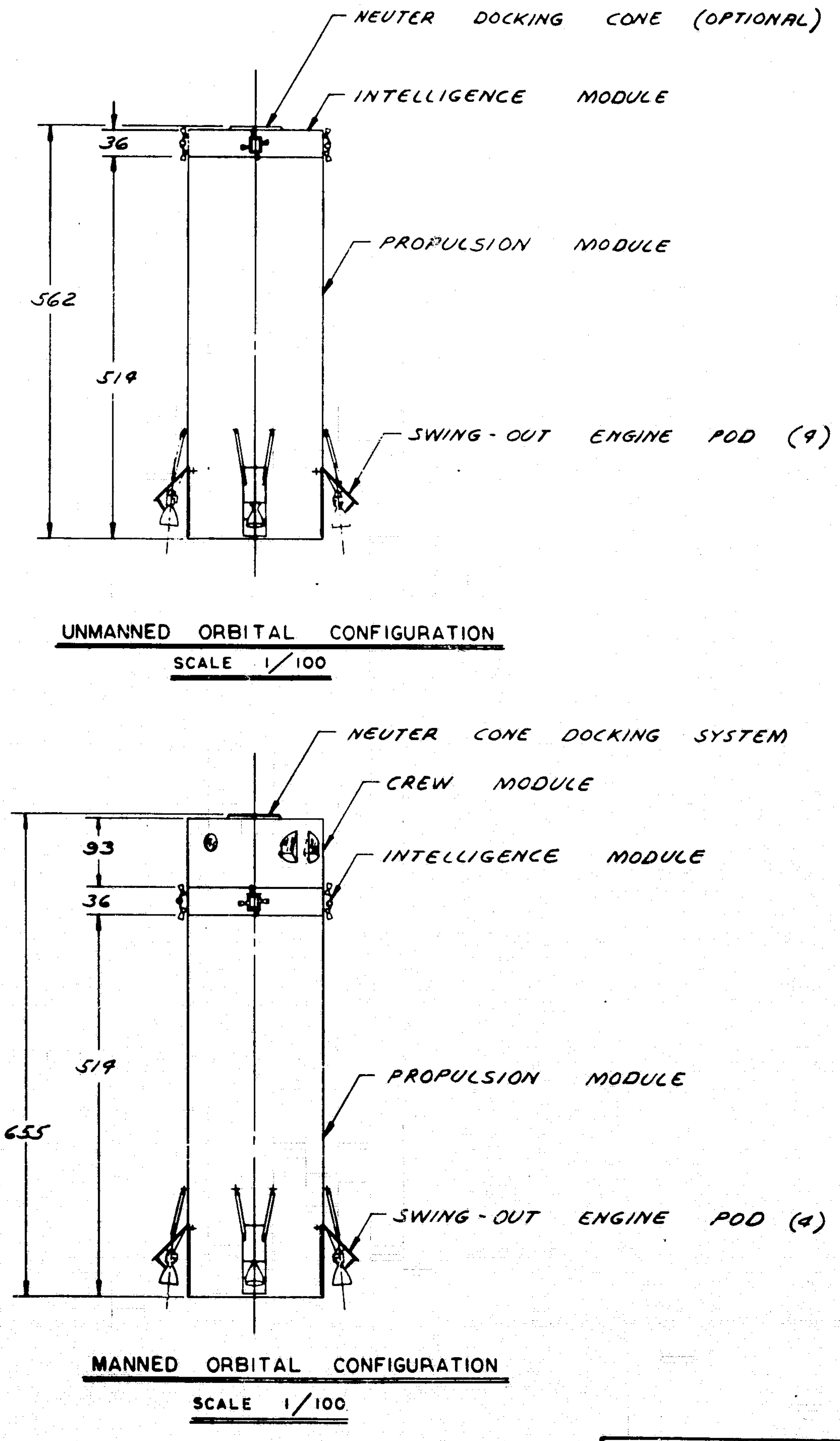


Figure 2-2. 80K Lb Propellant - Swingout Engines General Arrangement (Sheet 1 of 2)



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SHEET 2 OF 2

Figure 2-2. 80K Lb Propellant - Swingout Engines General Arrangement (Sheet 2 of 2)

The LH_2 and LOX tanks in the PM have ellipsoid bulkheads and cylindrical midsections. The LH_2 tank accommodates 3,300 cubic feet (93.4 cubic meters) of LH_2, and the LOX tank accommodates 1,100 cubic feet (31.1 cubic meters). Overall the PM is 514 inches (13.1 meters) tall. The crew module is attached to the aft end of the PM at a ground assembled joint.

The arrangement of the swing-out engines and cargo pods for this configuration is shown in Section A-A. The engines are located at 90 degrees to each other, with the cargo pods located between two engines opposing each other. The engines are supported from a segment of the outer shell of the PM. This segment or door is capable of pivoting 45 degrees outward. The engine is attached rigidly to this door with its gimbal actuators bridging from the engine to the door. The pivot point of the door also serves as the pivot point for an LH_2 and LOX swivel joint. This joint allows the lines from the tank to the swivel and the engine to the swivel to be solid lines, with all rotation of the supply lines being done by the swivel joint. In the stowed position, or when the engine pod doors are closed, the engines are nested near the bottom of the LOX tank. When the engines are deployed, the lower portion of the engine nozzle passes through swinging doors, which close off or complete the outer shell of the PM in the area of the engine nozzle. The doors prevent the engine plume from entering the inner part of the PM. When deployed, the engines are canted 5 degrees inward toward the top of the propulsion module. Engine pod stabilizers are provided (two per door). They pick up the door at the outer surface, where the engine mounts are located, and transmit the loads to the upper LOX tank support ring. The ring is approximately 54 inches (1.4 meters) forward of the engine pod hinge point.

Another approach to the CM is shown in the second configuration. It uses a horizontally attached module. This module is also ground-affixed to the aft end of the PM. This module is 12 feet (3.66 meters) in diameter and, when viewed from the top, has ellipsoid bulkheads. The floor inside the CM is approximately 36 inches (0.91 meters) from the lowest point.

A passive docking ring is attached to the central portion of the lowest part of the module. This module contains the same volume (1300 cubic feet) (36.8 cubic meters) as the vertical crew module shown in the first configuration. The change to a horizontal crew module results in the vehicle being 46 inches (1.2 meters) longer (higher) than the vertical crew module version and exhibits a center of gravity that is 39 inches (0.99 meters) higher. The greatest impact of this length increase can be seen in the landing gear spread radius, which increases by 60 inches (1.5 meters).

The concept of engine deployment and support is identical to that discussed. However, the location of the engine pods is different, as shown in the top view of this configuration. The engine pods are grouped into two

pairs of engines 180 degrees apart. The pods are located 26 degrees from each other on their center lines. The cargo pods are located between the engine pods and directly above the bulkhead of the CM.

The landing gears are mounted similarly in both configurations. The main struts attach to the upper LOX tank support ring, and the stabilizers attach to the aft portion of the crew module. The location of the stabilizers on the horizontal CM appears to be marginal for a useful diameter. It should probably be moved forward to some point where greater spread of the stabilizer can be obtained. The landing gear is considerably larger than that used on previous configurations and is considerably heavier, as discussed.

An unmanned and manned orbital configuration is shown at the end of the drawing. The unmanned configuration is identical to the lunar lander configuration with the cargo pods, landing gear, and CM removed. The manned orbital configuration attaches the vertical CM forward of the CM.

The configuration shown in Figure 2-3 is similar to that on Figure 2-2 in that several identical guidelines were used for each configuration. The concept accommodates 80,000 pounds (36,287 kilograms) of usable propellant, has single LH_2 and LOX tanks, and uses a forward-located IM. The configuration shown also accommodates two 20,000-pound-thrust (88,946-N) engines. The configuration is based on a geosynchronous mission mode, which accounts for the lack of a crew module. No docking system is shown at the forward end, but a neuter cone system could be added. Room exists in the central cylindrical portion of the IM to partially bury this system.

The basic PM structure is a honeycomb cylindrical shell with fore and aft honeycomb bulkheads. The LH_2 and LOX tanks are each supported from fore and aft circumferential rings attached to the shell. From the aft ring, which supports the LOX tank, a truss support extends aft to support the main engine beam. The engines are attached to this beam, and the engine loads are transmitted through the truss to the outer shell structure at the ring. The IM is toroidal and constructed of three wafer-like honeycomb discs, which comprise the forward and aft bulkheads and the centrally located equipment shelf. The torus is completed by an inner cylindrical honeycomb section and outer honeycomb covers.

The IM contains all of the subsystem equipment. It is mounted into the basic module as segments of the toroidal cylinder. Four of these segments are identical ACS sections complete with four-engine quads mounted on the periphery and tankage, associated plumbing, and central hardware mounted on the inboard shelf.

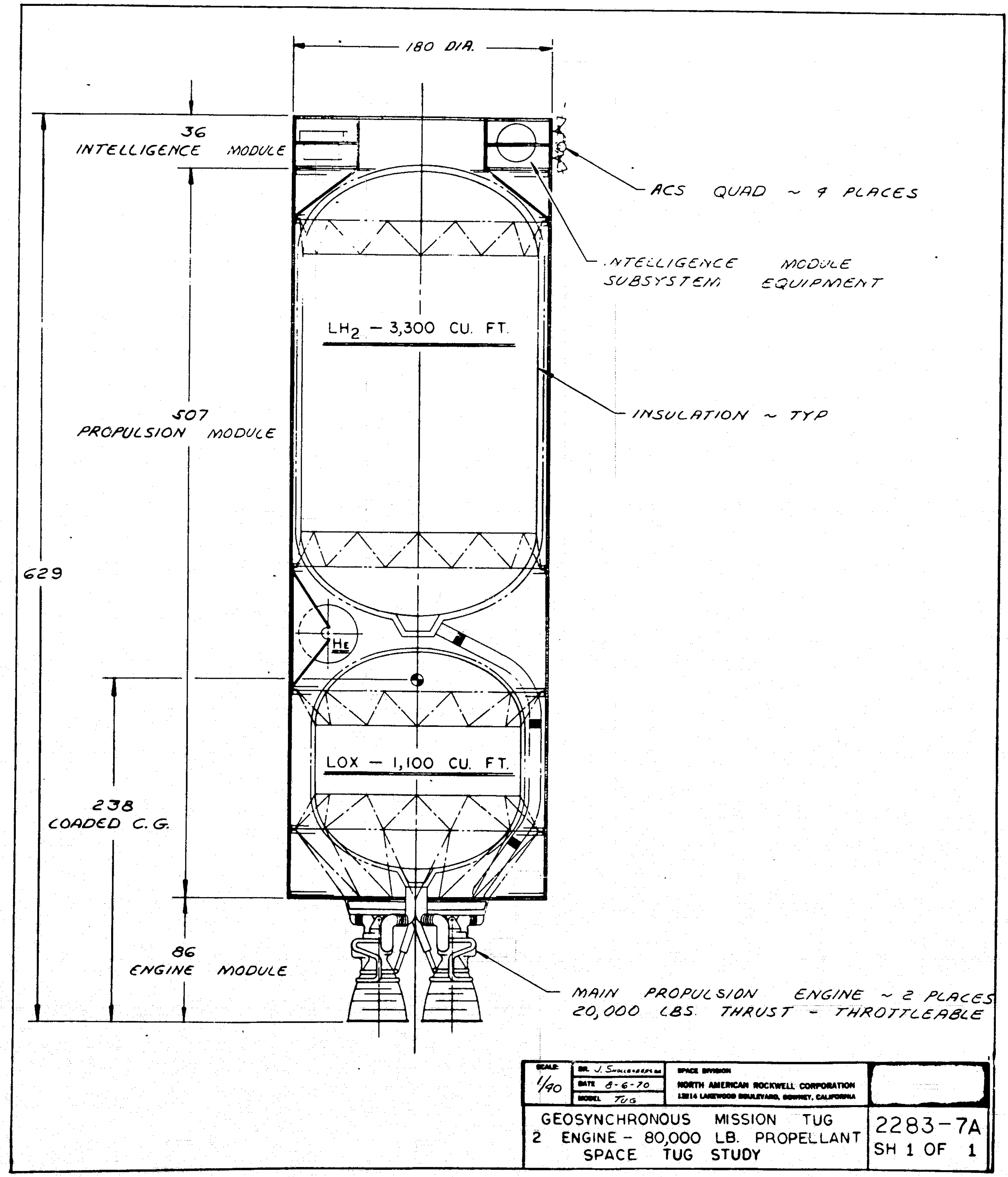


Figure 2-3. 2-Engine - 80K Lb Propellant Geosynchronous Mission Tug

The diameter of inner cylinder of the IM is sufficient to allow the mounting of a neuter cone docking system within the cylinder. This reduces to a minimum the lengthening of the vehicle from such an addition.

The LOX tank is smaller in diameter than the LH_2 tank to allow clearance between the tank and outer cylinder shell for the LH_2 transfer lines. Both the LH_2 and LOX tanks are cylindrical, with ellipsoid bulkheads of a 1.4 to 1 ratio.

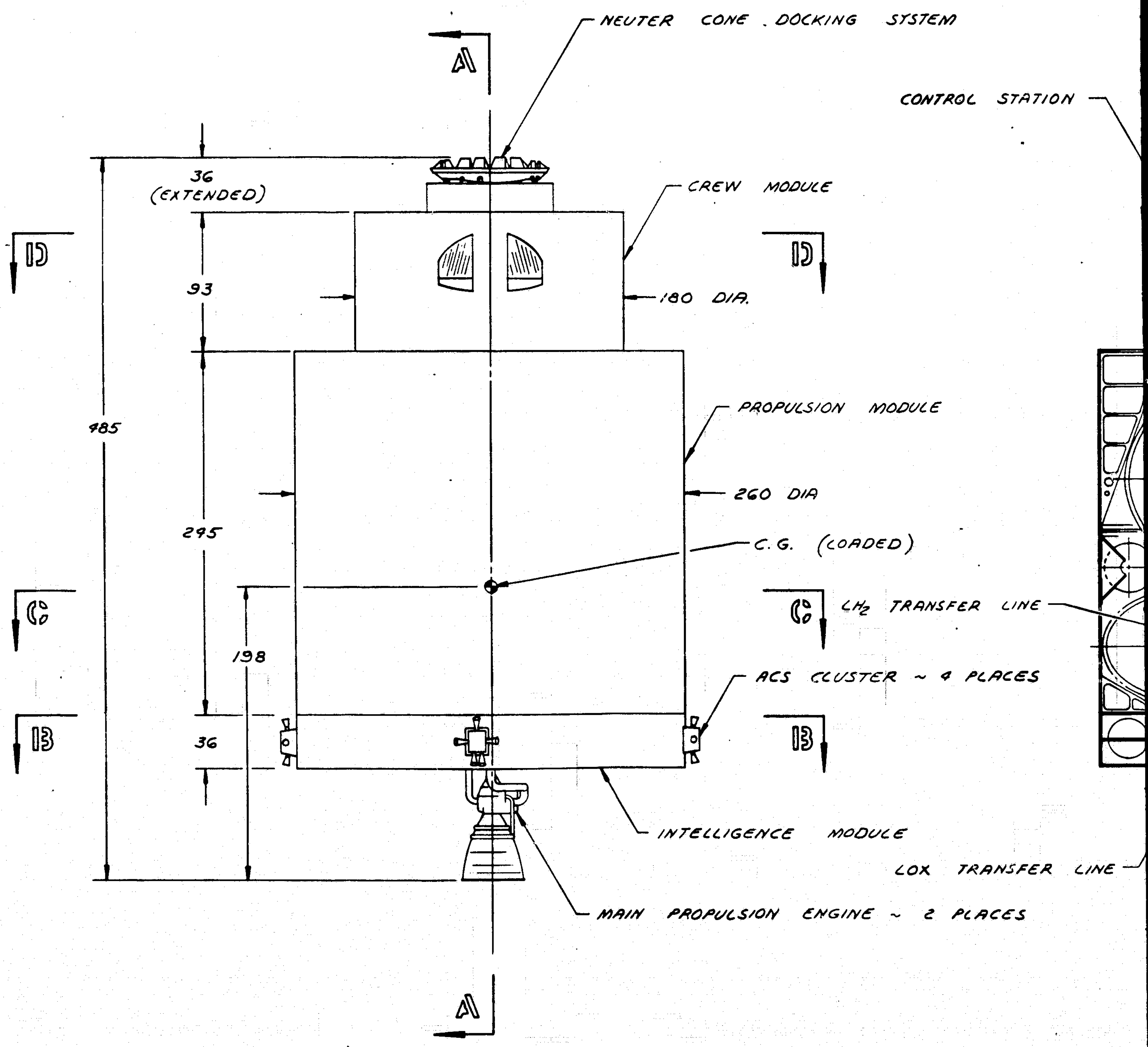
2.2.2 22-FOOT-DIAMETER (6.7-METER) CONFIGURATION STUDIES

Optimum CM Location, Earth-Orbital and Lunar

For a meaningful dimensional and weight comparison of the 15- and 22-foot diameter (4.57 and 6.7 meters) tug vehicles, two 22-foot (6.7 meters) vehicles were configured with the same groundrules as used on their 15-foot (4.57-meters) counterparts. Each configuration used one LH_2 tank and four LOX tanks. The first configuration is shown in Figure 2-4 and is configured for comparison to a 15-foot (4.57-meters) configuration as shown in Figure 2-5. The concept was to design the basic vehicle for both orbital and lunar surface operations. Consequently, it was determined that a forward-located CM for orbital operations and an aft-located one for lunar surface operations would accommodate both modes. Commonality in the CM to permit its use for both modes is desirable and was adopted as a groundrule.

The PM is made up of one ellipsoid hydrogen tank, four ellipsoid oxygen tanks, and two modified RL-10 engines. The PM is 22 feet (6.7 meters) in diameter and is structurally a honeycomb cylinder with radial beams from which the propellant tanks are supported. The two engines are mounted on the aft section of the PM and extend 112 inches (2.8 meters) aft of the aft bulkhead. A 22-foot-diameter (6.7-meters) torus shaped IM is mounted aft of the aft bulkhead. This module contains the four sets of RCS tanks, engines, and associated equipment, as well as all of the subsystem equipment. The module has much room for growth in all the subsystem areas since it essentially houses the same equipment as used in the 15-foot (4.57-meters) version. The IM is constructed of honeycomb removable outer panels, which enclose the donut-shaped forward bulkhead, aft bulkhead, and centrally located shelf. All of the subsystem equipment and tanks are mounted to the central shelf, which is assembled into the module in segments.

For orbital operations, the CM is mounted to the forward end of the PM. A neuter cone docking system is provided for on the forward end of the CM. This enables the tug to dock with all payloads having either an active or passive docking system. The CM is essentially a pressure vessel constructed



ORBITAL OPERATIONS CONFIGURATION

TOTAL PROPELLANT LOAD 60,000 LBS.

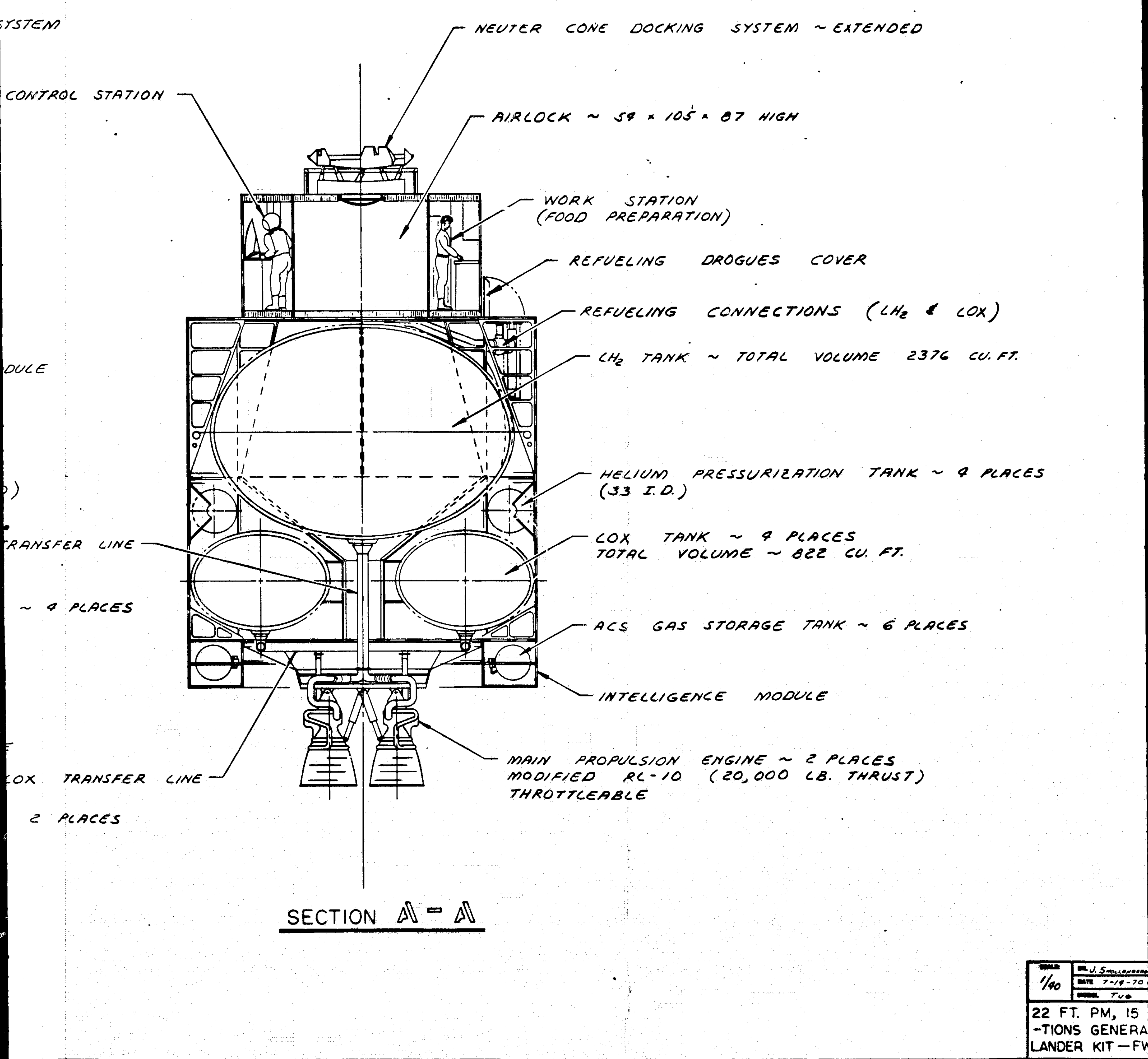


Figure 2-4. 22-Foot/Propulsion Module, 15-F
Operations General Arrangement - A
(Sheet 1 of 3)

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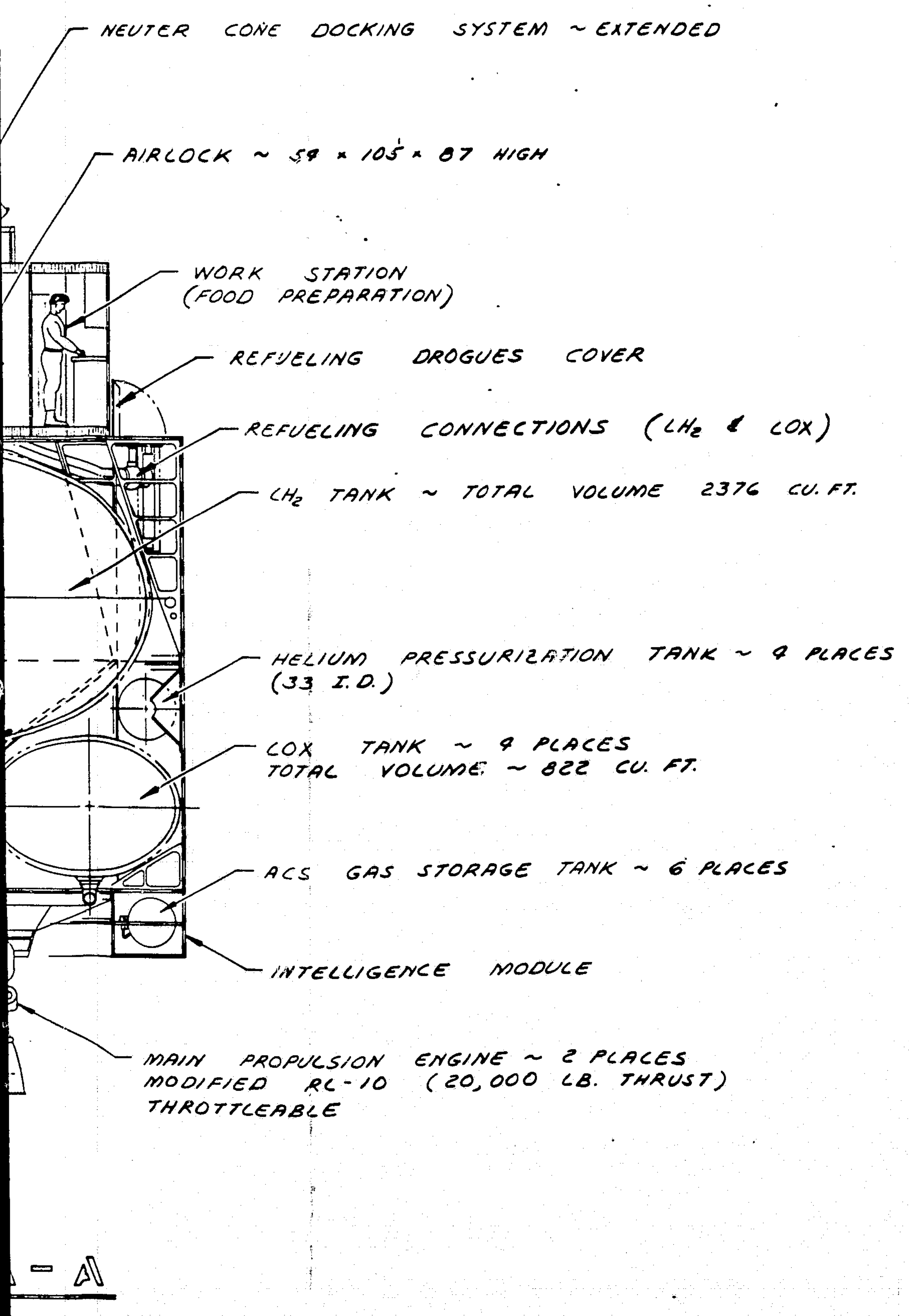
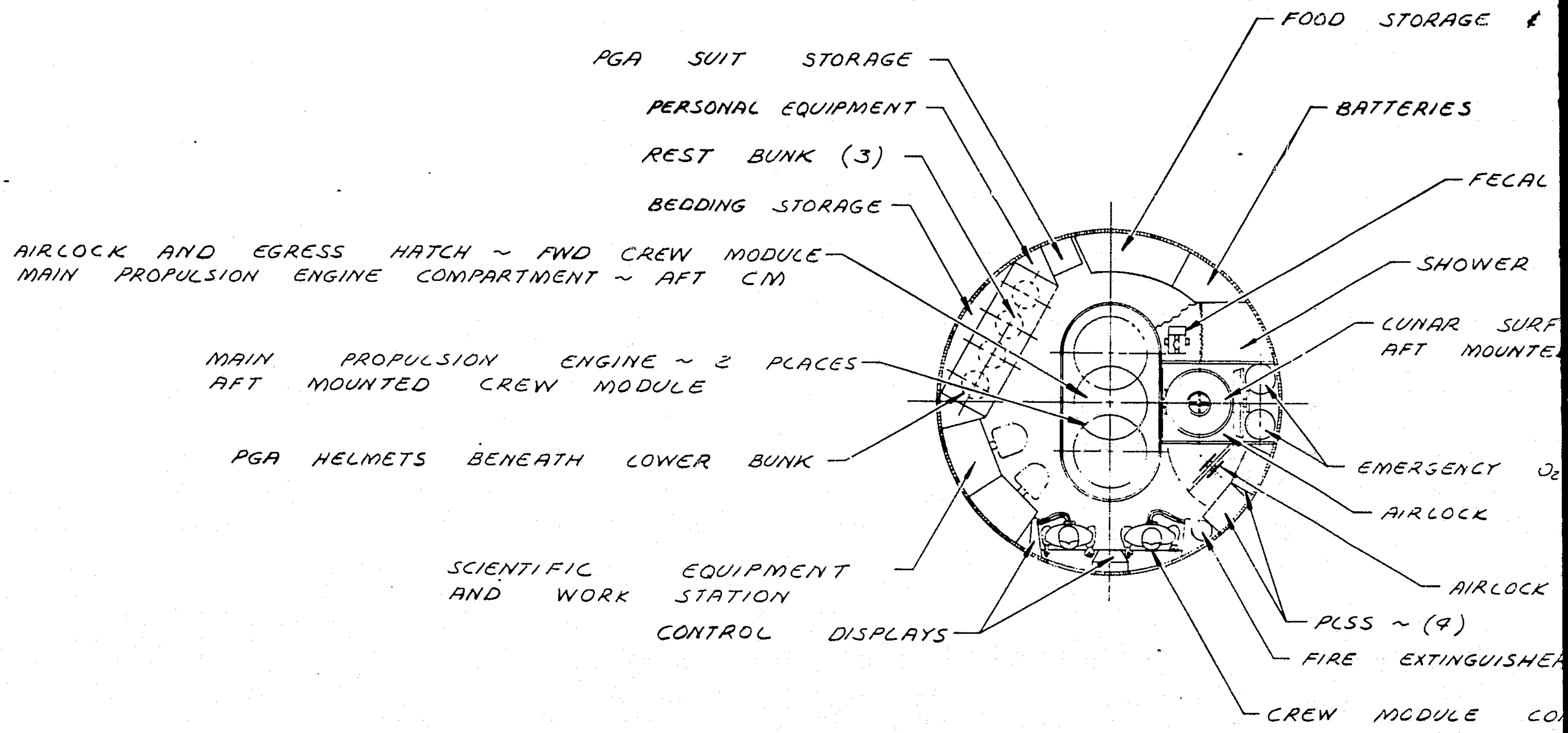


1/40 — J. Schollenberger — DATE 7-14-70 — MODEL TUG — SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

22 FT. PM, 15 FT. CM—ORBITAL OPERA-TIONS GENERAL ARRANGEMENT—ADD-ON LANDER KIT—FWD CM—SPACE TUG STUDY

2283-3C
SH 1 OF 3

Figure 2-4. 22-Foot/Propulsion Module, 15-Foot Core Module Orbital Operations General Arrangement - Add-On Lander Kit (Sheet 1 of 3)



SECTION D – D

CREW MODULE ARRANGEMENT

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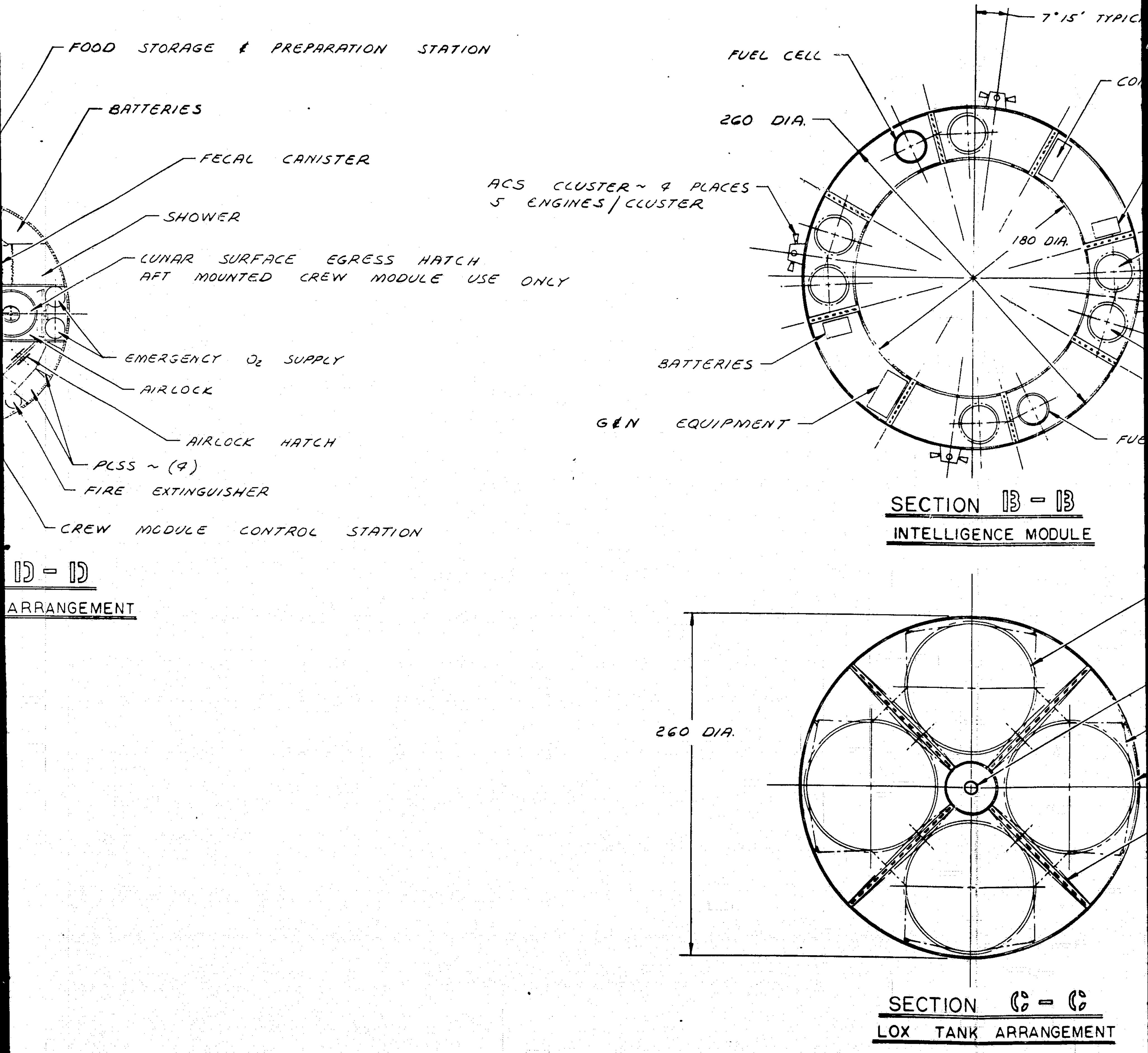

Figure 2-4. 22-Fo
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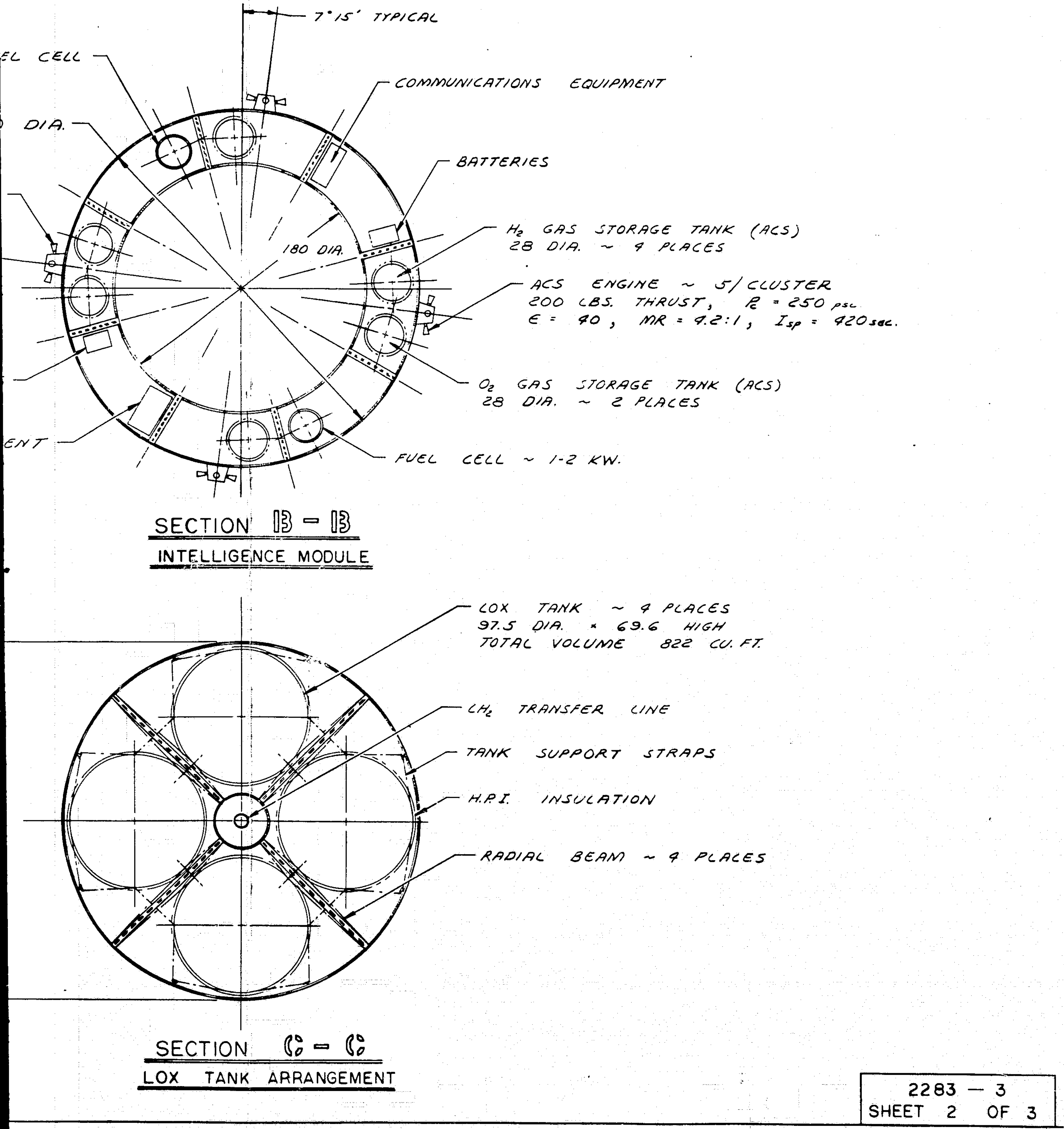

Figure 2-4. 22-Foot/Propulsion Module, 15-Foot Core Module Orbital Operations General Arrangement - Add-On Lander Kit (Sheet 2 of 3)

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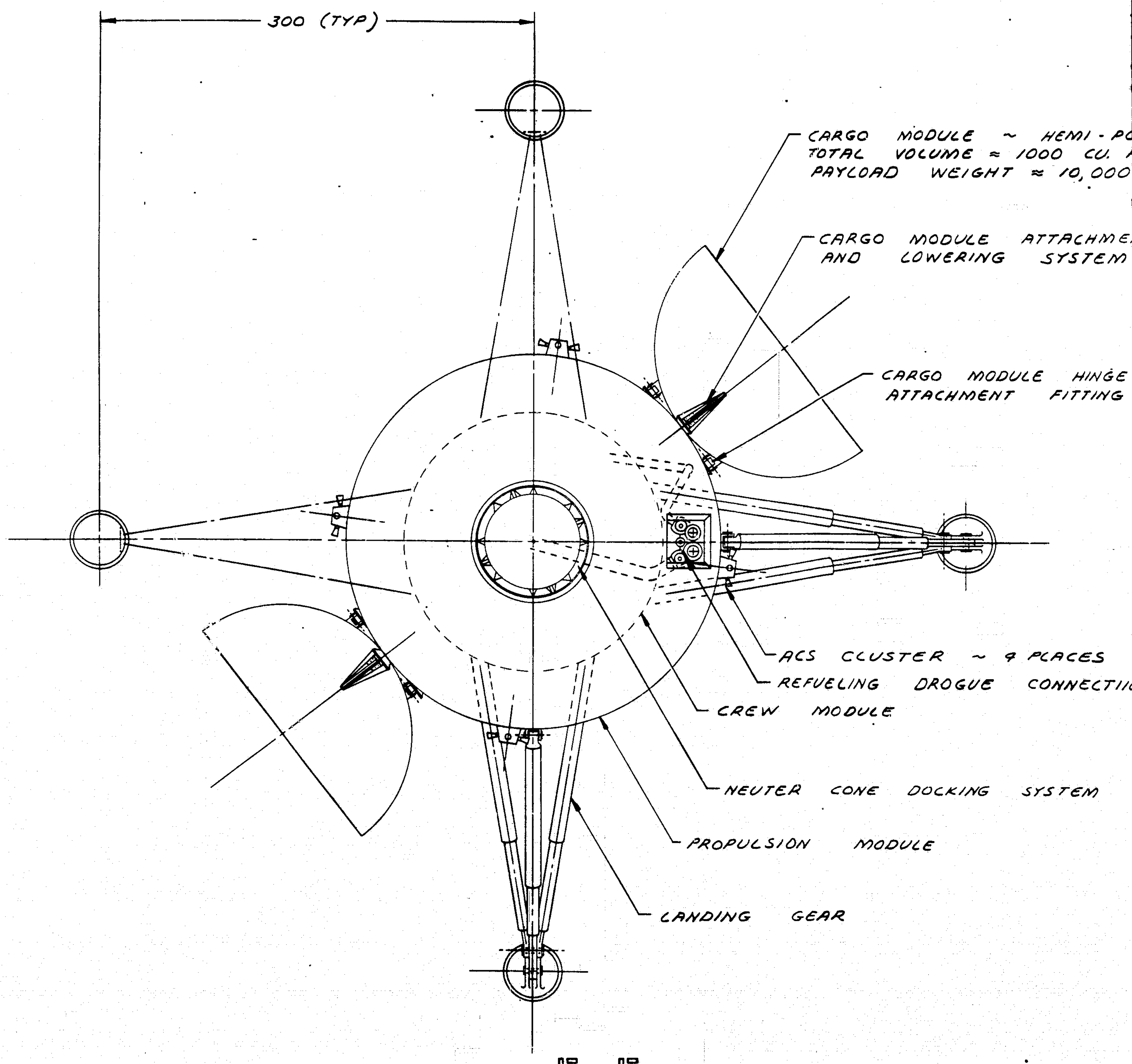

VIEW E - E

LUNAR LANDER TUG

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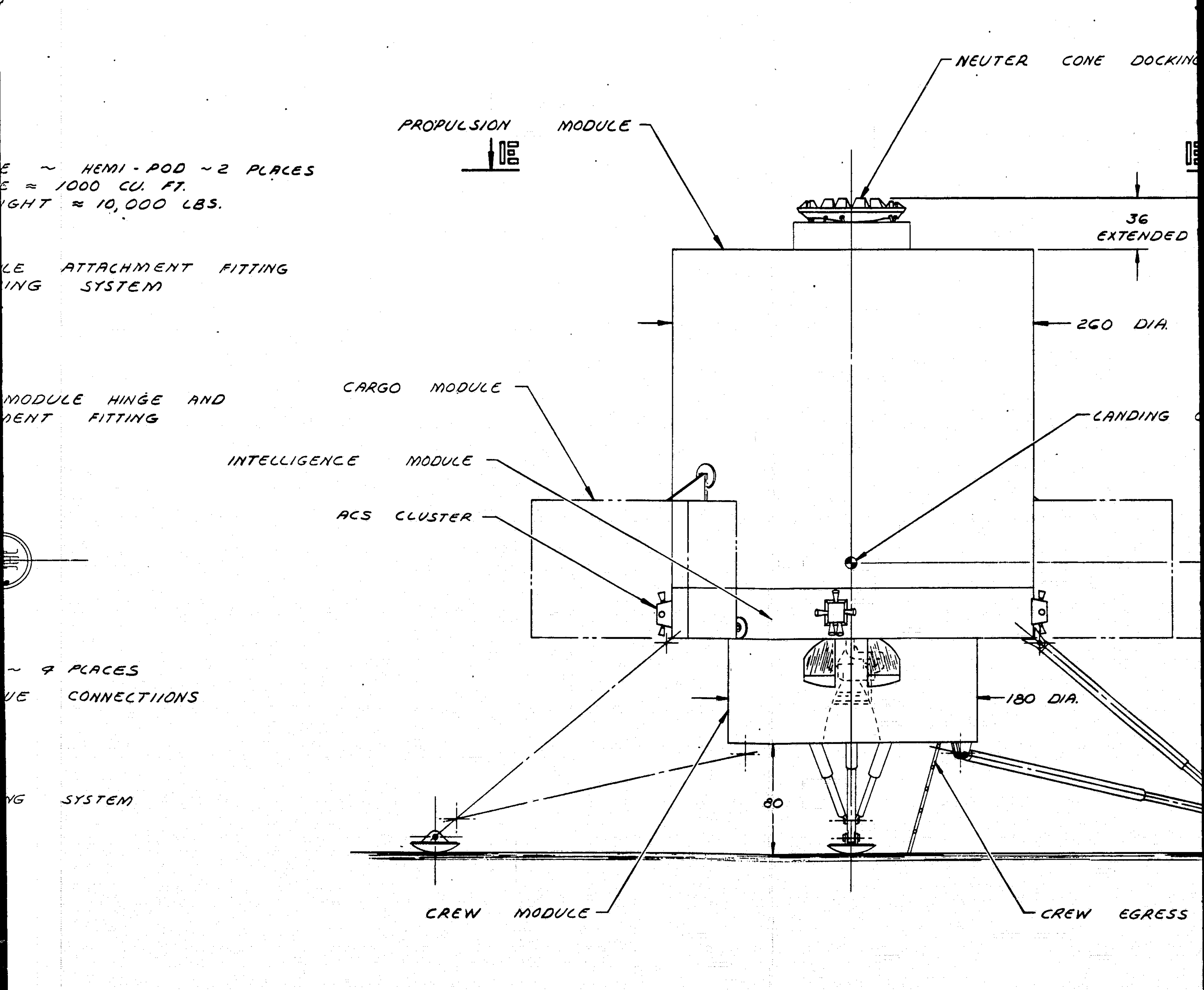

LUNAR SURFACE OPERATIONS CONFIGURATION

Figure 2-4. 22-Foot/Propulsion M
Operations General Arran
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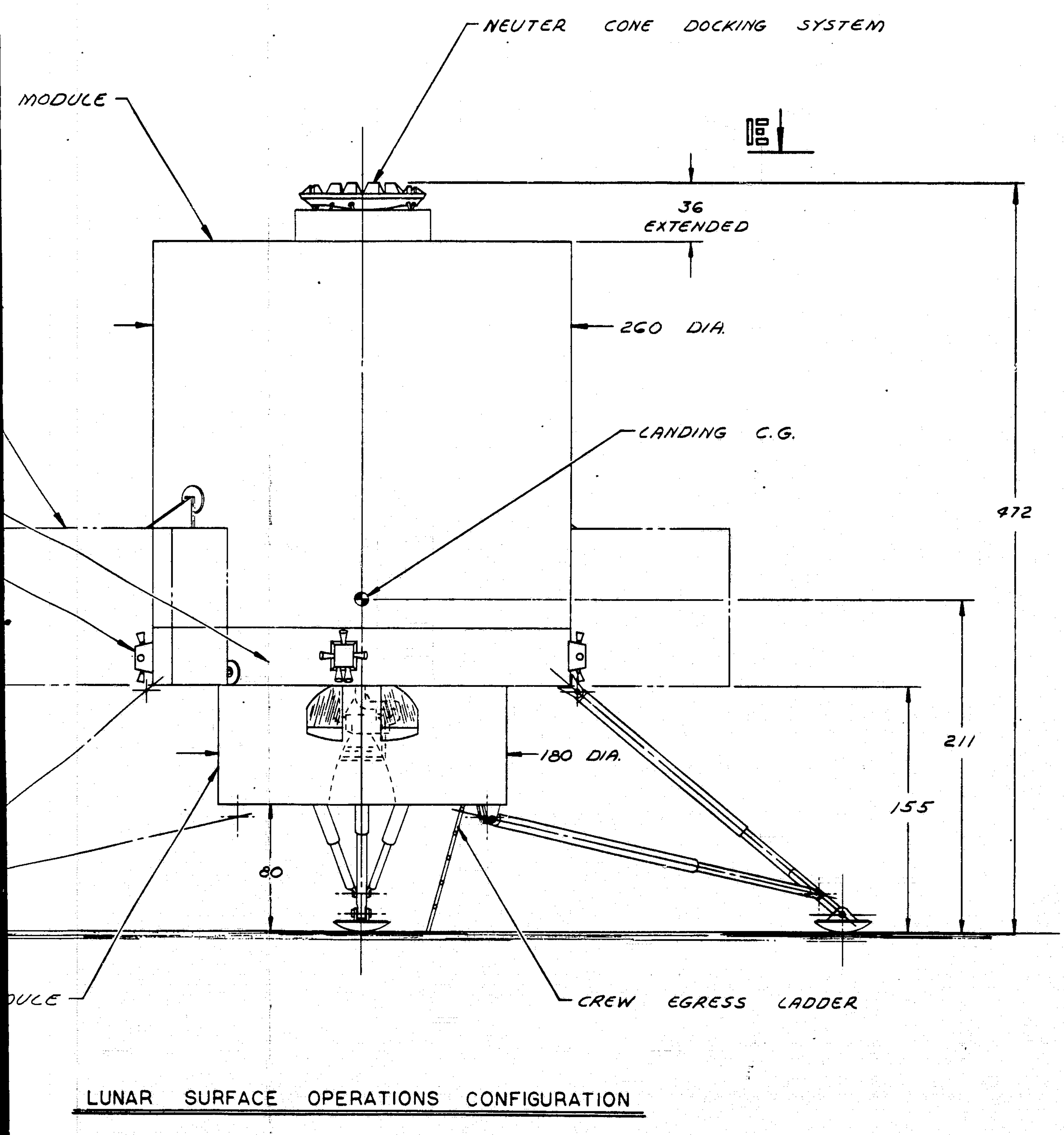
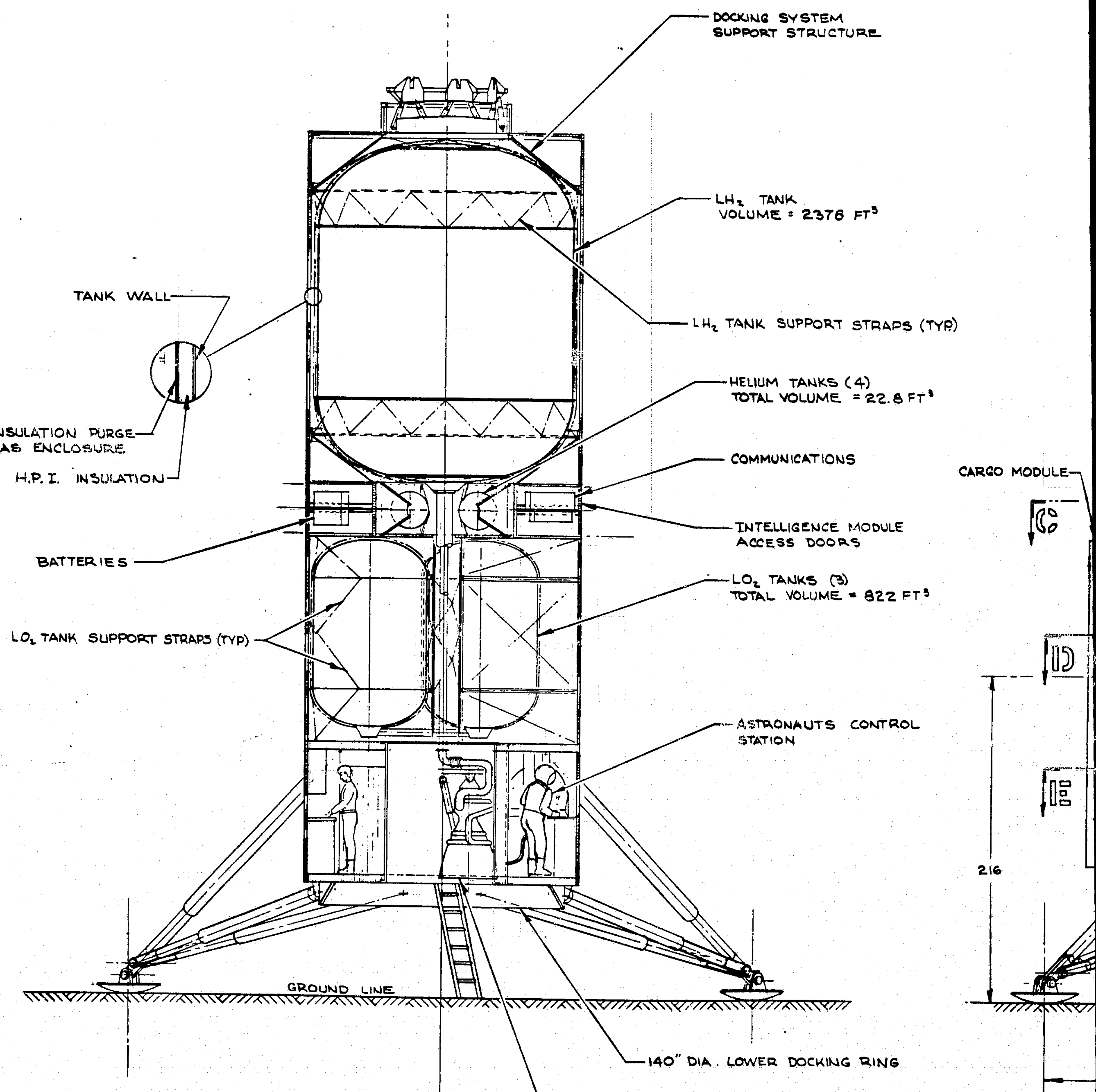


Figure 2-4. 22-Foot/Propulsion Module, 15-Foot Core Module Orbital Operations General Arrangement - Add-On Lander Kit (Sheet 3 of 3)



SECTION A-A

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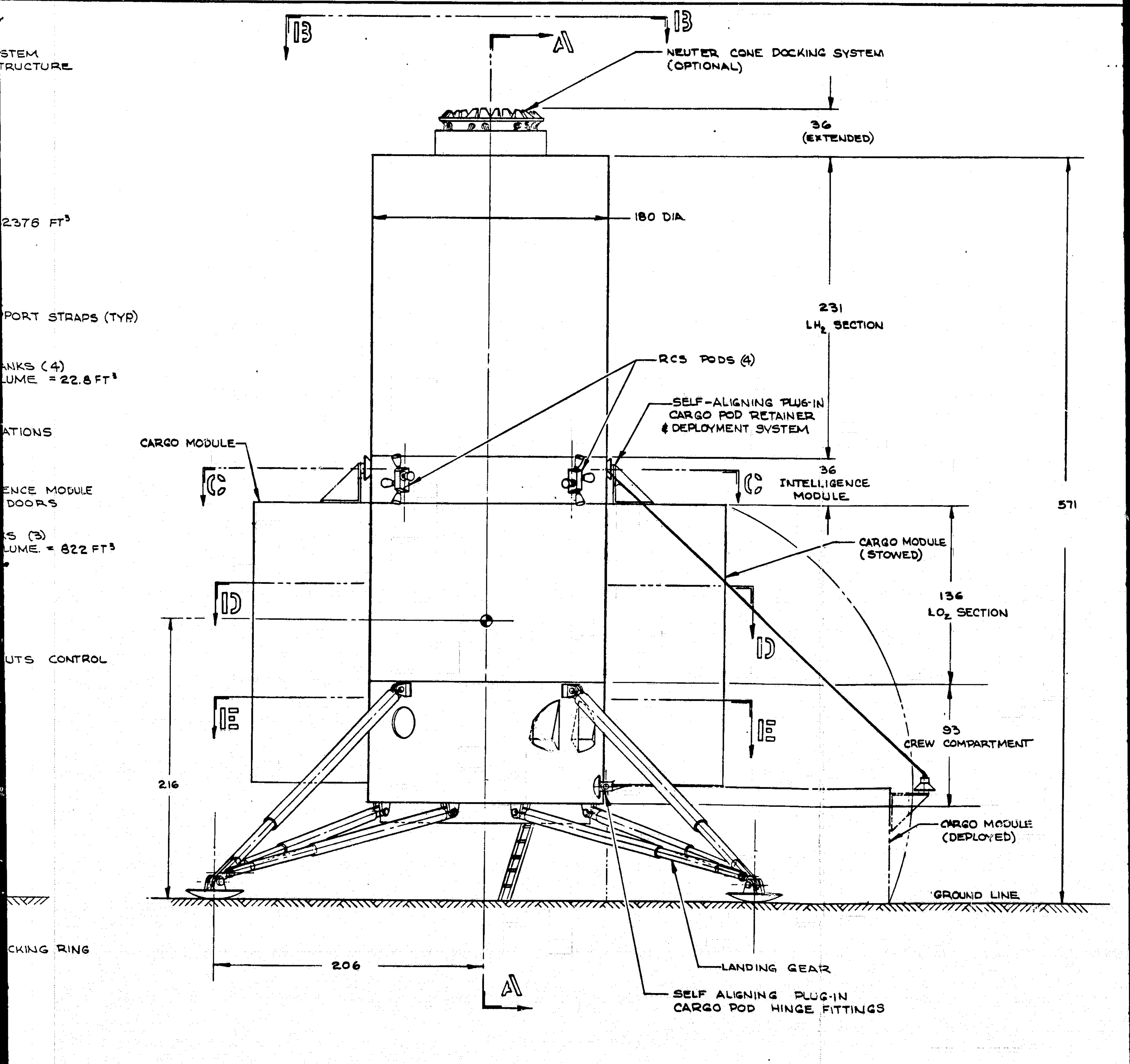

Figure 2-5. 15-Foot Diamet
Lander

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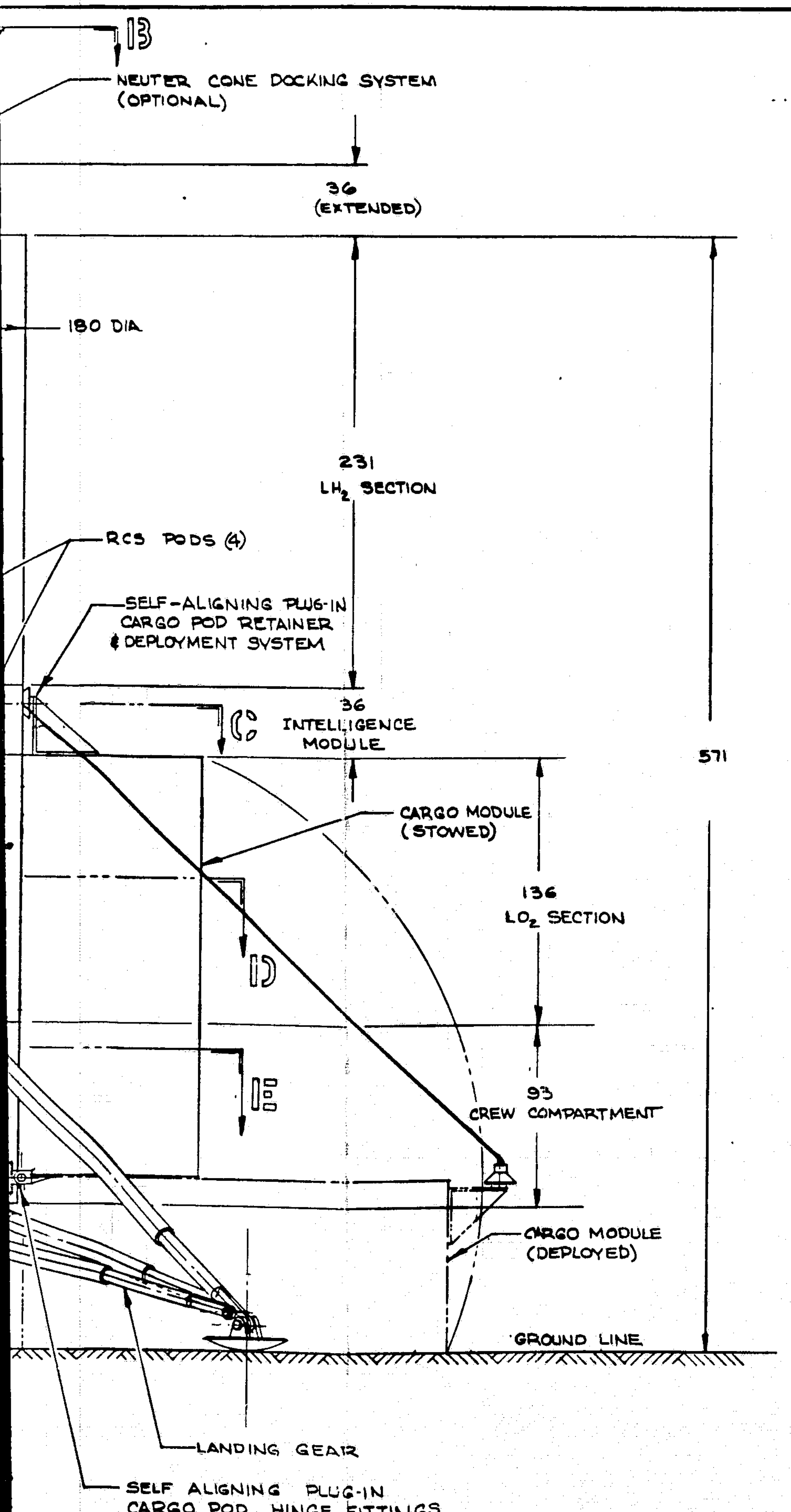

SCALE 1/40	DR. R.G. Cook / DATE 3-31-70 / MODEL	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
LUNAR LANDER GENERAL ARRANGEMENT - 15 FT DIA., 4 MAN CREW, AFT CREW MODULE — SPACE TUG STUDY -			5383-15A

Figure 2-5. 15-Foot Diameter, 4-Man Crew Aft Crew Module Lunar Lander General Arrangement (Sheet 1 of 2)

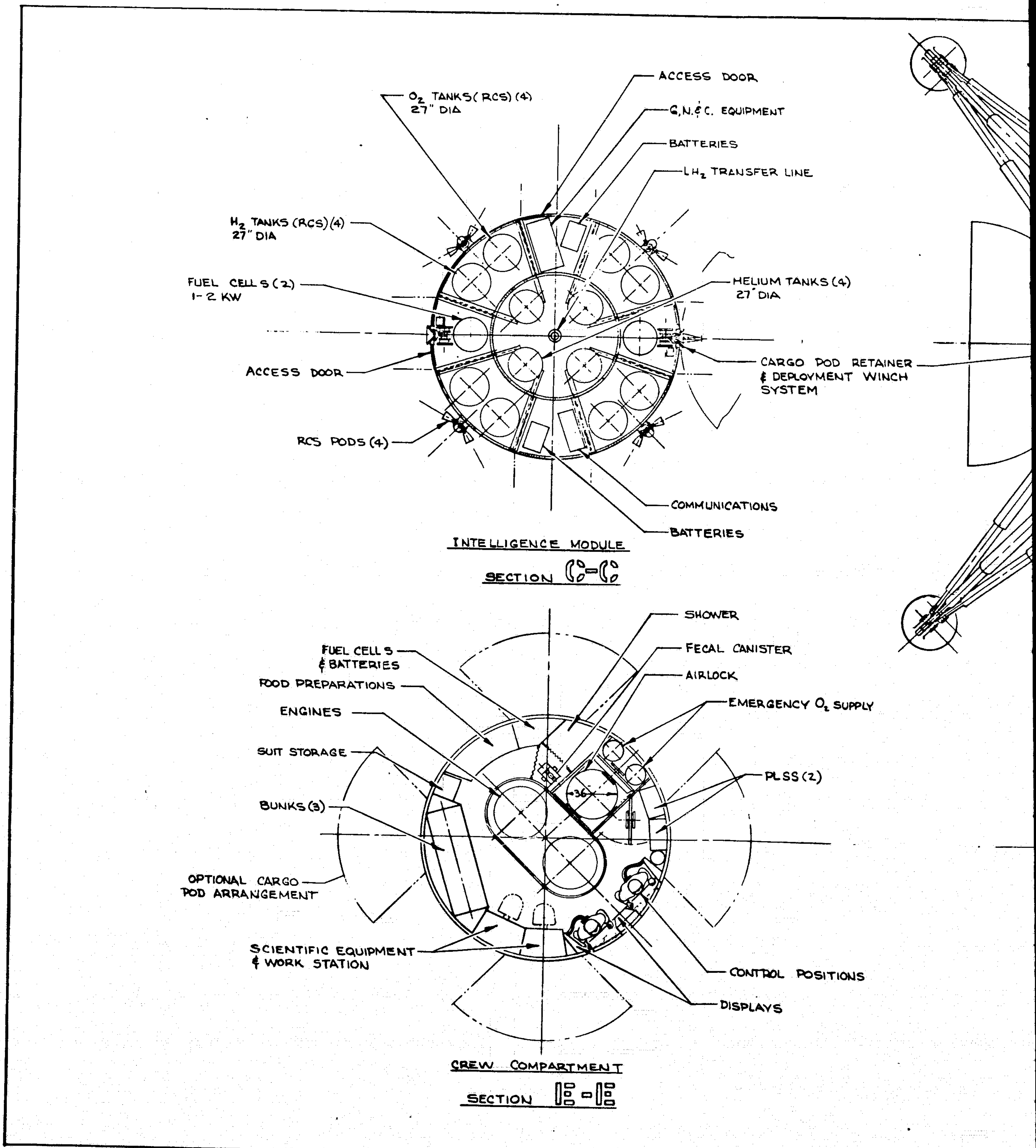

Figure 2-

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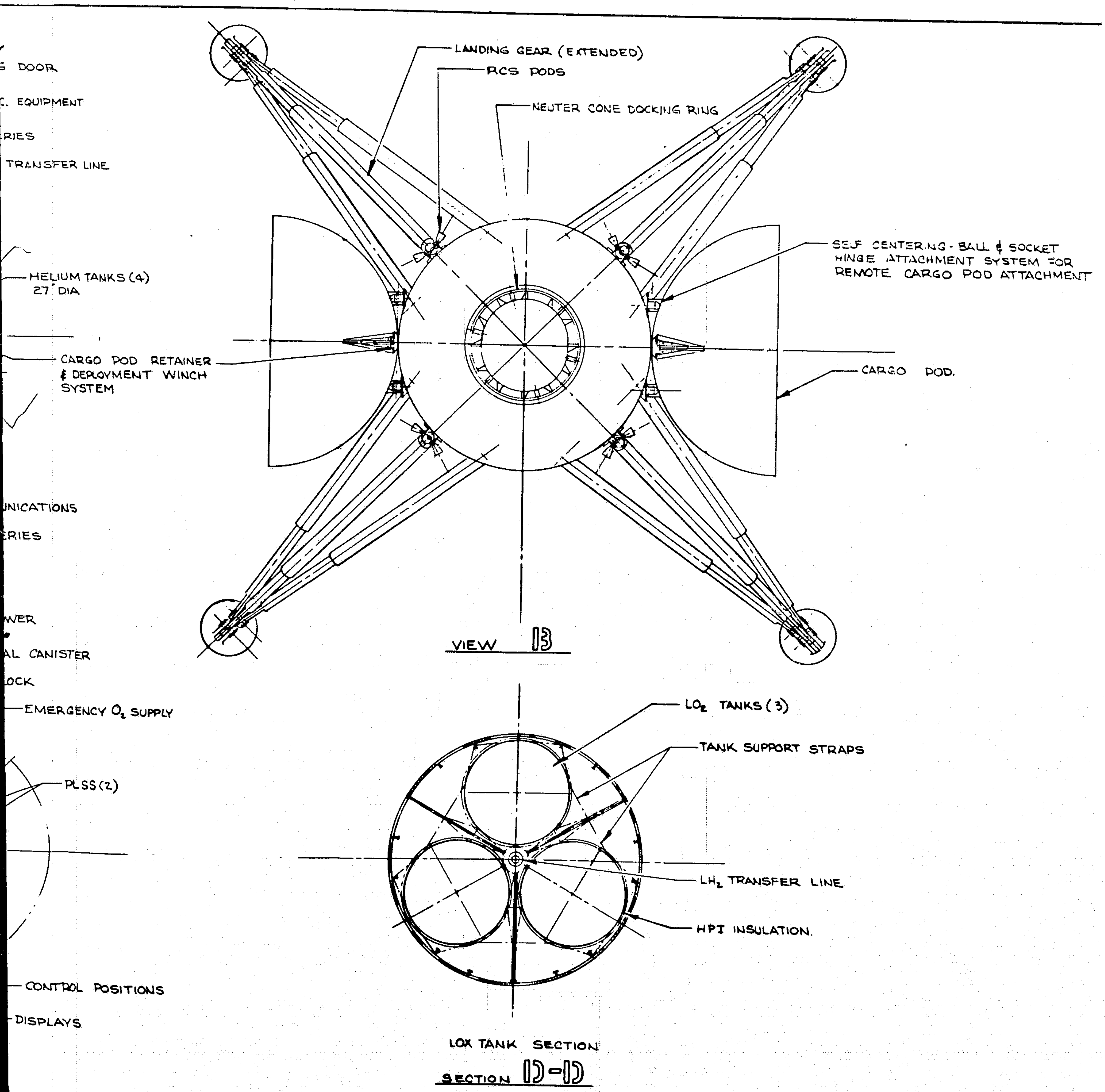


Figure 2-5. 15-Foot Diameter, 4-Man Crew Aft Crew Module Lunar Lander General Arrangement (Sheet 2 of 2)

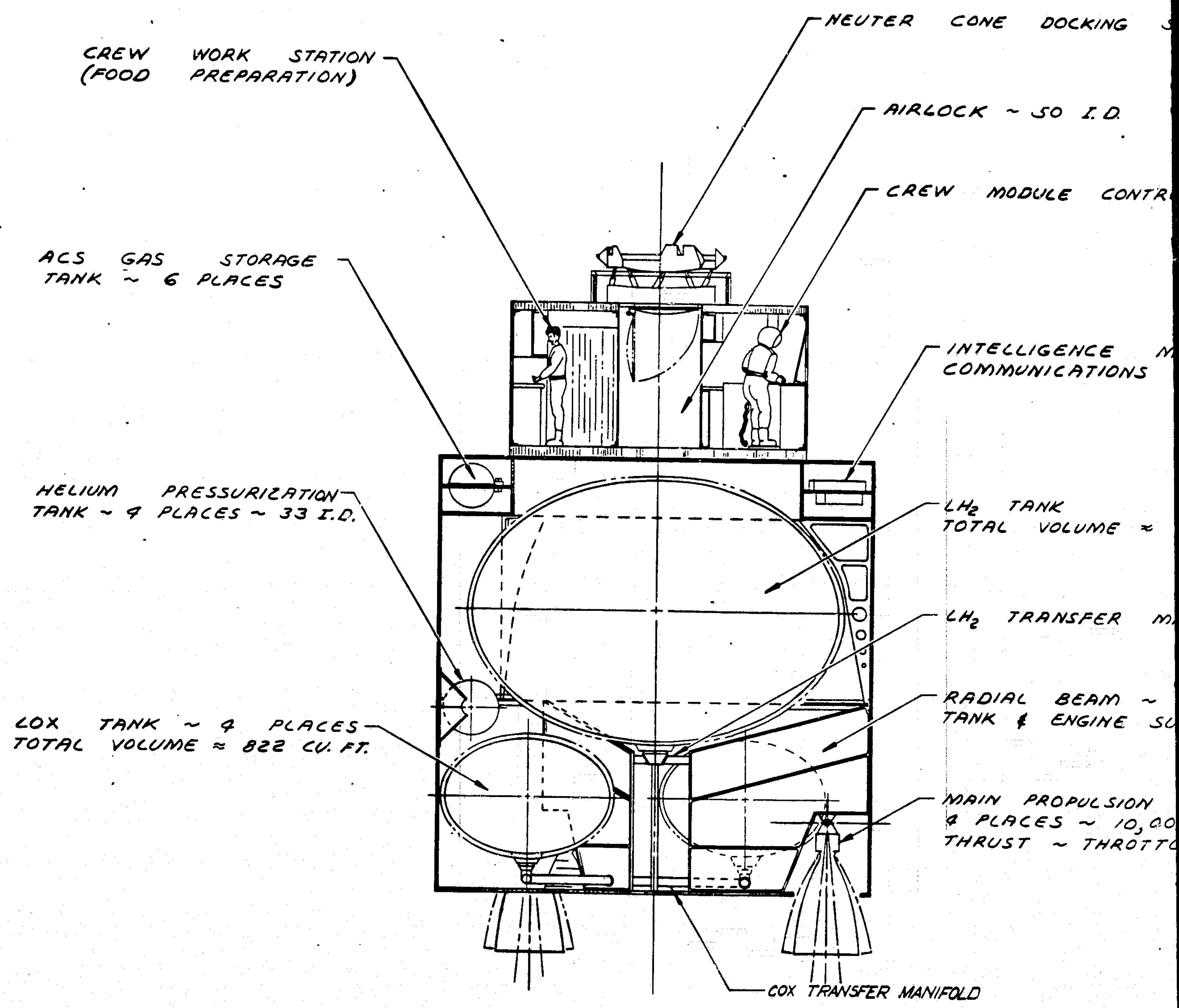
of honeycomb. The outer shell and inner air lock are constructed of honeycomb. The central area air lock is oblong or slotted-shaped and is used as an air lock when the CM is mounted forward. A second air lock is connected to this and divides the CM. This second air lock is used as the primary air lock when the CM is mounted on the aft end of the vehicle. The central air lock contains an egress hatch at the forward end through the docking system. The secondary air lock contains a hatch in the floor for crew access to the lunar surface when the vehicle is used as a lander. The CM contains both control stations and work stations, as well as provisions for use as a shelter for lunar or orbital operations.

For lunar landing, a landing gear kit is added, as well as cargo modules. In this mode the CM is moved to the aft end of the vehicle. The docking system is removed, as well as the aft bulkhead in the central air lock. This central slotted area is then used to house the two engines. The new module is attached to the IM at a ring near the equipment mounting shelf in the IM. This arrangement allows the CM to be accommodated within the center portion of the IM to a depth of 18 inches (0.45 meters). At this location, the lower edge of the IM barely obscures the forward-most edge of the control station observation windows. Consequently, maximum ground visibility is obtained, and the total length of the vehicle is minimized. The RCS quads on the IM offset 7°30' from the vehicle principal axis to accommodate attachment of the landing gear main struts. The hemipod cargo modules are mounted between the RCS engine quads in opposing quadrants. The secondary or stabilizing struts of the landing gear are attached to the lower edge of the CM.

In general, the configuration described accommodates the orbital and lunar landing operations by use of a common PM and a CM that is relocated from a forward location for orbital to an aft location for landing operations. There are distinct disadvantages to this concept in that, for orbital operations, a second air lock may be required. Also much changing is required to relocate the module (removal or additon of bulkheads and docking systems and pressure hatches). The distinct advantage is the commonality feature of using essentially the same modules for optimum configuration in each mode. The landing configuration offers maximum ground vision and access, as well as ease of cargo handling. The orbital operations configuration offers direct docking interface between the CM and other spaceborne modules such as space station. It also offers maximum payload observation out of the forward end of the CM.

Common CM Location (Top)

The second configuration of a 22-foot-diameter (6.7-meter) vehicle is shown in Figure 2-6 and can be compared to its 15-foot-diameter (4.57-meter) counterpart on Figure 2-1. The overall configuration is



SECTION E – E

INBOARD PROFILE

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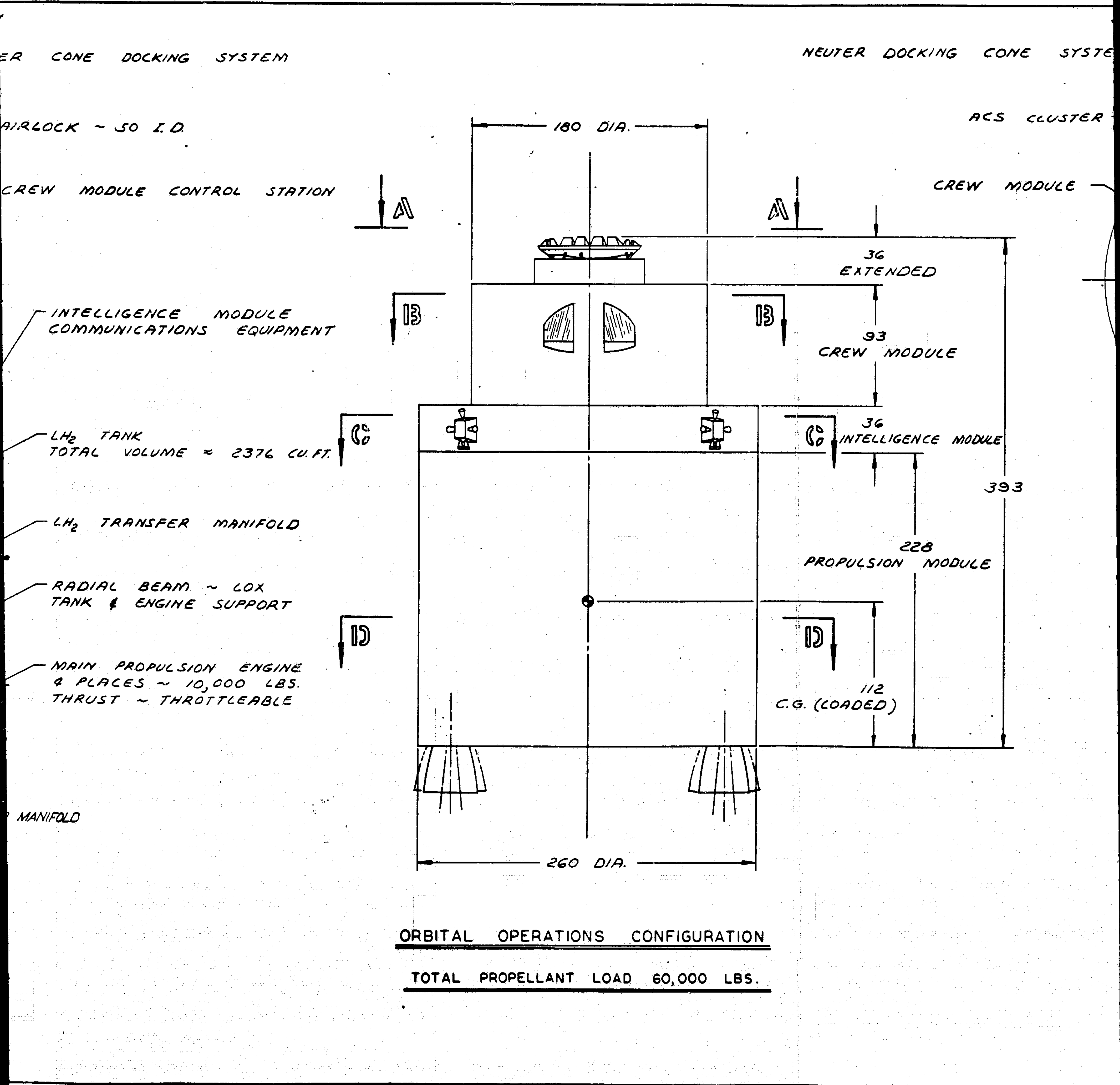

Figure 2-6. 4 Engines - 22-Foot
Core Module Orbital/Operations
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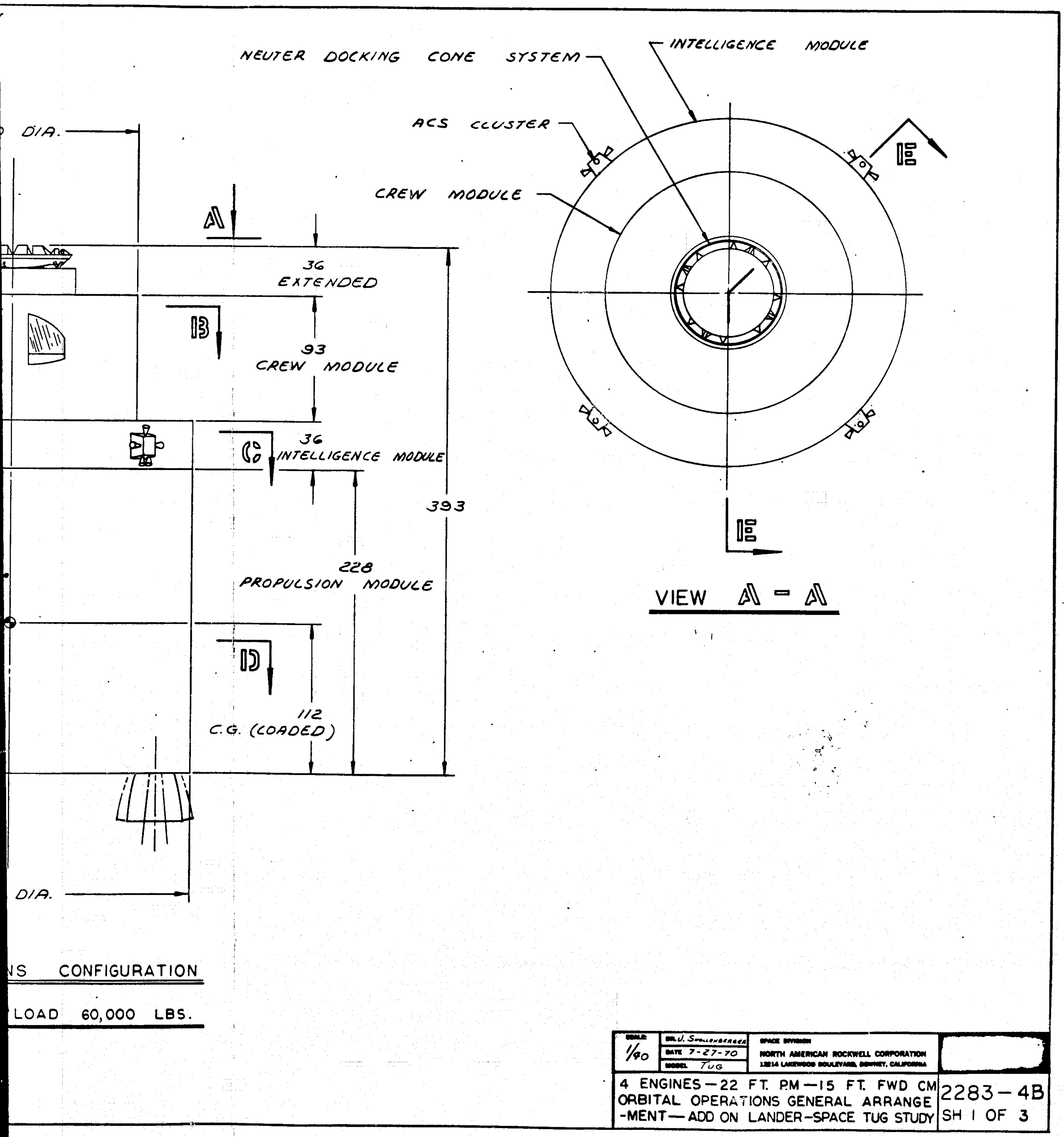
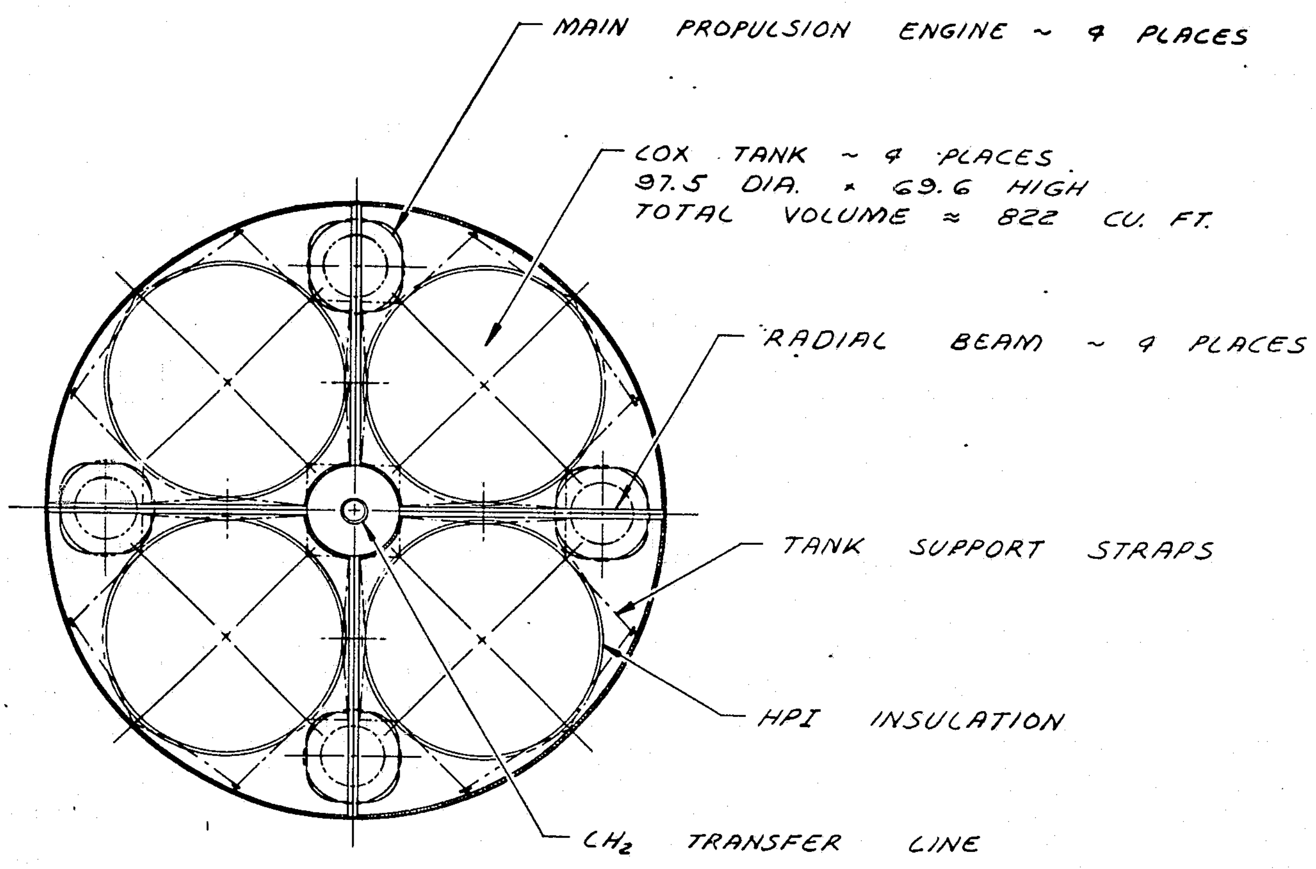
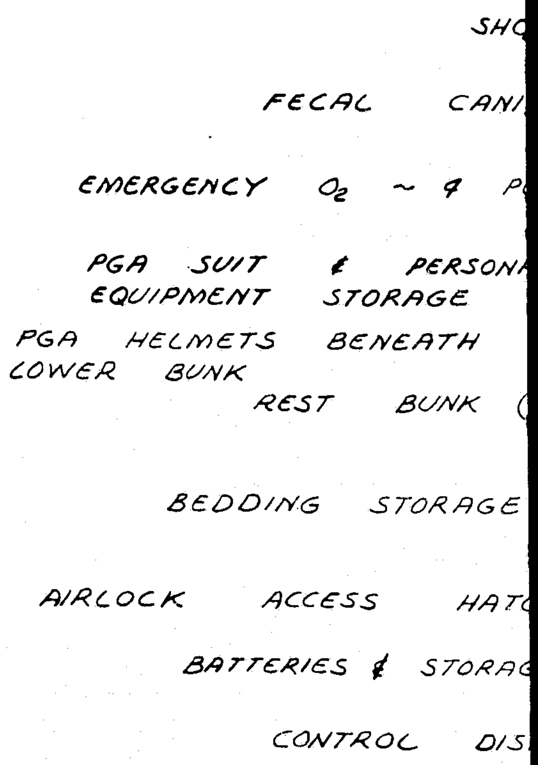


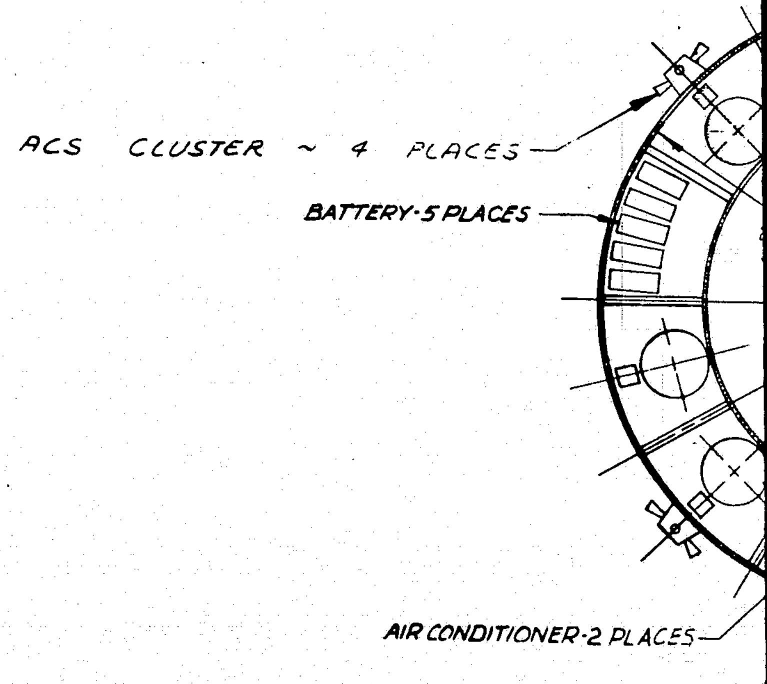
Figure 2-6. 4 Engines - 22-Foot Propulsion Module - 15-Foot Forward Core Module Orbital/Operations General Arrangement - Add-On Lander (Sheet 1 of 3)



SECTION D – D

LOX TANK ARRANGEMENT





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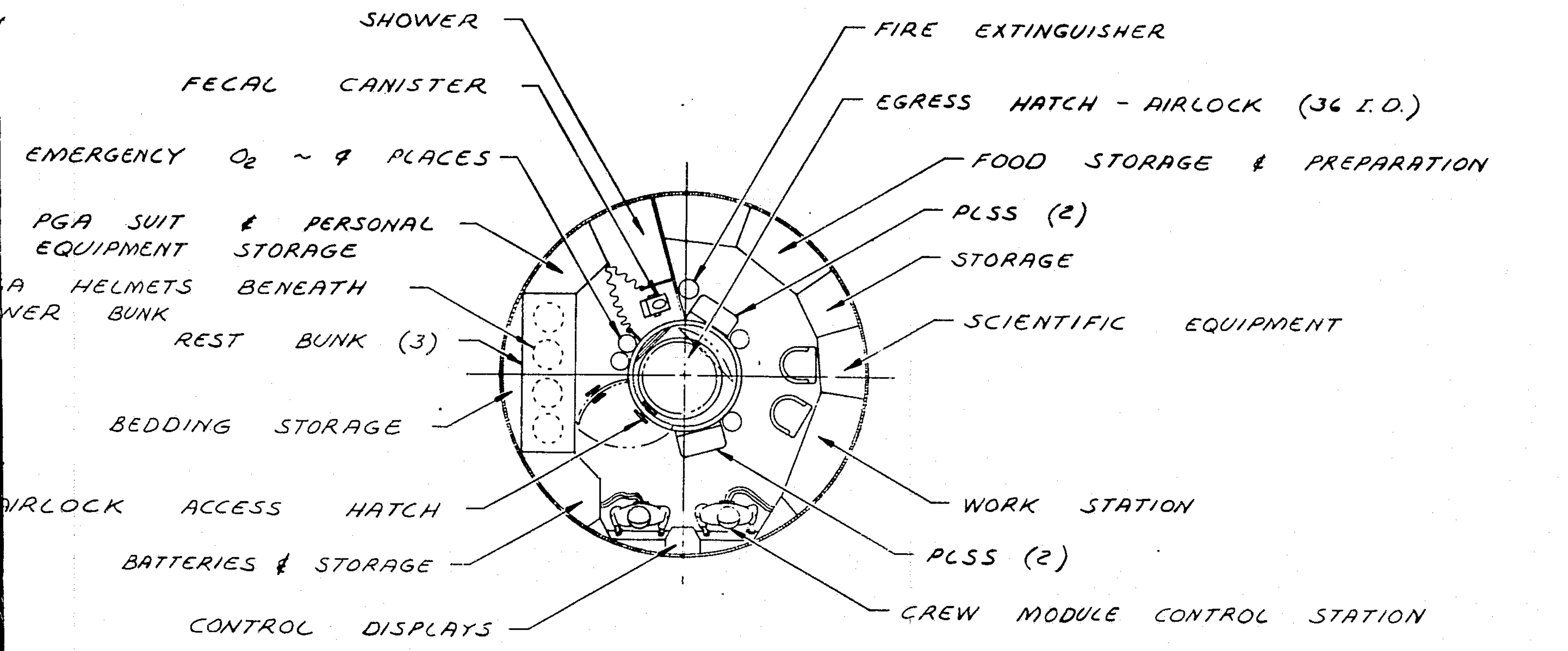
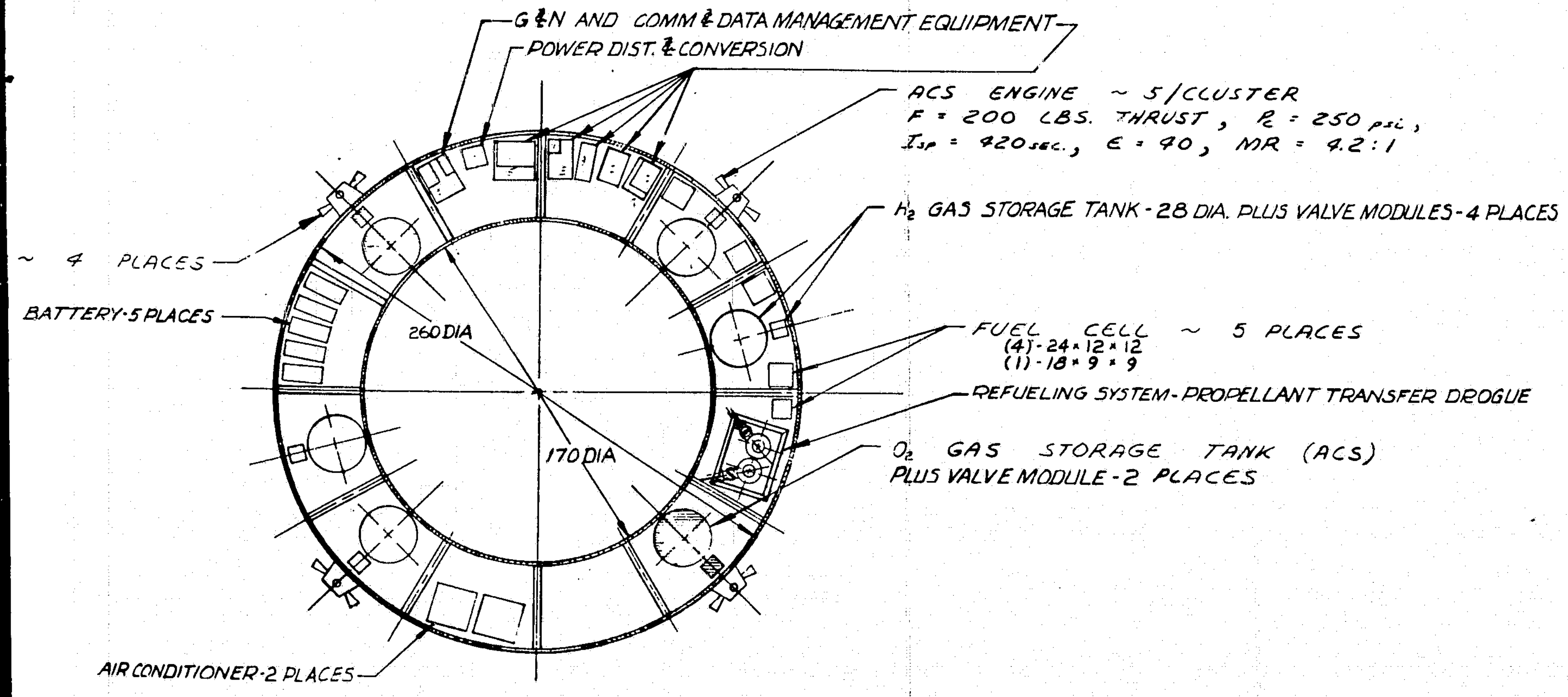


SECTION B - B

CREW MODULE ARRANGEMENT



SECTION C - C

INTELLIGENCE MODULE ARRANGEMENT

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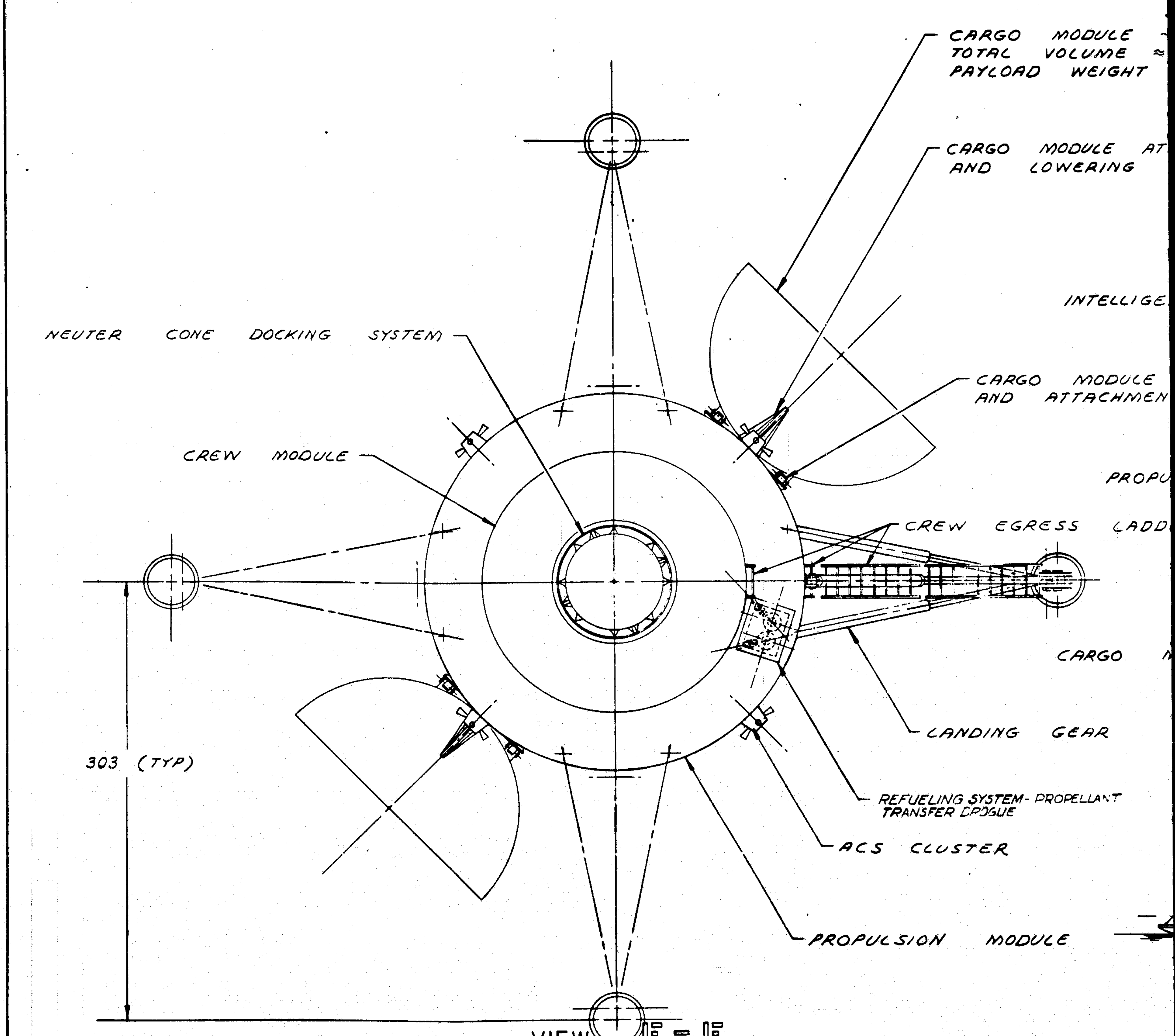
SHEET 2 OF 3

Figure 2-6. 4 Engines - 22-Foot Propulsion Module - 15-Foot Forward Core Module Orbital/Operations General Arrangement - Add-On Lander (Sheet 2 of 3)



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LUNAR LANDER ARRANGEMENT

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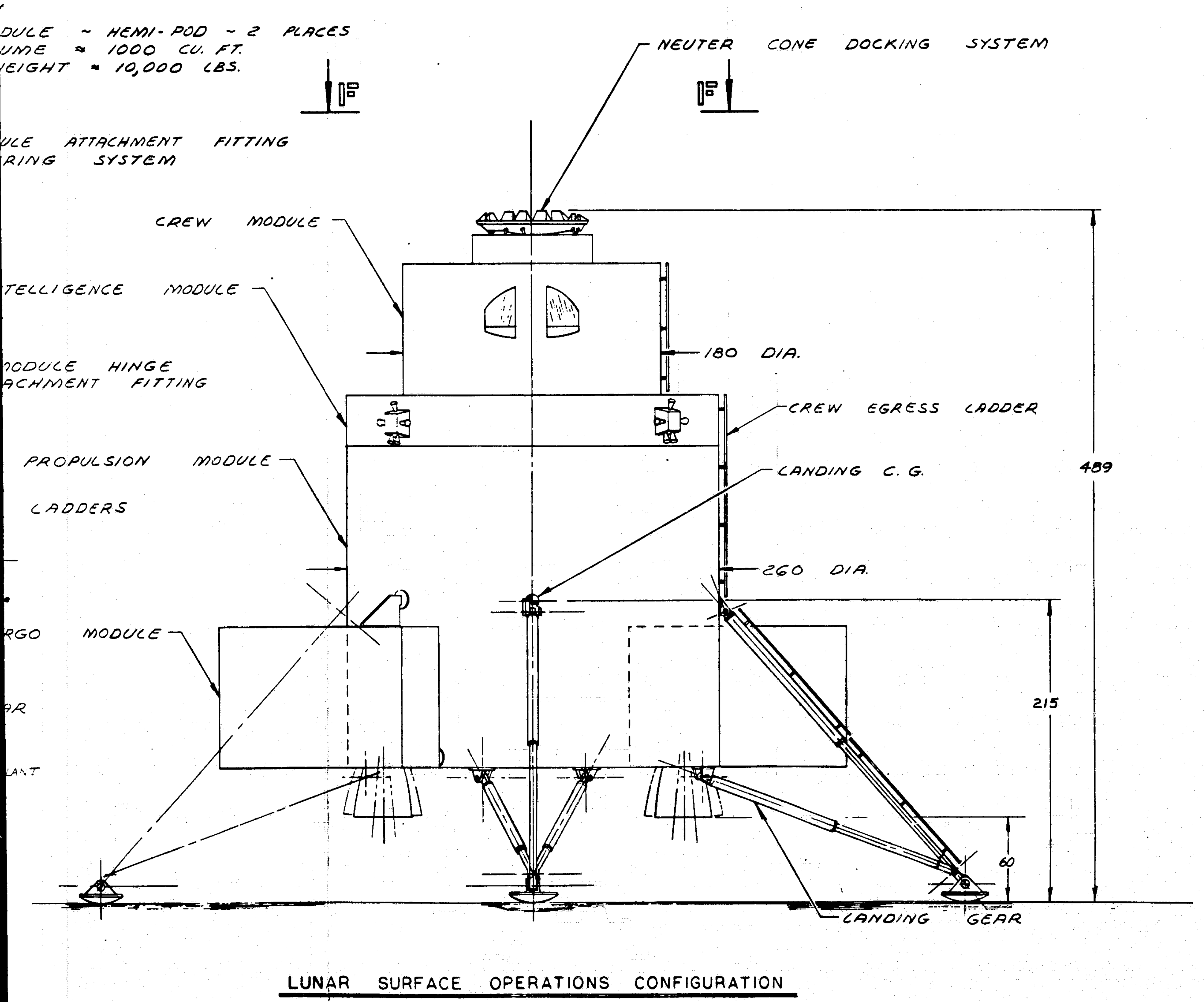

2283 - 4
SHEET 3 OF 3

Figure 2-6. 4 Engines - 22-Foot Propulsion Module - 15-Foot Forward Core Module Orbital/Operations General Arrangement - Add-On Lander (Sheet 3 of 3)

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similar to that shown on Figure 2-4 and previously discussed. This configuration uses a forward-located CM and IM for both orbital and lunar landing operations. For this configuration the CM is identical to that shown in Figure 2-1. The PM is similar to that used on the first 22-foot (6.7-meter) configuration shown in Figure 2-4.

The PM contains one LH_2 tank and four LOX tanks in an arrangement identical to that of Figure 2-4. Four 10,000-pound-thrust (44,482-N) engines are used and are supported from radial beams between the LOX tanks. The engines are mounted outboard within the PM and are located between the LOX tanks such that they do not extend below the aft bulkhead of the PM. Four radial beams support the engines and LOX tanks in the lower part of the module. The LH_2 tank is supported from lighter radial beams in the forward section. The beams also support the IM. The PM structure is completed by the cylindrical outer shell of honeycomb and the honeycomb aft and forward bulkheads. The four lower radial beams are notched to accept the engines and their mounts.

A 22-foot-diameter (6.7-meter) IM is mounted forward of the PM. The IM is a square section torus with a 170-inch (4.3-meter) inside diameter. It is attached to the upper radial beams of the PM, and its forward surface is the interface for attachment of the CM. The IM is constructed of honeycomb (inner shell, outer shell, forward and aft bulkheads, and equipment mounting shelf). The equipment is mounted to the shelf and assembled into the vehicle in segments. This arrangement allows for a maximum of growth in the individual subsystem areas. The IM also houses four sets of RCS quads and equipment.

A 15-foot-diameter (4.57-meter) CM is supported from the forward surface of the PM. This module is a pressure vessel constructed of a honeycomb outer cylindrical shell, a centrally located air lock, and forward and aft bulkheads of honeycomb. The forward bulkhead supports a neuter cone docking system in a central position above the air lock. The air lock is circular and has an inside diameter 50 inches (1.3 meters) with a pressure hatch forward for crew egress through the docking system. The internal arrangement of the CM offers maximum space for crew work and living. The air lock is the only unusable area in the module from the standpoint of accommodating stores or personnel working in a shirtsleeve environment. The neuter cone docking system enables the tug to dock with all spaceborne elements, using either the active or passive part of this system. The forward-mounted CM offers maximum visibility of payloads being docked.

For lunar landing operations, no changes are made to the basic configuration. Landing gear and cargo modules are added to the structure, and crew egress ladders are added. The main landing gear struts are attached to the lower radial beams, and the secondary or stabilizing struts are

attached to the aft edge of the propulsion module. The RCS quads on the IM are located between the landing gear legs. Cargo is transported to the lunar surface in two hemipod cargo modules of 15-foot-diameter (4.57-meter) total. This allows the cargo modules and CM (also 15 feet in diameter) (4.57-meters) to be transported in the earth orbiting shuttle cargo bay. The cargo modules are attached at the lower edge of the PM beneath two of the RCS quads. The modules are more than 10 feet (3.0 meters) from the aft firing RCS engine exit plane, which should not make impingement of the exhaust plume a problem from either a thermal protection or attitude control viewpoint. The crew leaves the CM through a neuter cone docking system and then descends a series of ladders down the side of the IM-PM combination and down ladders affixed to the landing gear main strut to the lunar surface.

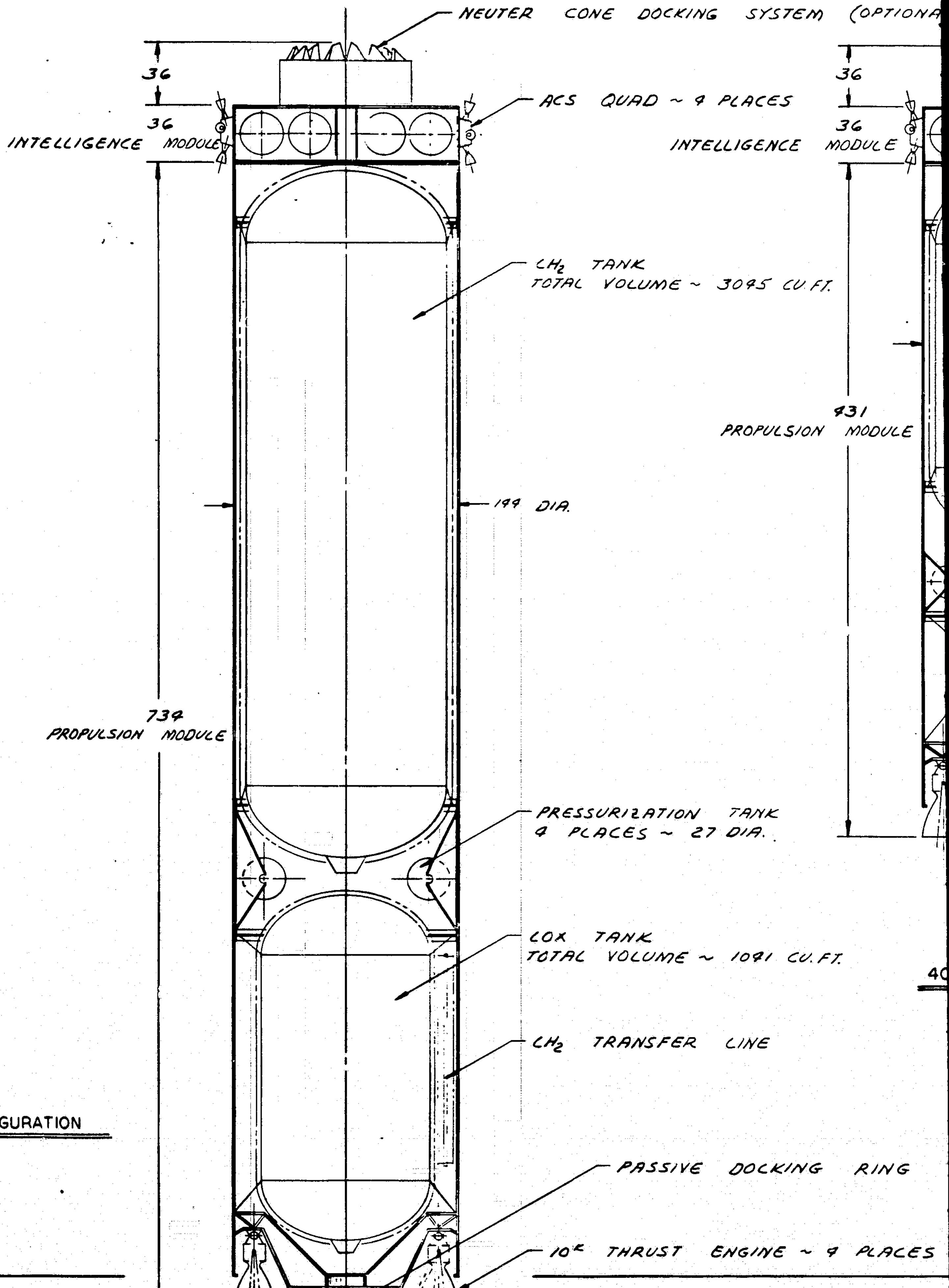
The configuration arrived at is a minimum of height and would only be 6 inches (0.15 meters) shorter if the IM were completely integrated with the PM. The PM is at a minimum height with a liberal clearance between the LH_2 and LOX tanks. Both LH_2 and LOX tanks are ellipsoid with 1.4 bulkhead ratios. The engines are mounted within the PM outer mold line without requiring a minimum of clearance between the engine excursions and the LOX tanks. A minimum height of 5 feet (1.5 meters) has been used for the distance from the engine nozzle exit plane to the lunar surface. With these considerations taken into account, the present configuration is believed to be the shortest practical 22-foot-diameter (6.7-meters) vehicle for a 60,000-pound (27,215-kilograms) propellant load.

For orbital operations, the vehicle affords excellent visibility and interface features. It affords a minimum-height lunar surface lander but poor visibility with a forward (top) CM. The cargo modules are easily accessible from the ground in the lander version. The surface-descent ladders may appear awkward, but the actual distance to the lunar surface from the CM is under 28 feet (8.5 meters). Another approach to crew descent would be an elevator assembly of the type shown in Figure 2-1.

2.2.3 12-FOOT-DIAMETER (3.66-METER) CONFIGURATION STUDY

Parametric Sizing

For the full range of feasible tug vehicle diameters (12, 15, and 22 feet) (3.66, 4.57 and 6.7 meters), Figure 2-7 was prepared to encompass the 12-foot-diameter (3.66-meter) family. Three configurations that are shown were selected for comparison with their larger-diameter counterparts. Selected propellant loadings (usable) were 20,000, 40,000, and 80,000 pounds (9074, 18,147 and 36,297 kilograms). The three configurations are identical, except for the length of the cylindrical section of the LH_2 and LOX tanks. Each configuration used a single LH_2 tank and LOX tank, a forward-mounted IM, an optional forward-mounted neuter cone docking system, four 10,000-pound-thrust (44,482-N) throttleable



80,000 LB. PROPELLANT CONFIGURATION

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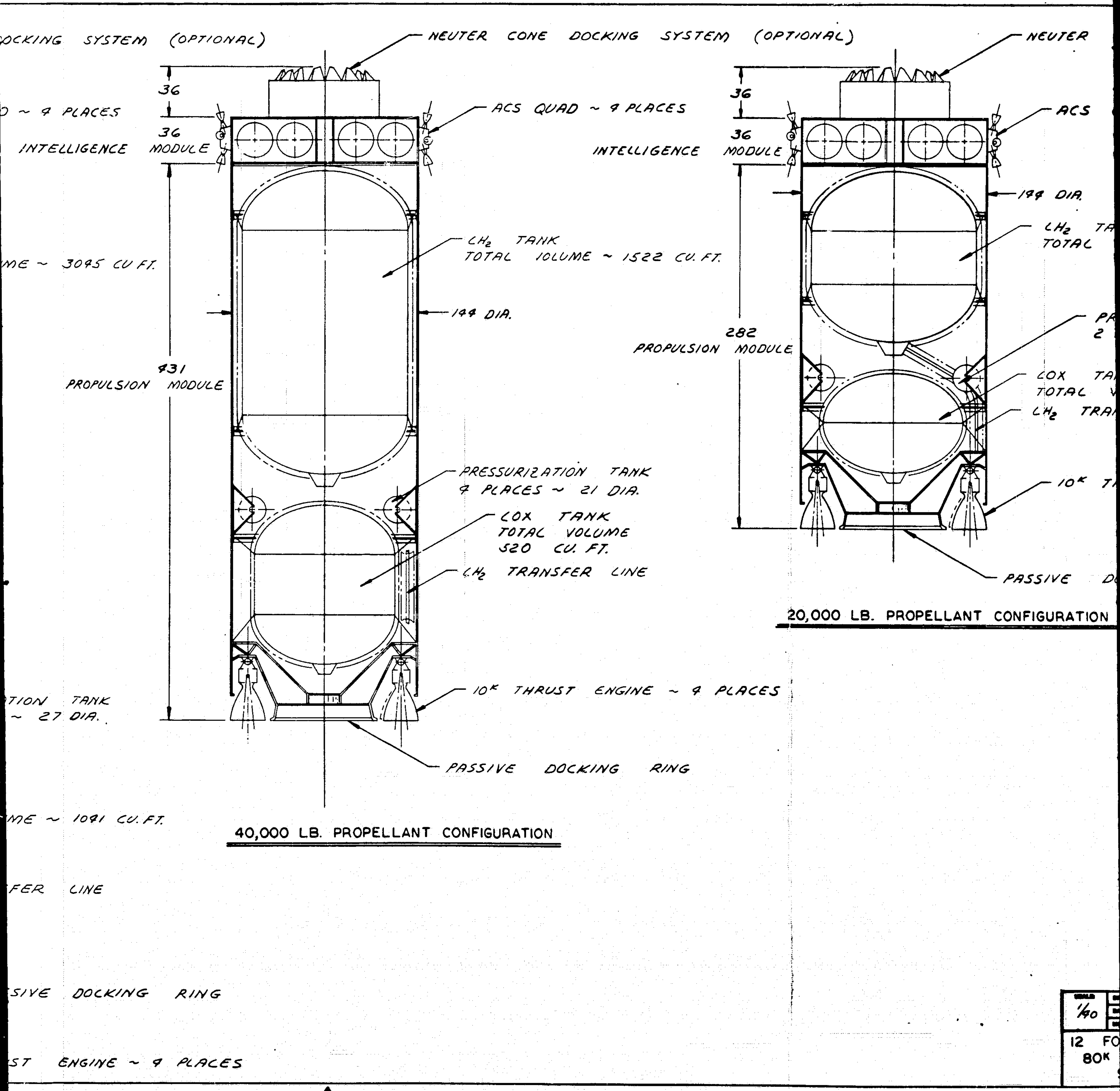

Figure 2-7. 12-Foot Diameter - 20K, 40

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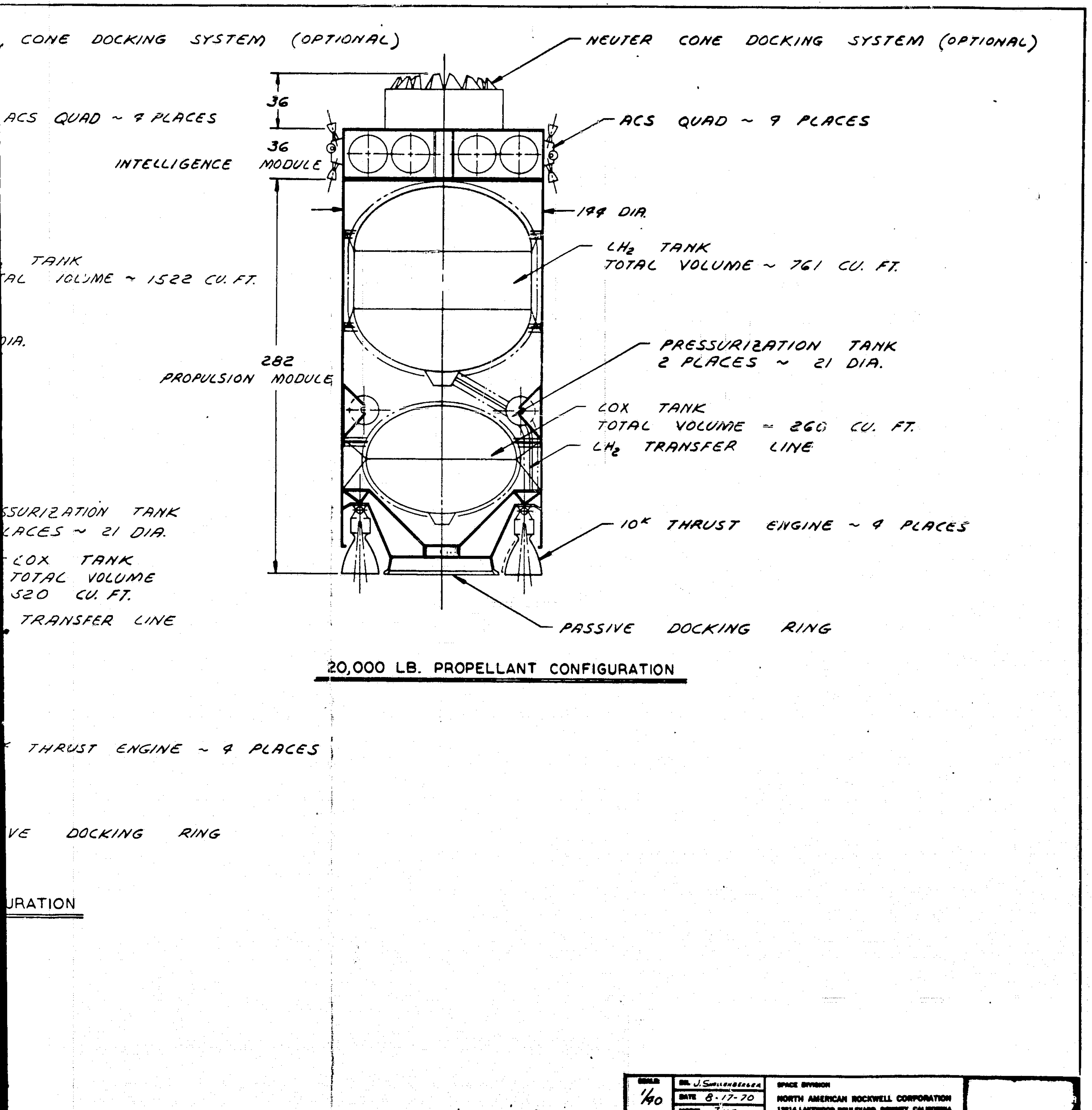


Figure 2-7. 12-Foot Diameter - 20K, 40K, and 80K Propellant Concepts

engines, and a passive docking ring at the aft end. No new modules were shown in the configurations.

In each configuration the basic propulsion module structure is identical. This structure consists of a honeycomb cylindrical shell with a forward honeycomb bulkhead. At the aft end are the four main engines and passive docking ring. Each engine is supported from a radial beam attached to the cylindrical shell at the forward end. The other end of each of the four radial beams flairs out to support the passive docking ring.

Another approach to the support of the engines and the docking ring would be a combination cylindrical and conical adapters of skin stringer construction. The cylindrical portion would support the docking ring and allow clearance for the engine gimbaling. The conical piece would adopt the cylindrical section to the propulsion module shell. The engines would be supported from fittings on the conical section.

The outer shell of the basic structure is scalloped in the areas of the four engines to allow clearance for the engines in a fully gimbaled position. Circumferential rings on the outer shell structure are located fore and aft of the LH_2 and LOX tank girth rings to support each tank. The upper part of each radial beam also interfaces with the aft LOX support ring on the basic shell. This concept is identical to each of the three configurations shown.

The IM is mounted on the forward end of the propulsion module. Its shape is not toroidal as are its 15- and 22-foot (4.57- and 6.7-meter) counterparts. This intelligence module, being 12 feet (3.66 meters) in diameter, requires all of the circular area for mounting of the equipment. Consequently, this module is wafer shaped. The equipment is put into the module in segments similar to the other larger diameter modules, but the segments are not neat pie-shaped wedges. The structure is basically a honeycomb fore and aft bulkhead with a central honeycomb shelf. The outer honeycomb panels are removed in sections with each segment. The available volume of this module is approximately 340 cubic feet (9.6 cubic meters) which is about 60 cubic feet (1.7 cubic meters) less than that available in the 15-foot-diameter (4.57-meter) version. This loss does not appear severe when it is considered that the 15-foot (4.57-meter) module had an excess of growth volume. The 15-foot-diameter (4.57-meter) IM had a packing density of about 9 pounds per cubic foot (144 kilograms per cubic meter). The 12-foot-diameter (3.66-meter) version exhibits about 10 pounds per cubic foot (160 kilograms per cubic meter) packing density. All of the subsystem equipment is located in the IM. The ACS engine quads are located about the periphery of the module.

A neuter cone docking system is attached to the forward end of each vehicle. This system is an optional item. When required, it lengthens the vehicle by the actual length of the mechanism since it cannot be buried partially within the IM. The passive docking ring on the aft end of each vehicle will enable it to dock with the one active docking cone on the space station. This ring also enables two stages to be put together in tandem fashion to perform missions requiring more delta V than is available with one stage. Such an arrangement is separated without destroying either vehicle's docking mechanism.

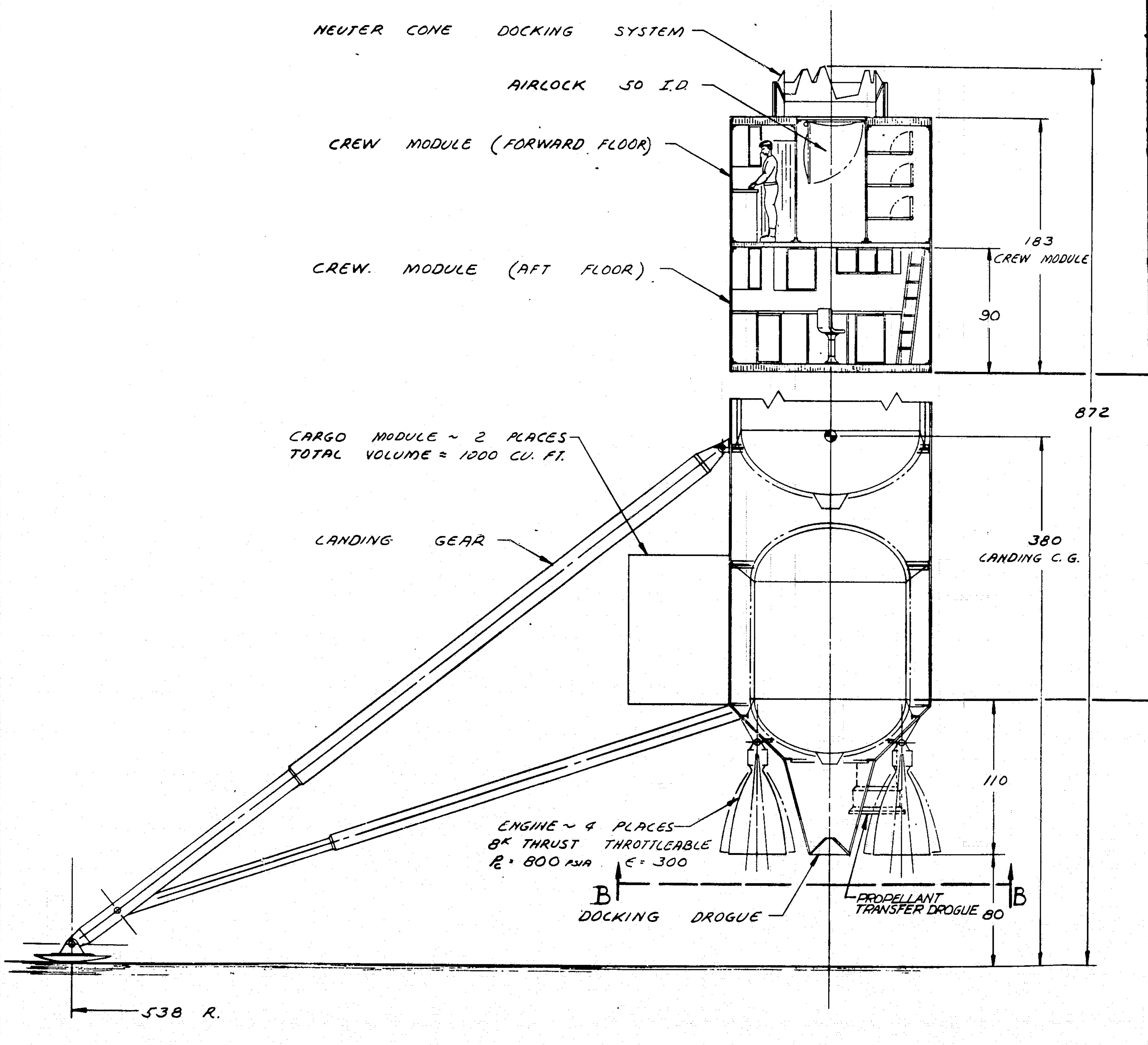
The tanks used in the propulsion module are cylindrical with 1.4 to 1 ratio elliptical bulkheads. The LOX tank is of smaller diameter than the LH_2 tank to allow sufficient clearance between the tank wall and insulation and the propulsion module inner mould line for the LH_2 transfer lines to the engines.

The 20,000- and 40,000-pound (9074- and 18,147-kilogram) propellant loading vehicles are not extremely lengthy being 30 and 40 feet (9.1 and 12.2 meters) overall, respectively. The 80,000 pound (36,297 kilogram) propellant vehicle, however, is longer than 67 feet (20.4 meters). When it is considered that these lengths do not include a payload or a new module, the 80,000-pound (36,297-kilogram) version is much too long to be feasible. The other two configurations, however, do not appear unwieldy even when seven or eight feet are added to their lengths for a CM.

Typical Point Design for Lunar Lander

The concepts shown in Figure 2-8 are configured for a 12-foot-diameter (3.66-meter) lunar landing vehicle in a single-stage mode. The usable propellant load is 55,000 pounds (24,974 kilogram). Two configurations are shown. They are based on four very high-pressure (1500 psia, 1034 N/cm^2) engines and a single-deck CM and four high-pressure (800 psia, 552 N/cm^2) engines with a double-deck CM. The first concept is configured basically for comparison with its 15- and 22-foot-diameter (4.57- and 6.7-meter) counterparts. This configuration has a single LH_2 and a single LOX tank. An IM is mounted on the forward end. The single deck CM is located forward of the IM and incorporates a neuter cone docking system at its forward end. Four very high-pressure engines (P_c = 1500 psia, 1034 N/cm^2) are fixed-mounted and clustered about a passive docking ring at the aft end.

Use of a single LOX tank gives the shortest configuration when compared to a concept with four LOX tanks with the engines between the tanks. The four engines are supported by a conical skin-stringer structure at the aft end of the cylindrical section. This conical structure also supports the passive docking ring. The IM is wafer shaped and does not have a hole in its center as does the 15-foot-diameter (4.57-meter) IM's. The equipment



FOLDOUT FRAME

1

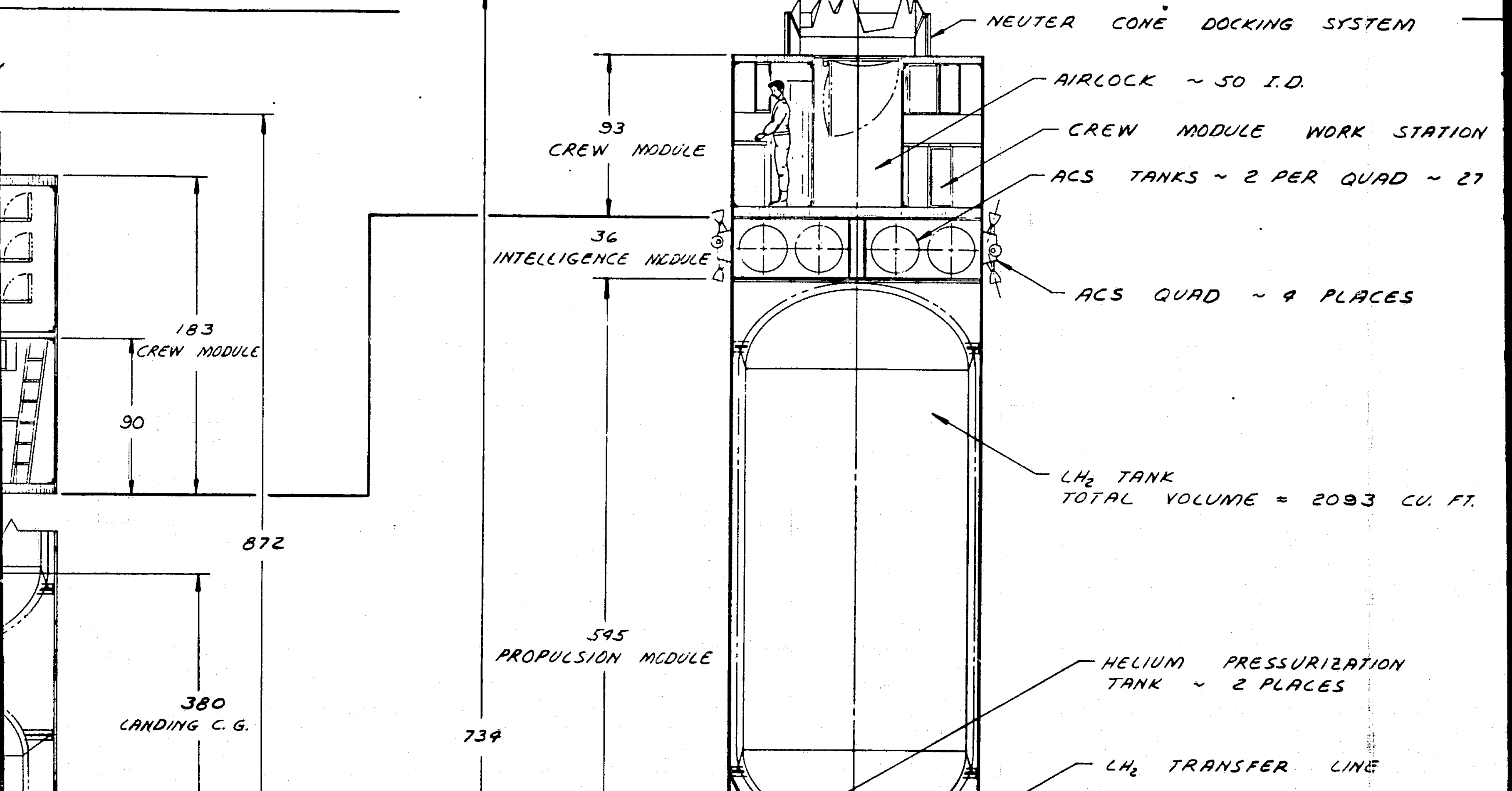


Figure 2-8. 55K Po

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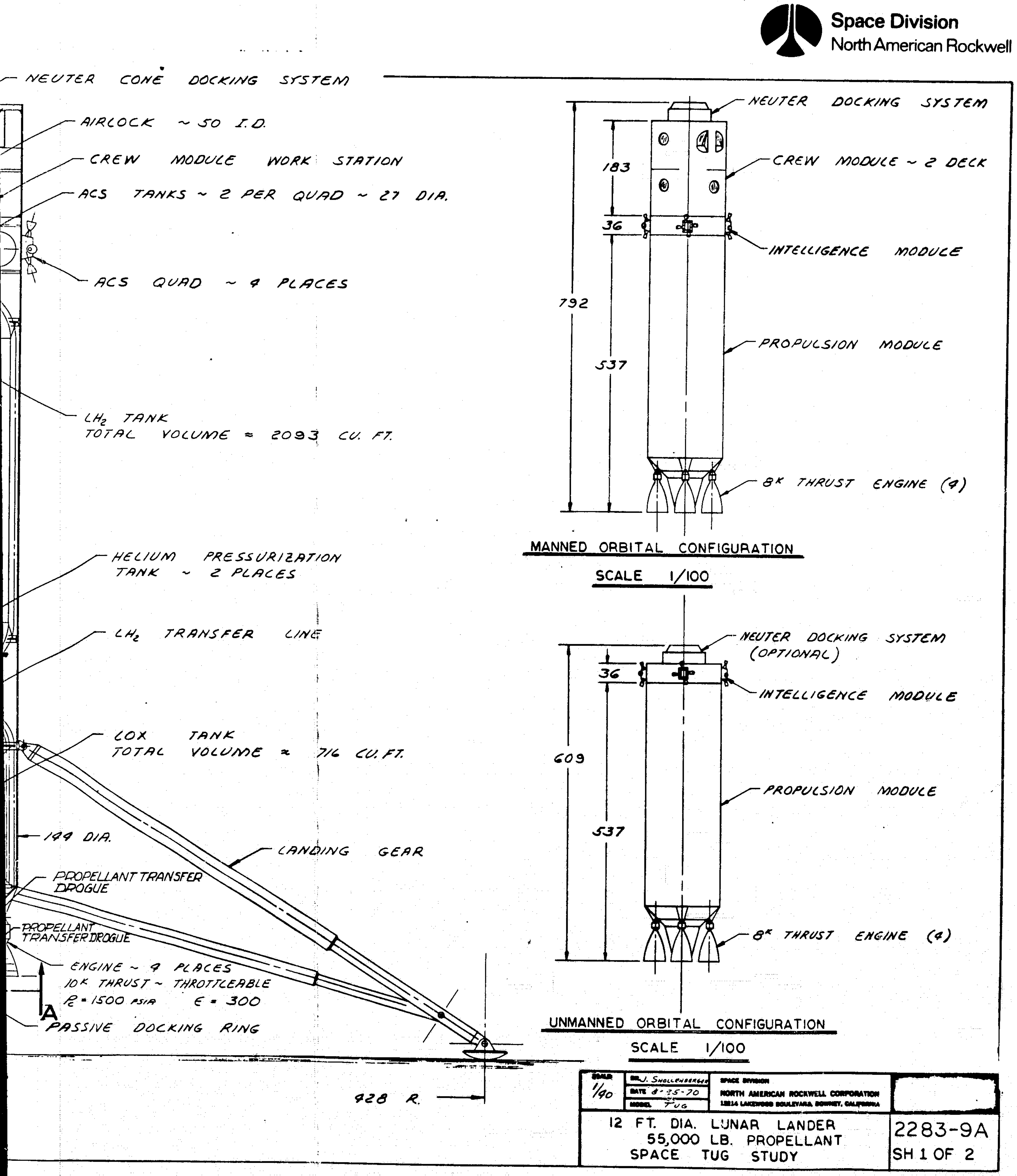

Figure 2-8. 55K Pound Propellant, 12-Foot-Diameter Lunar Lander (Sheet 1 of 2)

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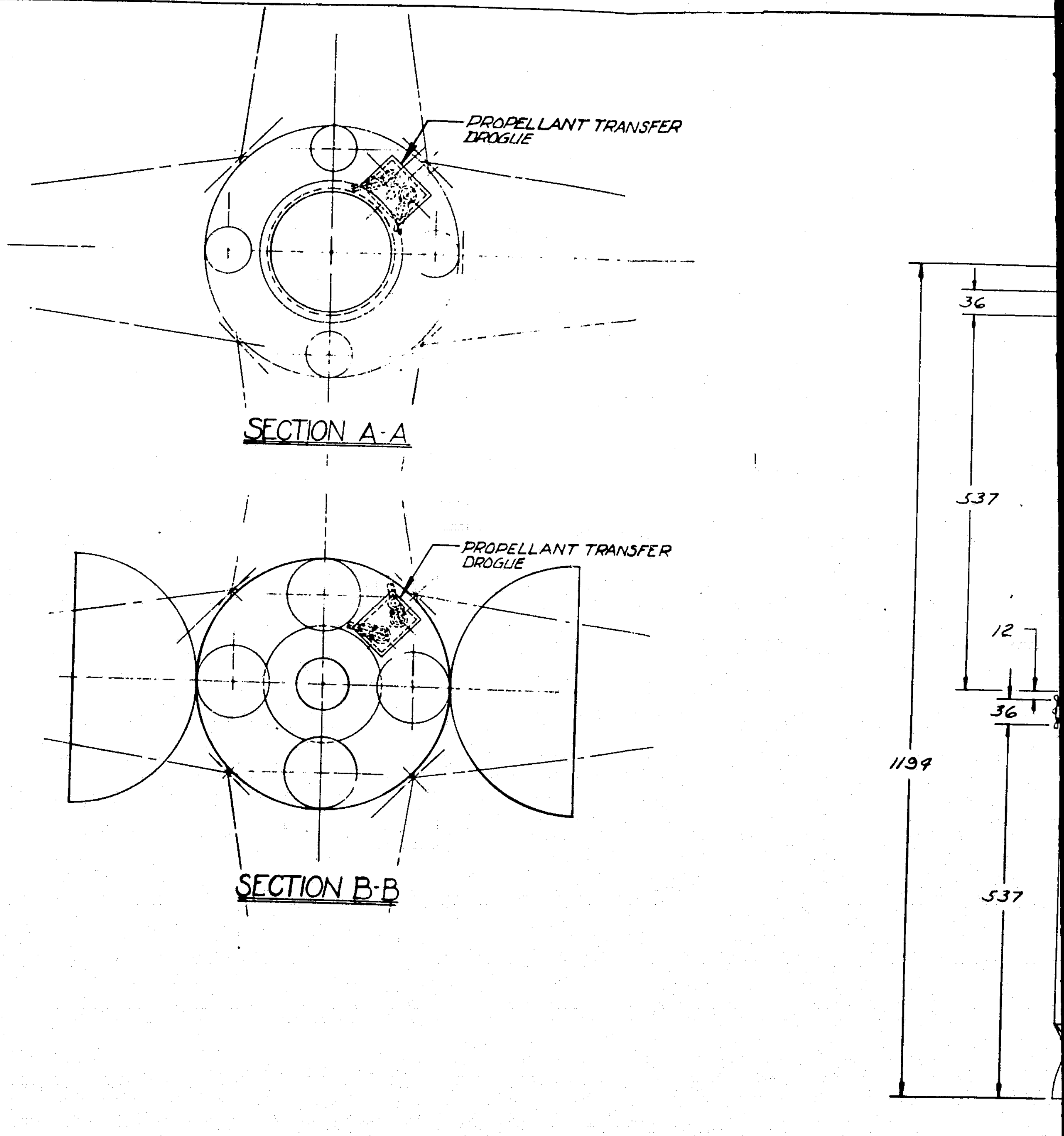


GEOSYNCHRONOUS

MODE B3, BOTH

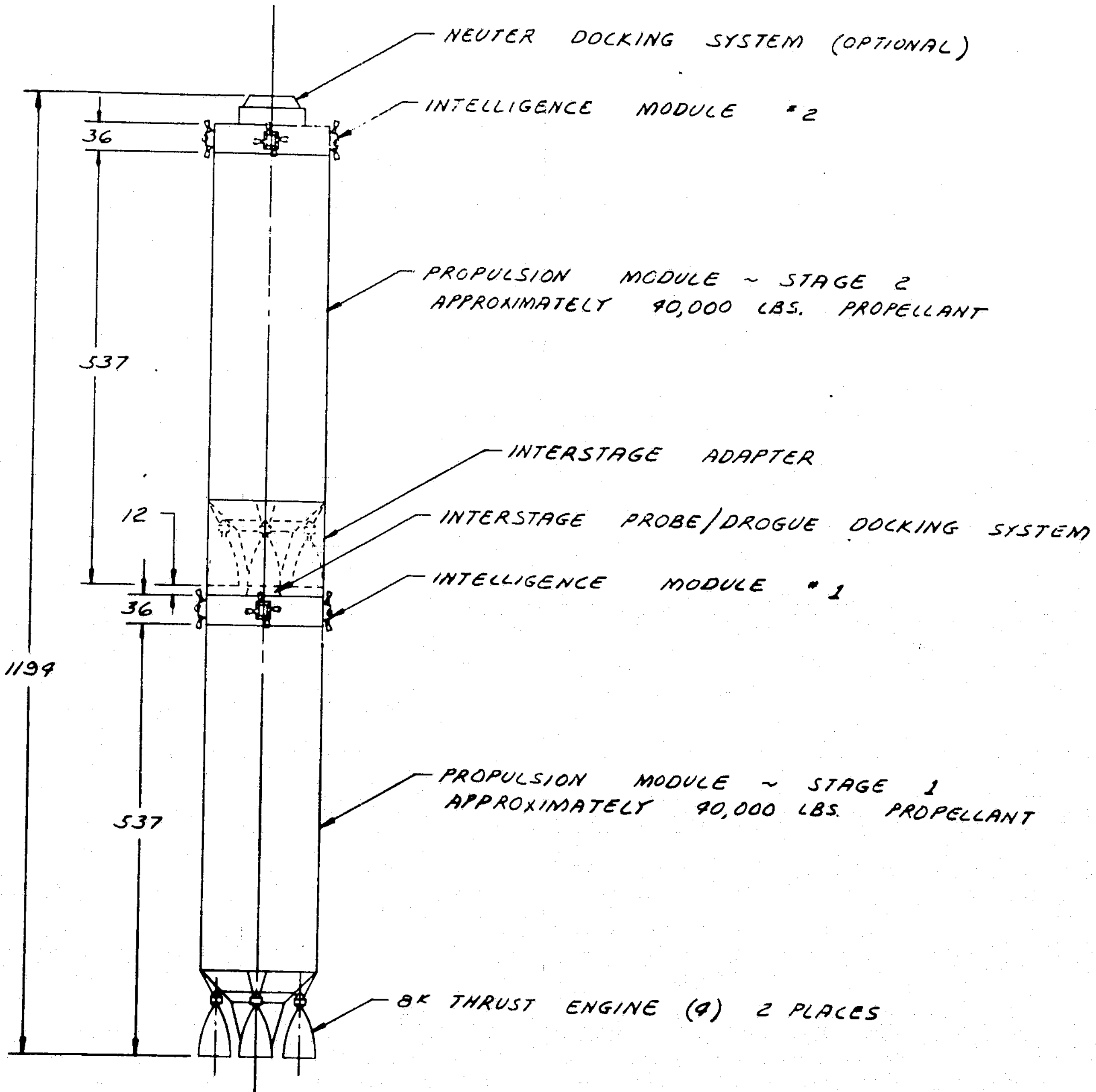
SCALE

Figure 2-8.

FOLDOUT FRAME 1

TRANSFER

ANSFER



GEOSYNCHRONOUS MISSION CONFIGURATION
MODE B3, BOTH STAGES RECOVERED

SCALE 1/100

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SHEET 2 OF 2

Figure 2-8. 55K Pound Propellant, 12-Foot-Diameter Lunar Lander (Sheet 2 of 2)

and volume required for stowage of the ACS engine clusters uses nearly all the available volume within the IM. The IM consists structurally of a forward and aft honeycomb bulkhead with a centrally located honeycomb shelf. The cylinder is closed out by removable honeycomb outer covers. Four of these covers providing mounting for the ACS clusters. These four covers with clusters are capable of being rotated 180 degrees to stow the clusters within the 12-foot-diameter (3.66-meter) outer moldline.

The CM shown in this concept is a single-deck version with a centrally located air lock. The air lock is 50 inches (1.3 meters) in diameter and interfaces at the forward end with a neuter cone docking system. The available volume within the CM for living quarters and equipment (does not include air lock) is approximately 680 cubic feet (19.3 cubic meters) as compared with the 1110 cubic feet (31.4 cubic meters) available in the 15-foot-diameter (4.57-meter) vehicle. The smaller overall diameter (12 feet) (3.66 meters) in conjunction with the air-lock diameter (50 inches, 1.3 meters) allows only approximately a four-foot toroid area for crew movement and equipment. When the work and equipment station are subtracted from this floor area, the actual area for crew movement becomes marginal for a comfortable environment.

Because of the reduced volume available in the 12-foot-diameter (3.66-meter) CM, it is felt that a two-deck module is required to be comparable to the 15-foot-diameter (4.57-meter) configuration.

The two-deck module is shown on the second configuration. The second deck adds 770 cubic feet (21.8 cubic meters) of volume to the first deck for a total of 1450 cubic feet (41.1 cubic meters). There is no air lock in the second deck, only a pressure hatch between the two decks. In the event of a pressure failure in either deck, that deck could be isolated from the other. The air lock can be entered from either deck, through the side in top deck or through the aft end in the second deck.

This configuration also depicts the high-pressure (P_c = 800 psia, 552 N/cm^2) engines, which are close to state-of-the-art-type engines. To accommodate these engines, an Apollo-type probe and drogue docking systems has been used at the aft end. The four engines have been clustered about a drogue that is supported from a skin-stringer conical structure. The engines are also supported from this conical structure. As can be seen from a comparison of these two configurations, the increase in length of the basi_ vehicle is 118 inches (3.0 meters) to accommodate the two-deck crew module and the high-pressure engines. The revised configuration is 792 inches (20.1 meters) long and could not be stowed with a 60-foot (18.3-meter) shuttle orbiter payload bay. The first configuration is 674 inches (1.71 meter) long and would fit in the payload bay. However, it is felt that this concept is not adequate in CM volume, and the engines would require an increase in the state of the art.

The tall thin configurations that arise out of the 55,000-pound (24,974-kilogram) propellant vehicles exhibit exceedingly high center of gravities. Because of these high c.g.'s, the landing gear spread is large (428- and 538-inch (10.9- and 13.7-meter) radius for the concepts shown), which requires a large and heavy gear. It is estimated that the gear required would increase in weight by nearly 100 percent over that required on 15- and 22-foot-diameter (4.57- and 6.7-meter) vehicles of the same propellant weight. Crew egress to the lunar surface would be either by an elevator assembly or a series of ladders affixed to the side of the vehicle. The extremely long distance from the CM upper bulkhead, where the crew would begin its descent to the lunar surface, a distance of 60 and 70 feet (18.3 and 21.3 meter) would probably preclude use of ladders. An elevator assembly is the most practical system. This assembly could be similar to that shown on the 15-foot diameter (4.75-meter) vehicle in Figure 2-1. It would be stowed on the top of the forward bulkhead of the CM and would pivot into the operational position. Braces and other hardware would also be stowed on the bulkhead which would ease assembly of the system.

These vehicles are refueled through a drogue near the aft end of each vehicle. The drogues are located between two adjacent engines. The refueling assembly is supported from the aft conical thrust structure and is closed off by a pivoting door that is opened when the refueling supply probe is being inserted into the drogue.

Also shown on this drawing are overall views of the manned and unmanned orbital configurations, as well as a geosynchronous mission configuration. The manned orbital configuration shown is the same as the two-deck CM lander without the landing gear. This concept accommodates four high-pressure engines (P_c = 800 psia, 552N/cm^2) and a neuter cone docking system on the CM. The configuration is 792 inches (20.1 meters) long. The unmanned concept removes the CM and can accommodate a neuter cone docking assembly at the forward end as an optional item. Overall the configuration is 609 inches (15.5 meters) long, including the neuter cone docking system, or 573 inches (14.5 meters) if the neuter cone is removed.

The last configuration shown on the drawing is a geosynchronous concept that uses two basic vehicles in tandem. These vehicles would both be recoverable and would probably be offloaded to perform the mission. A probe-and-drogue docking system is used to join the two vehicles, as well as a cylindrical interstage about 118 inches (3.0 meters) long. All parts of both vehicles are recoverable.

Hemipod cargo modules 12 feet (3.66 meters) diameter are used in each lander configuration. The pods are located as far aft as practicable from an attachment standpoint.

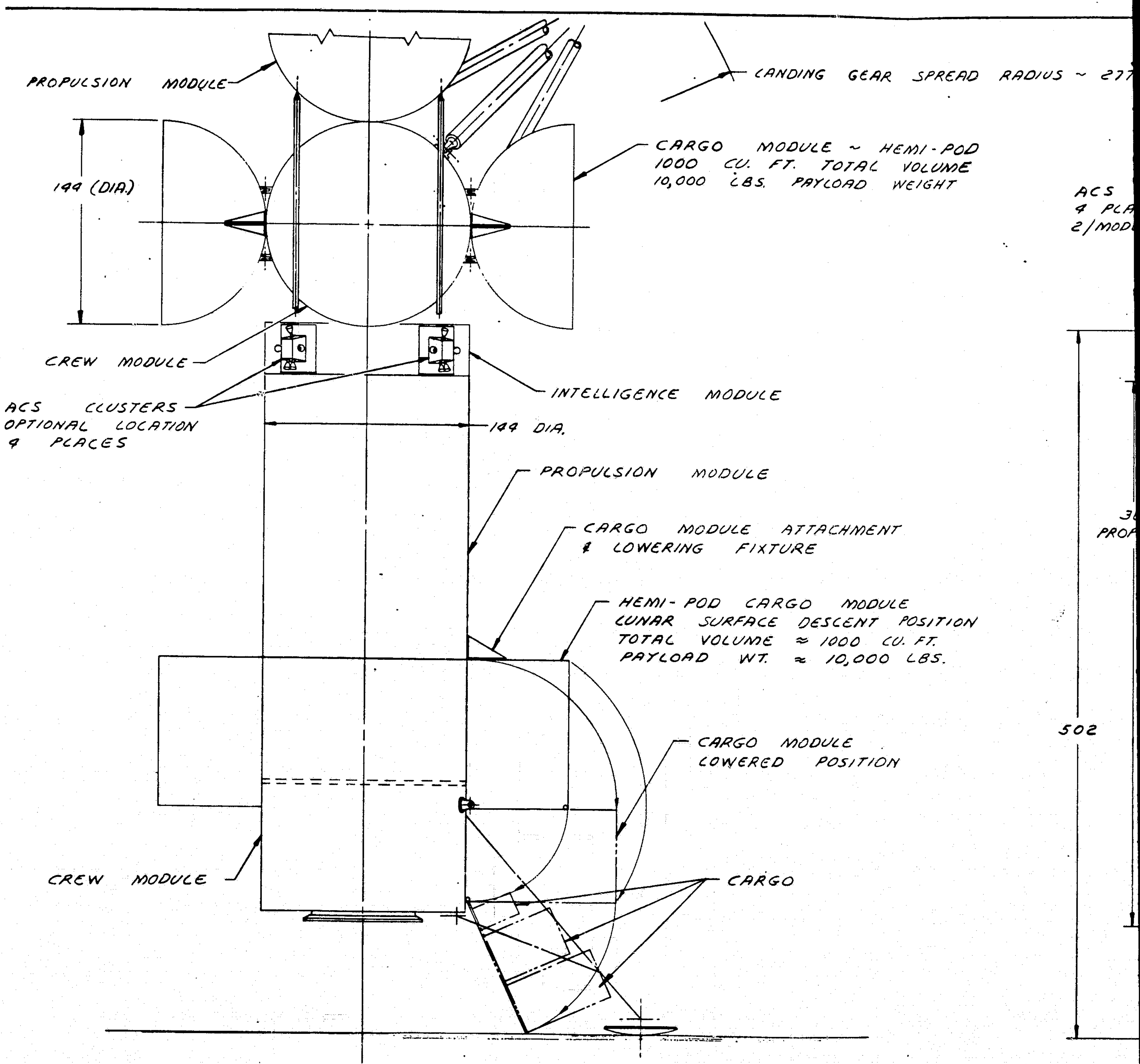
Parallel and Tandem PM Landers

The configuration shown in Figure 2-9 is based on a lunar lander parallel-vehicle concept. Each vehicle is sized for 30,000 pounds (13,607 kilograms, of usable propellant. Two vehicles are arranged in a parallel fashion, with a crew module and cargo module between them. All modules are 12 feet (3.66 meters) in diameter. Each propulsion module incorporates one LH_2 and one LOX tank. Four high-pressure (P_c = 800 psia, 552 N/cm^2) engines are clustered about an Apollo-type drogue at the aft end. A conical skin-stringer-type thrust structure supports the engines and the docking drogue. Between two adjacent engines on the aft end of each vehicle is a refueling connection drogue assembly. This assembly is also supported from the thrust structure. A cover is provided over the connection, which is pivoted out of the way during refueling.

The two PM's are structurally tied together in several points. At the forward end of each PM, a structural tie exists between the parallel vehicles. Farther aft, at the crew module location, the ties are between propulsion module and crew module at several locations. At the forward end of each propulsion module an intelligence module is located.

The intelligence module is constructed of a forward and aft honeycomb bulkhead with a centrally located honeycomb equipment shelf. The bulkheads are wafer shaped, and the cylinder is closed out externally by honeycomb covers. Four covers are removable for equipment access, and four others provide mounting for the ACS clusters. The ACS clusters pivot 180 degrees such that the entire cluster may be stowed within the IM 12-foot-diameter (3.66-meters) outer moldline. The IM houses all subsystem equipment, as well as the entire ACS system. There is no docking provision at the forward end of each PM since a drogue is provided at the aft end for docking.

A 12-foot-diameter (3.66-meter) two-deck crew module is mounted between the propulsion modules at the aft end of the vehicle. The first deck contains the control station and crew quarters, as well as the centrally located air lock. At the aft end of the air lock is a passive docking ring. Crew egress to the lunar surface is through the docking ring. Crewmen also transfer to the space station through this ring. The passive docking ring permits the tug to dock at the space station aft end, which contains an active neuter cone docking system. The second deck of the crew module contains equipment and work stations. This deck is separated from the first by a pressure hatch but has access to the air lock from the forward end through the second deck floor. There is no docking provision on the forward end of the crew module since it is buried aft of the propulsion module forward end and would not be easily accessible. Two 12-foot-diameter (3.66-meter) hemipod cargo modules are attached to the crew module at an elevation just forward of the control station landing windows. These pods can be lowered



VIEW A - A

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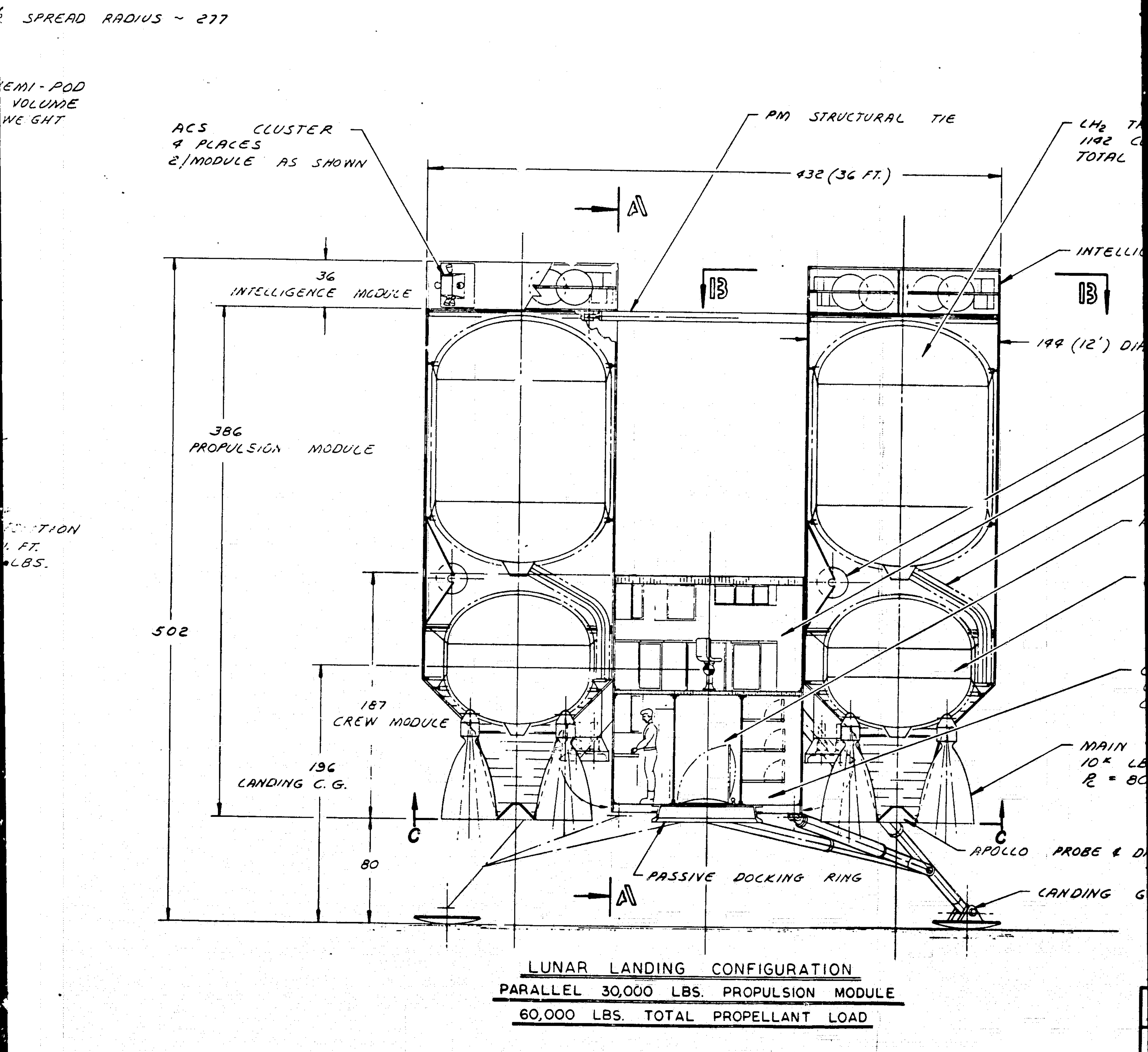

Figure 2-9. 12-Foot-Diameter Para
Pound Propu

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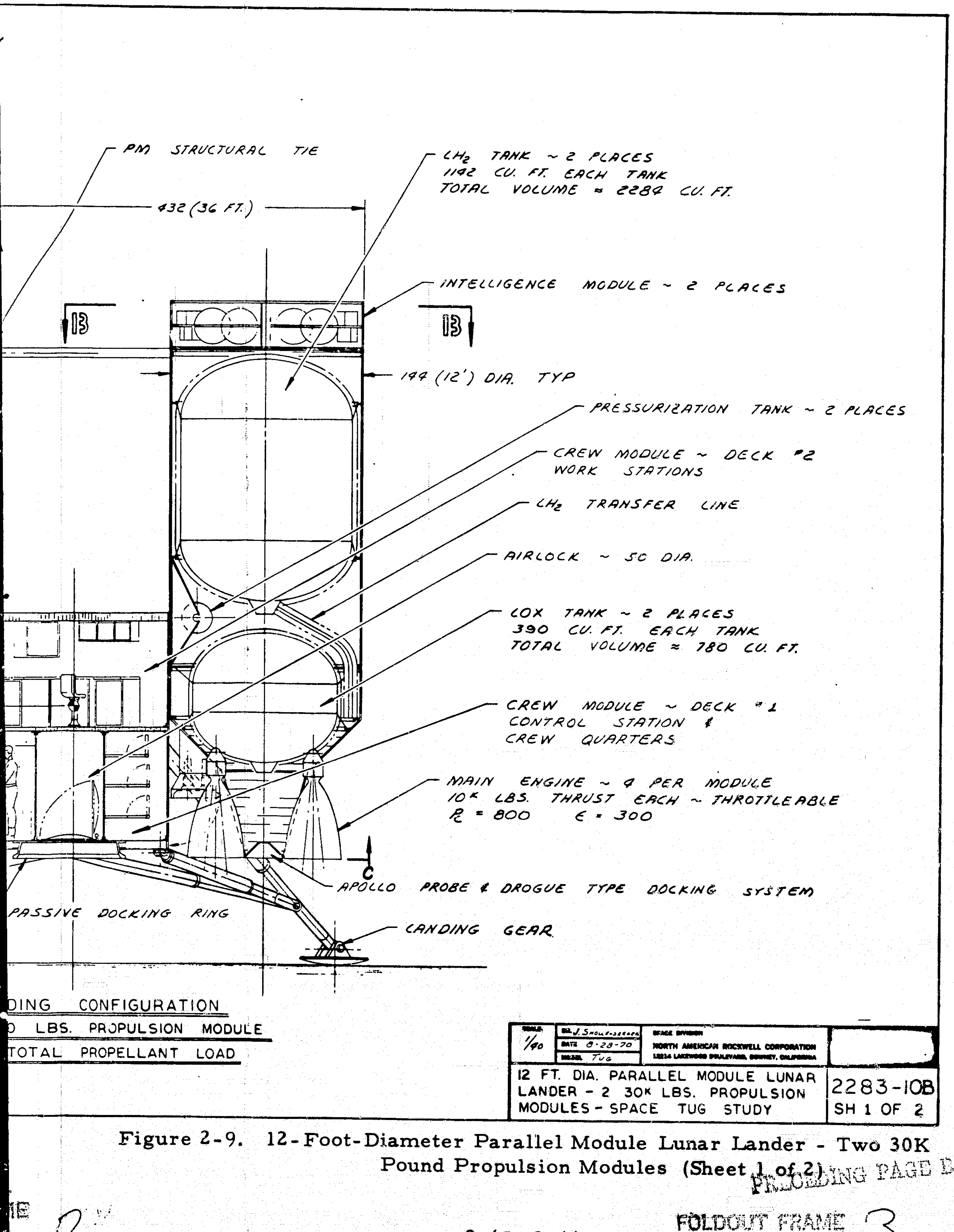

Figure 2-9. 12-Foot-Diameter Parallel Module Lunar Lander - Two 30K Pound Propulsion Modules (Sheet 1 of 2)

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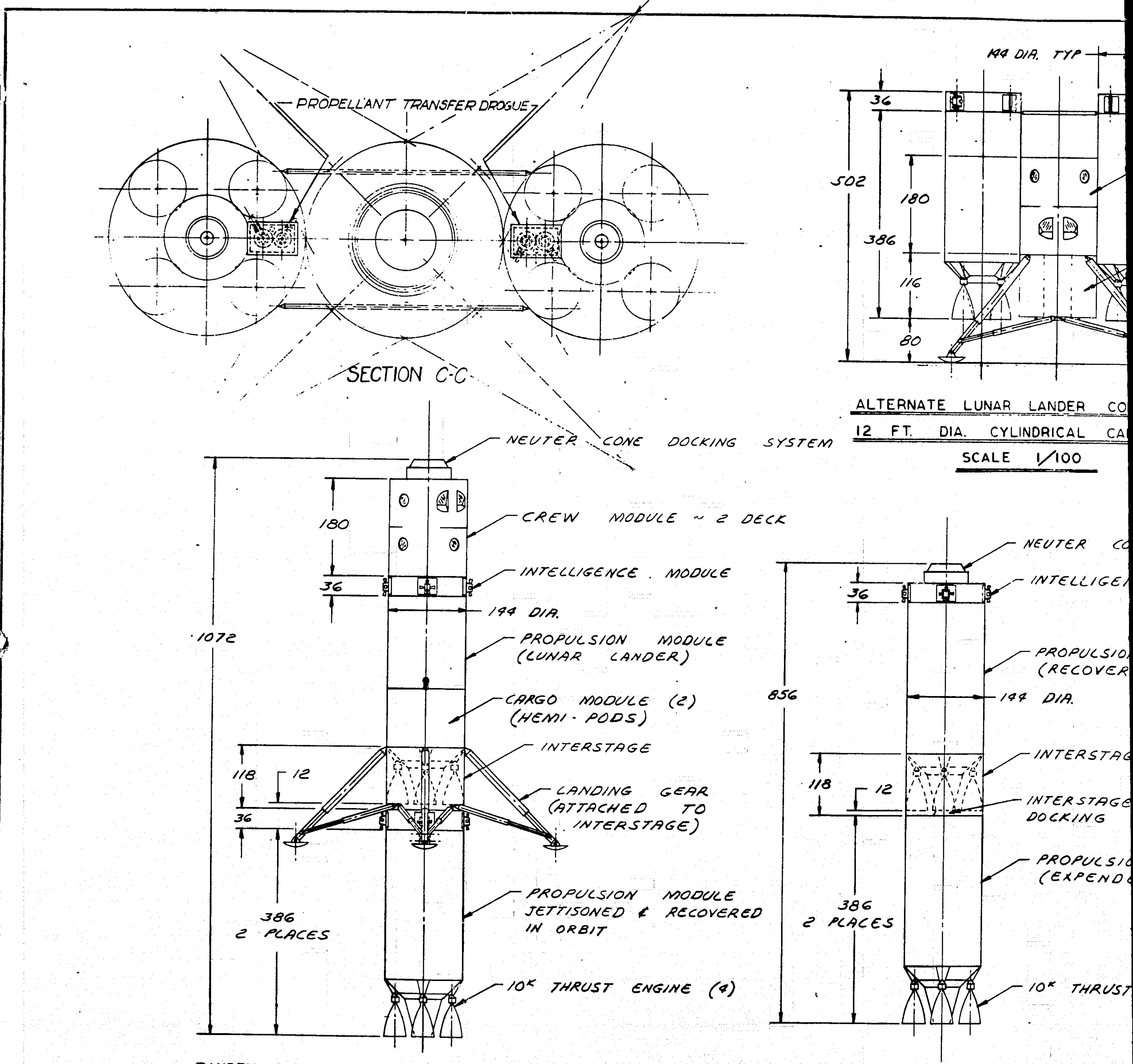

ALTERNATE LUNAR LANDER CO
12 FT. DIA. CYLINDRICAL CA
SCALE 1/100

TANDEM LUNAR LANDER CONFIGURATION
SCALE 1/100

GEOSYNCHRONOUS MISSION CONFIGURATION
SCALE 1/100

FOLDOUT FRAME 1

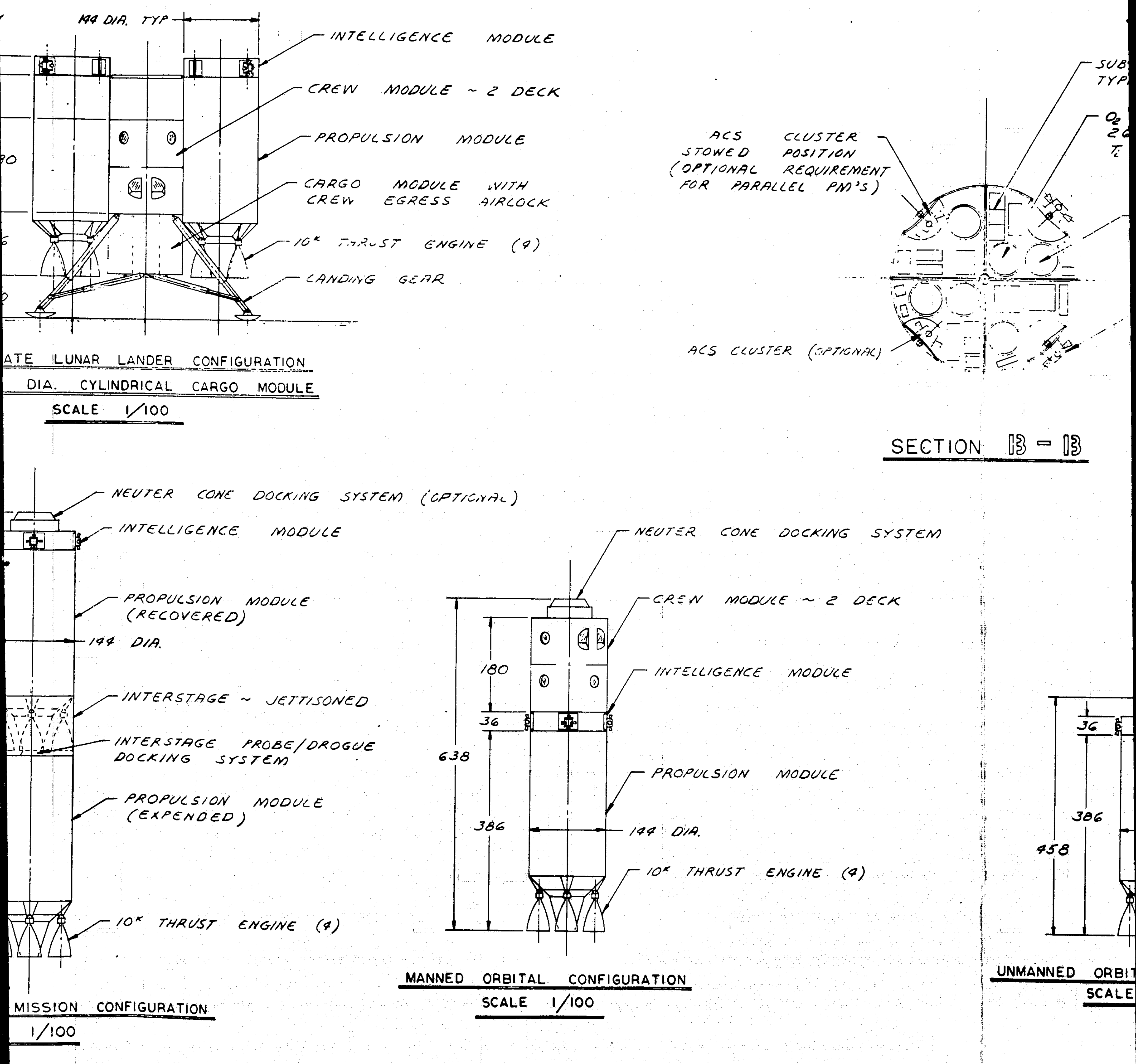

Figure 2-9. 12-Foot-D

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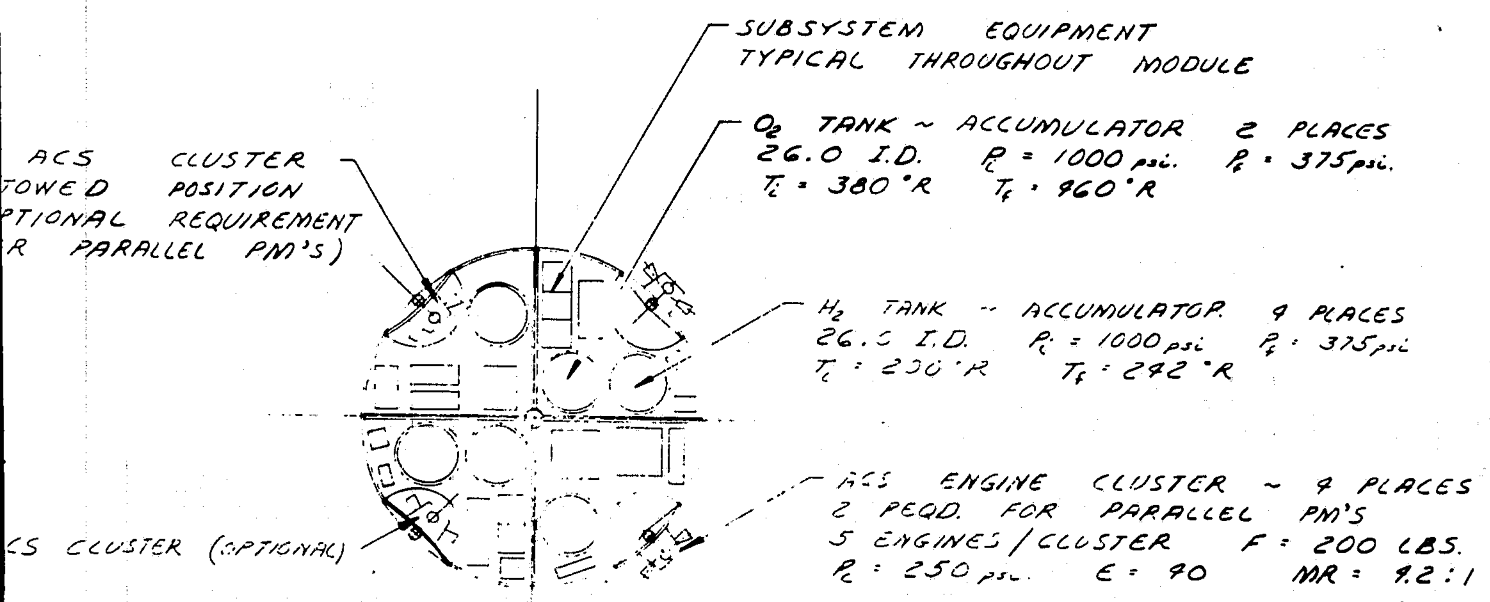
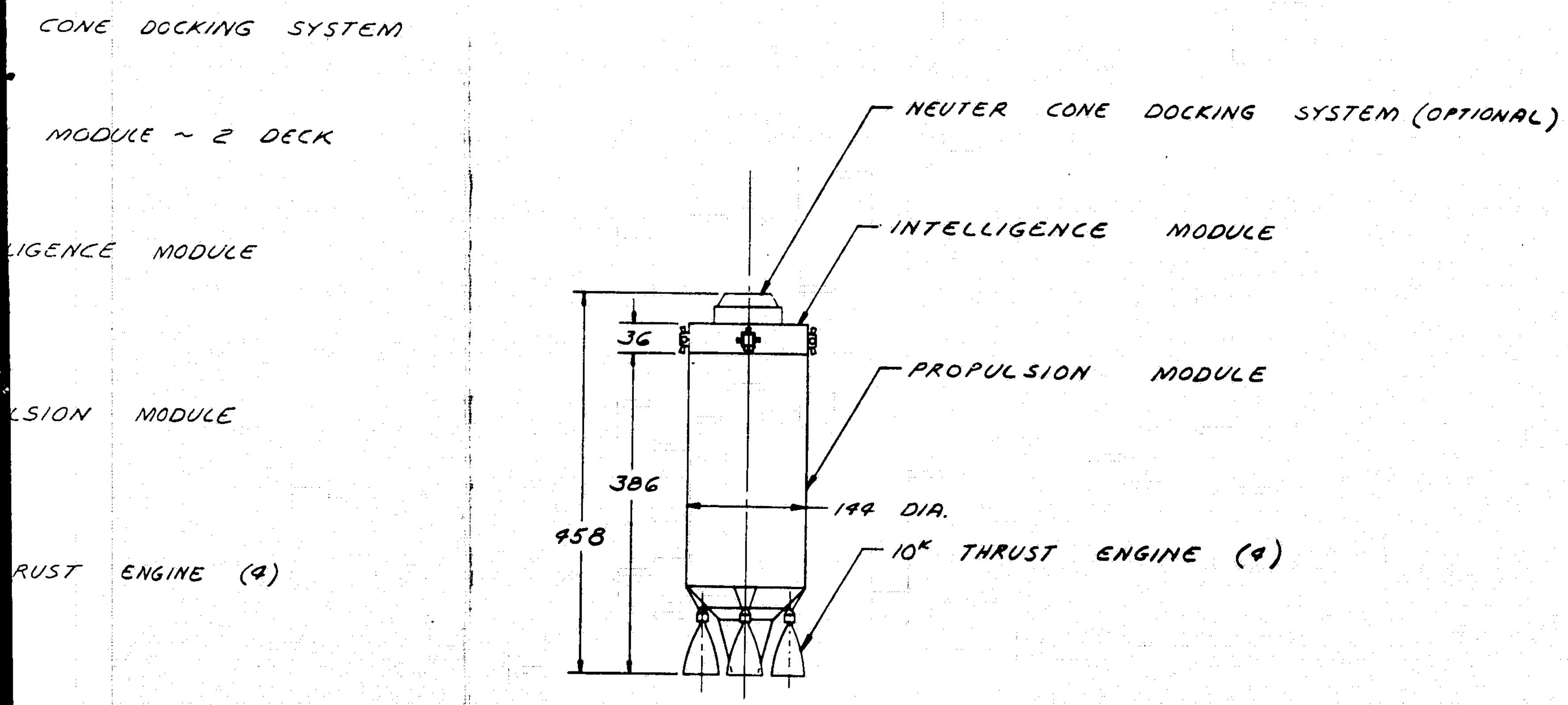


SECTION B - B



UNMANNED ORBITAL CONFIGURATION
SCALE 1/100

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SHEET 2	OF 2

Figure 2-9. 12-Foot-Diameter Parallel Module Lunar Lander - Two 30K Pound Propulsion Modules (Sheet 2 of 2)

and pivoted to a position against the lower deck of the crew module. Once the pods are in this position (rotation 90 degrees from the stowed position), the bottom bulkhead of the pod, which acts like a door, can be pivoted to the lunar surface. Then the cargo, which is strapped to the cover, can be removed easily.

The landing gear is attached to the lower deck half of the crew module. This arrangement then keeps the landing gear scar weight off of the propulsion modules. The gear may be attached to the crew module as a kit and does not influence the design of the propulsion module.

Figure 2-9 also depicts small-scale configurations of other lander arrangements. The first of these raises the two-deck crew module approximately 116 inches (2. 9 meters) to accommodate a cylindrical cargo module. The cargo module then accommodates the air lock, and additional volume is gained in the lower deck of the crew module by removal of the air lock. The landing gear is attached to the cargo module and eliminates the scar weight on both the propulsion module and the crew module. This concept makes the cargo easily accessible from the ground.

The second lander alternative is a tandem configuration in which the first stage, which consists of a propulsion module and an IM, is jettisoned during descent and recovered in orbit. The second stage is then used to complete the landing operation. The landing gear, which is on the second stage, is attached to the interstage, which joins the two vehicles. The IM is on the forward end of the propulsion module, and the crew module (two decks) is forward of it. A neuter cone docking system is affixed to the forward end of the crew module. The lander vehicle is 650 inches (16. 5 meters) long. Operationally, this vehicle's landing mode is undesirable and it is difficult to stage.

A manned and unmanned orbital operation's configuration are also shown. The unmanned concept is 458 inches (11. 6 meters) long with a neuter cone docking system attached to the forward end of the IM. If the docking system is removed, the configuration is 422 inches (10. 7 meters) long. The manned orbital vehicle accommodates a two-deck crew module forward of the intelligence module. This concept is 638 inches (16. 2 meters) long with a neuter cone docking system attached to the forward end of the crew module.

The last configuration is a tandem vehicle geosynchronous mission concept. The first stage of this vehicle does not have an IM, and the propulsion module is expended to perform the mission. The second stage has an IM at the forward end and is recovered. A 118-inch (3. 0-meters) interstage joins the two stages and is expended with the first stage.

Overall, the concepts shown on this drawing offer a number of desirable features for the orbital and lunar landing missions. The only drawback to these concepts is the expending of one propulsion module to accomplish the geosynchronous mission. The orbital missions offer a small propellant load tug vehicle with a forward located crew module. The landers (parallel propulsion module) offer low center of gravities, good crew access to the ground, and ease of cargo removal and eliminate landing gear scar weight from the propulsion modules. The lower-center-of-gravity landers also exhibit shorter gear spread radius, which accounts for the lighter weight landing gear. In the lander mode, two intelligence modules have been used but they may be only partially loaded with equipment. The ACS clusters nearest the crew module on the IM's have been eliminated for the lander version; however, they could be carried for additional redundancy or backup. The four main engines on each propulsion module in the parallel arrangement may not also be used. The inboard engines may be eliminated or carried as backup.

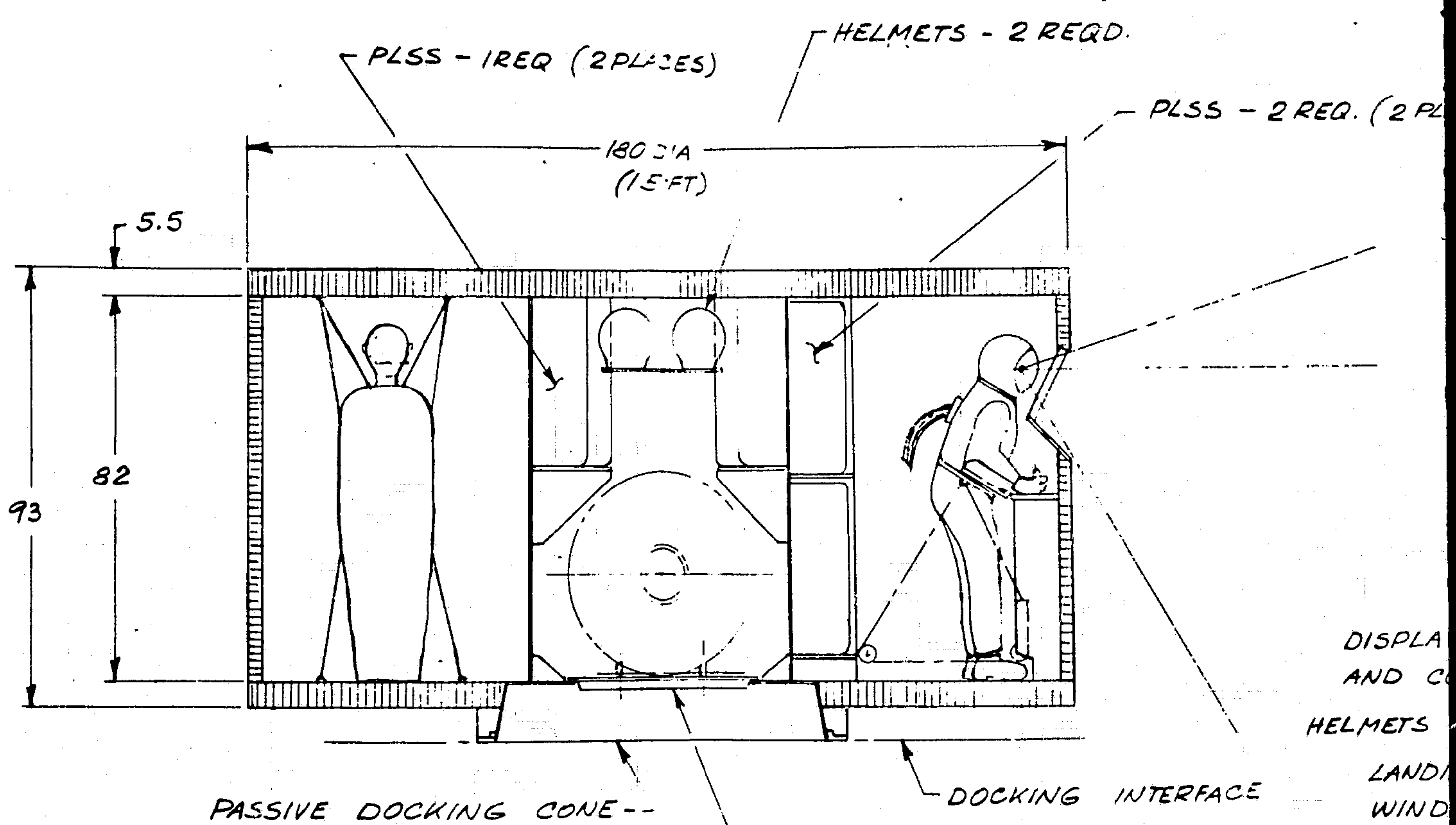
2.2.4 12-, 15-, AND 22-FOOT-DIAMETER (3.66-, 4.57-, AND 6.7-METER) CREW MODULE STUDY

12- and 15-Foot (3.66- and 4.57-Meter) Modules

The first drawing of this study (Figure 2-10) investigated 12-foot- and 15-foot-diameter (3.66 meters and 4.57-meters) diameter crew modules. The spectrum of configurations included a 15-foot-diameter (4.57-meters) vertical cylinder for six men seven days; 12-foot-diameter (3.66 meters) vertical cylinder with two decks for six men seven days; and a 12-foot-diameter (3.66-meters) horizontal cylinder for six men seven days. Each of the CM's has been arranged so that they can be used by 12 men in a crew rescue mode with no change to the basic floor plan except for the folding of bunks to obtain standing area.

All of the configurations featured the same type of construction—aluminum honeycomb pressure vessel with a centrally located air lock. Each module also incorporated an emergency egress hatch in the side wall of 36 inches (0.91 meters) in diameter. The control station is a stand-up station similar to that on the Apollo lunar module. Work stations consist of chairs with tables. Above the work station benches are storage cabinets for food preparation and environmental control equipment and scientific equipment. Fold-up bunks are provided on the side wall. These bunks, when folded up, provide the needed floor area for crew rescue operations.

The air lock in the 15-foot-diameter (4.54-m) combination is 60 inches (1.5 meters) in diameter with a pressurized volume of 134 cubic feet (3.79 cubic meters). In each of the 12-foot (3.66-meters) modules, the air lock's inside diameter is 50 inches (1.27 meters). The pressurized



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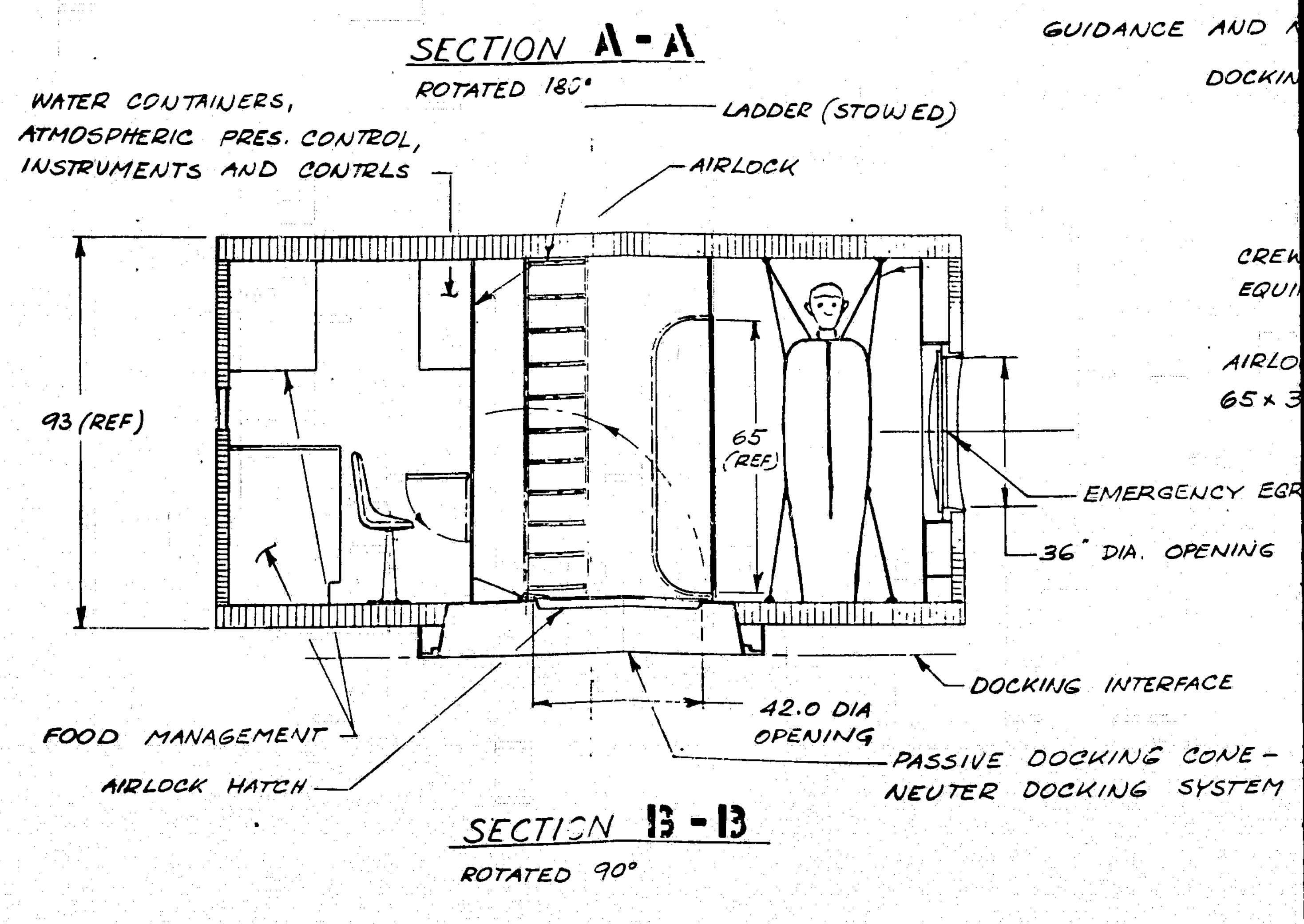

SECTION B-B

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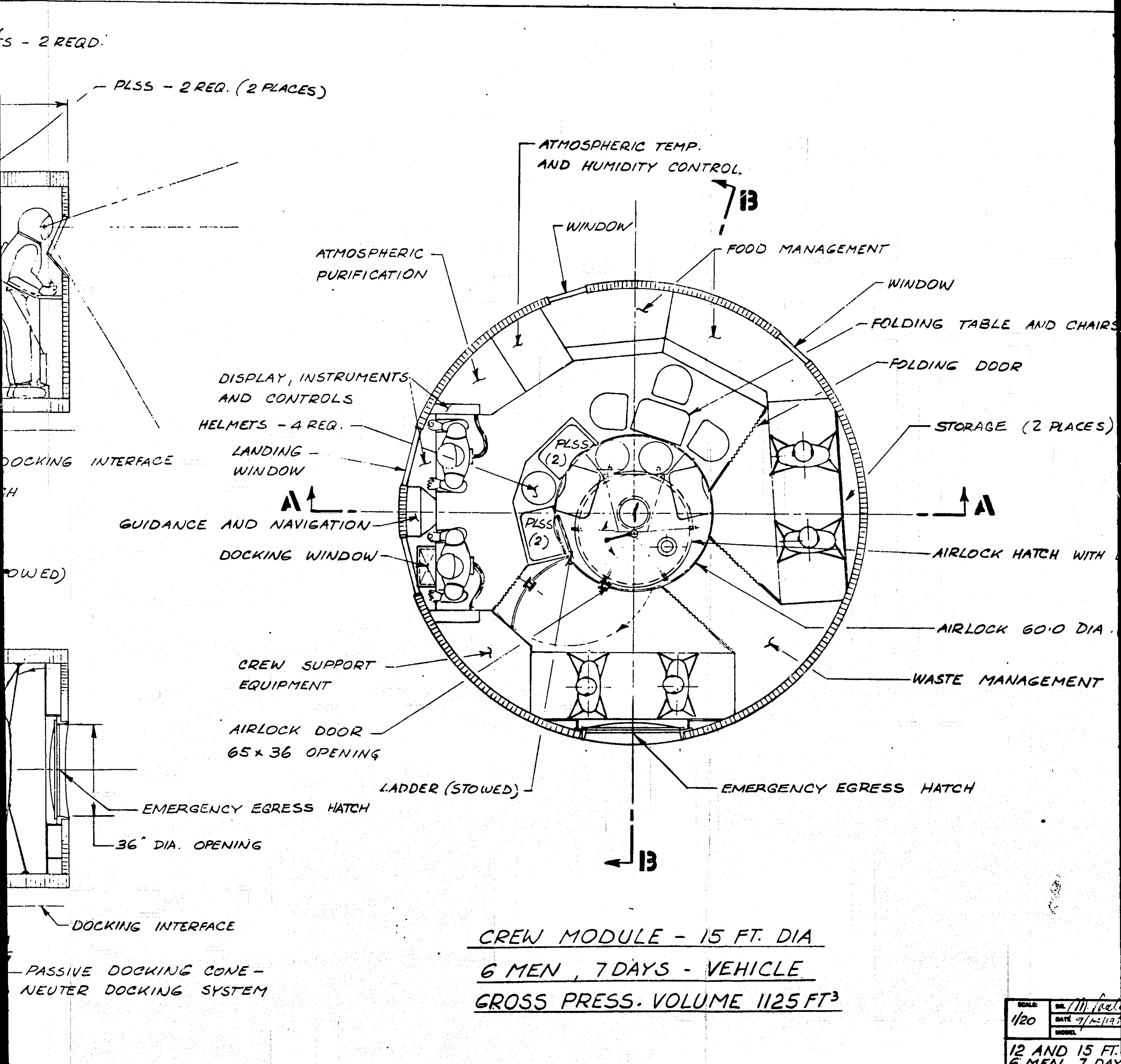


Figure 2-10. 12- and 15-Foot-Diameter Crew
12 Man Rescue Vehicl

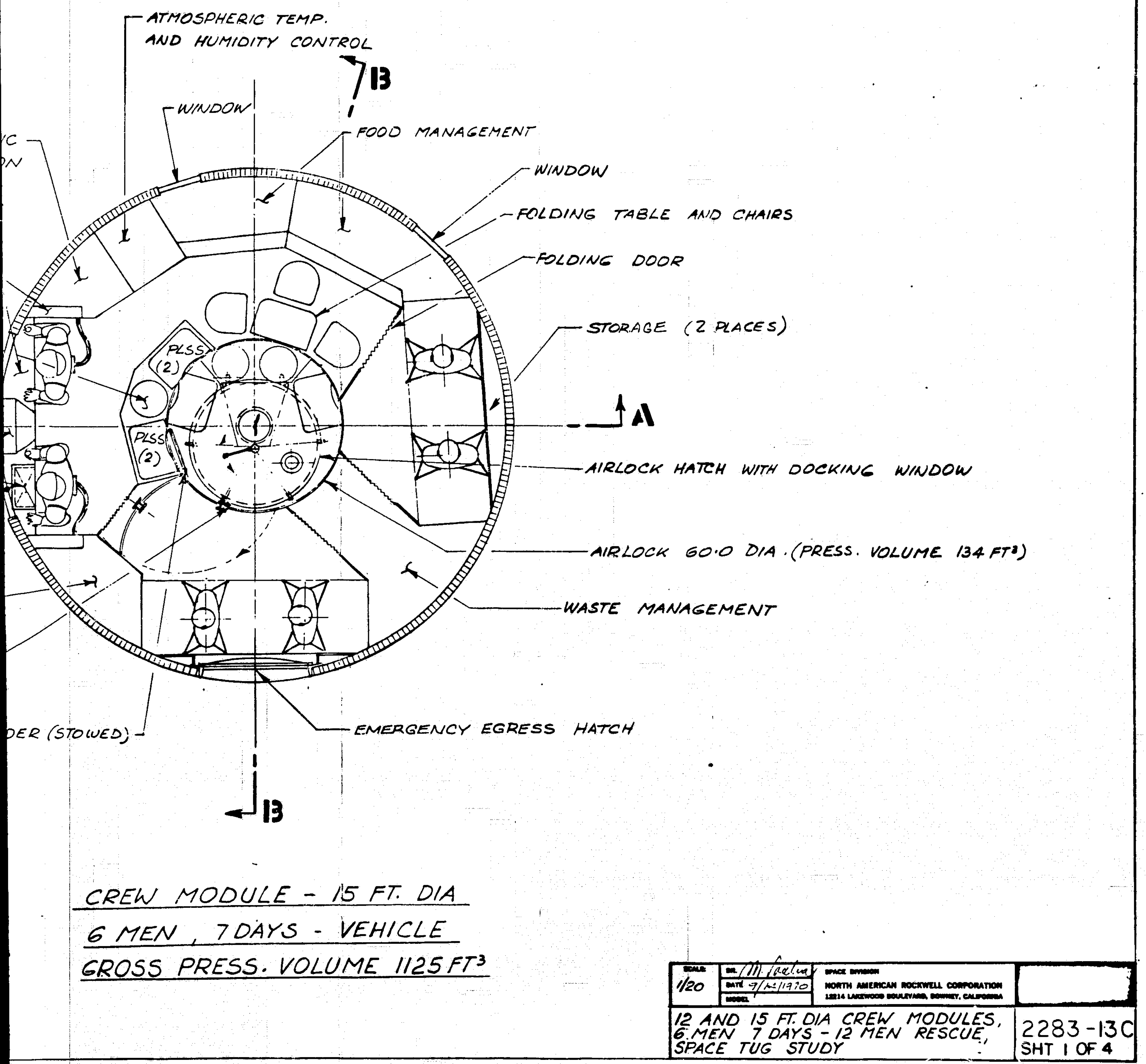
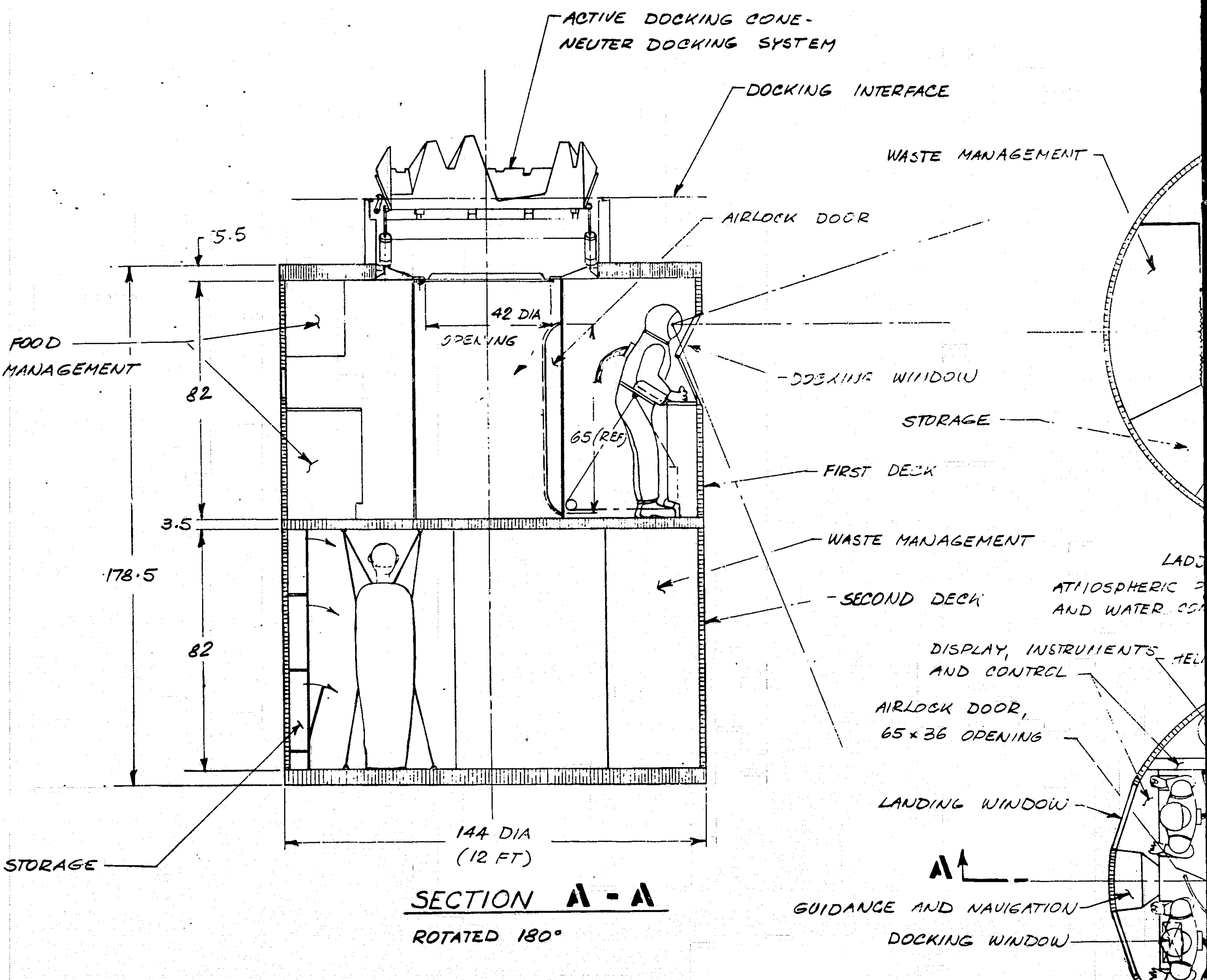


Figure 2-10. 12- and 15-Foot-Diameter Crew Modules, 6 Man 7 Days - 12 Man Rescue Vehicle (Sheet 1 of 5)



CREW MODLE - 12 FT. DIA.
2 DECKS - 6 MEN, 7 DAYS -
VEHICLE
GROSS PRESS. VOL. 1460 FT³

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FOLDOUT FRAM

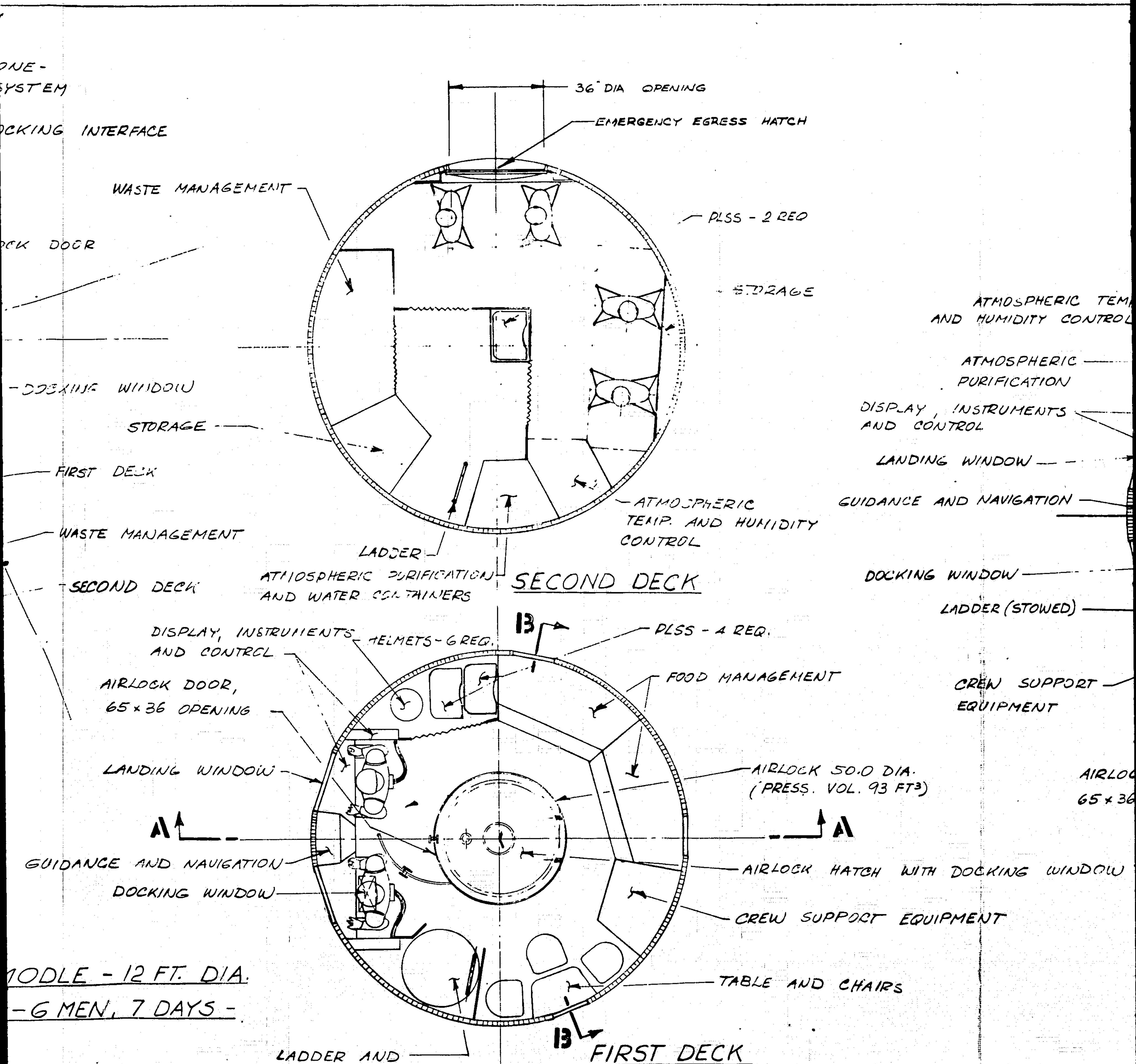

Figure 2-10. 12- and

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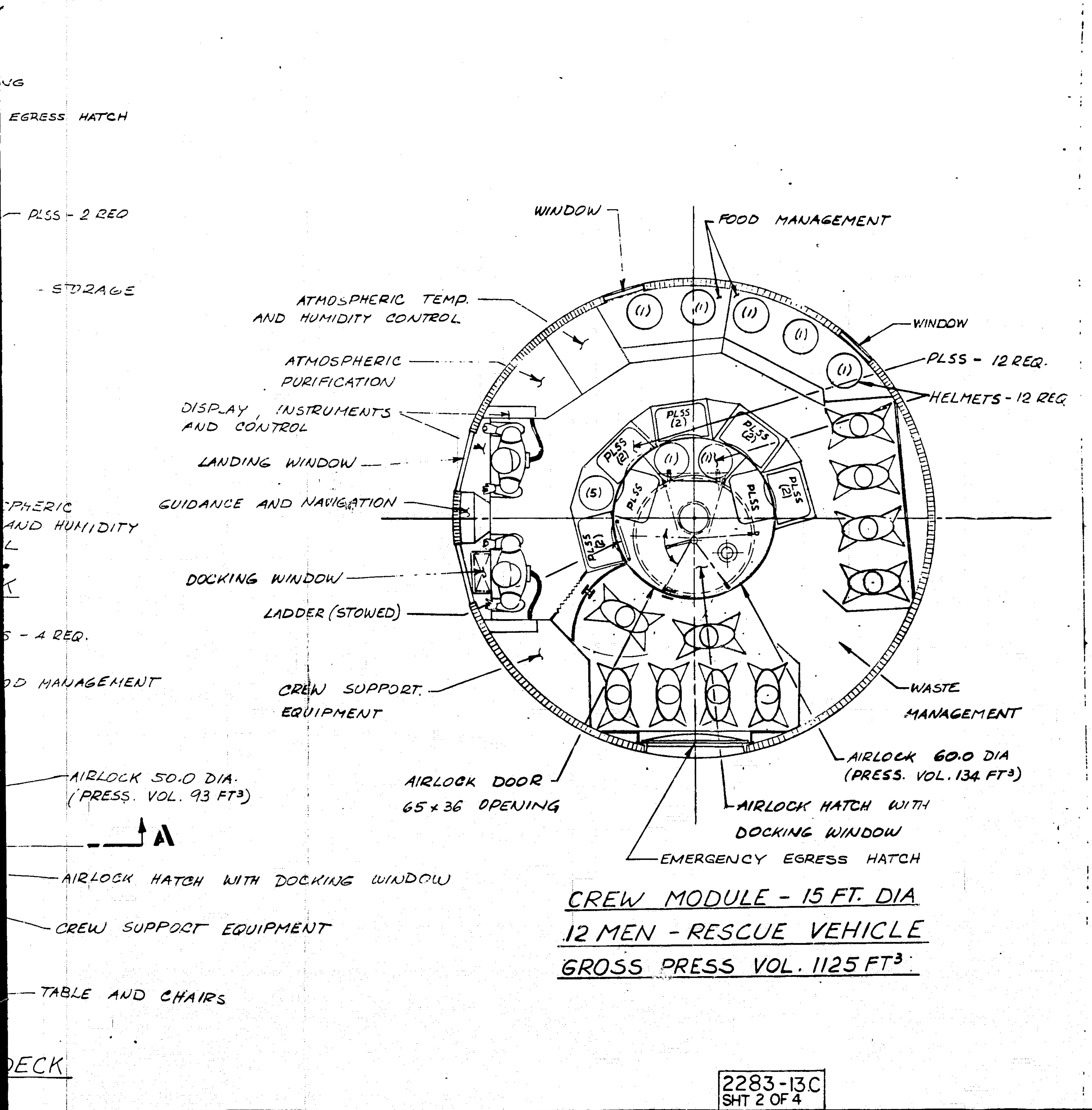

Figure 2-10. 12- and 15-Foot-Diameter Crew Modules, 6 Man 7 Days - 12 Man Rescue Vehicle (Sheet 2 of 5)

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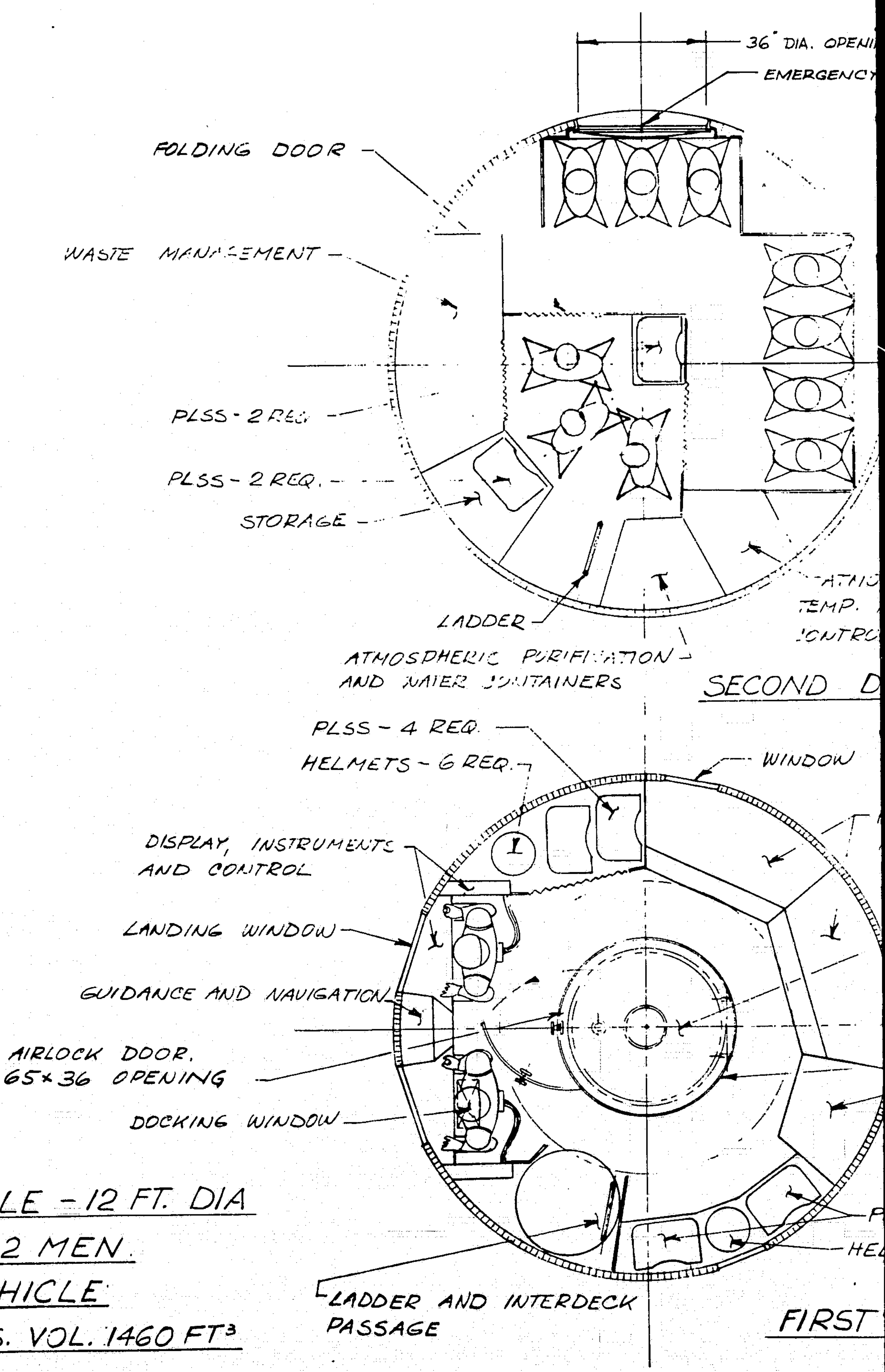

CREW MODULE - 12 FT. DIA
2 DECKS - 12 MEN.
RESCUE VEHICLE
GROSS PRESS. VOL. 1460 FT³

FOLDOUT FRAME 1

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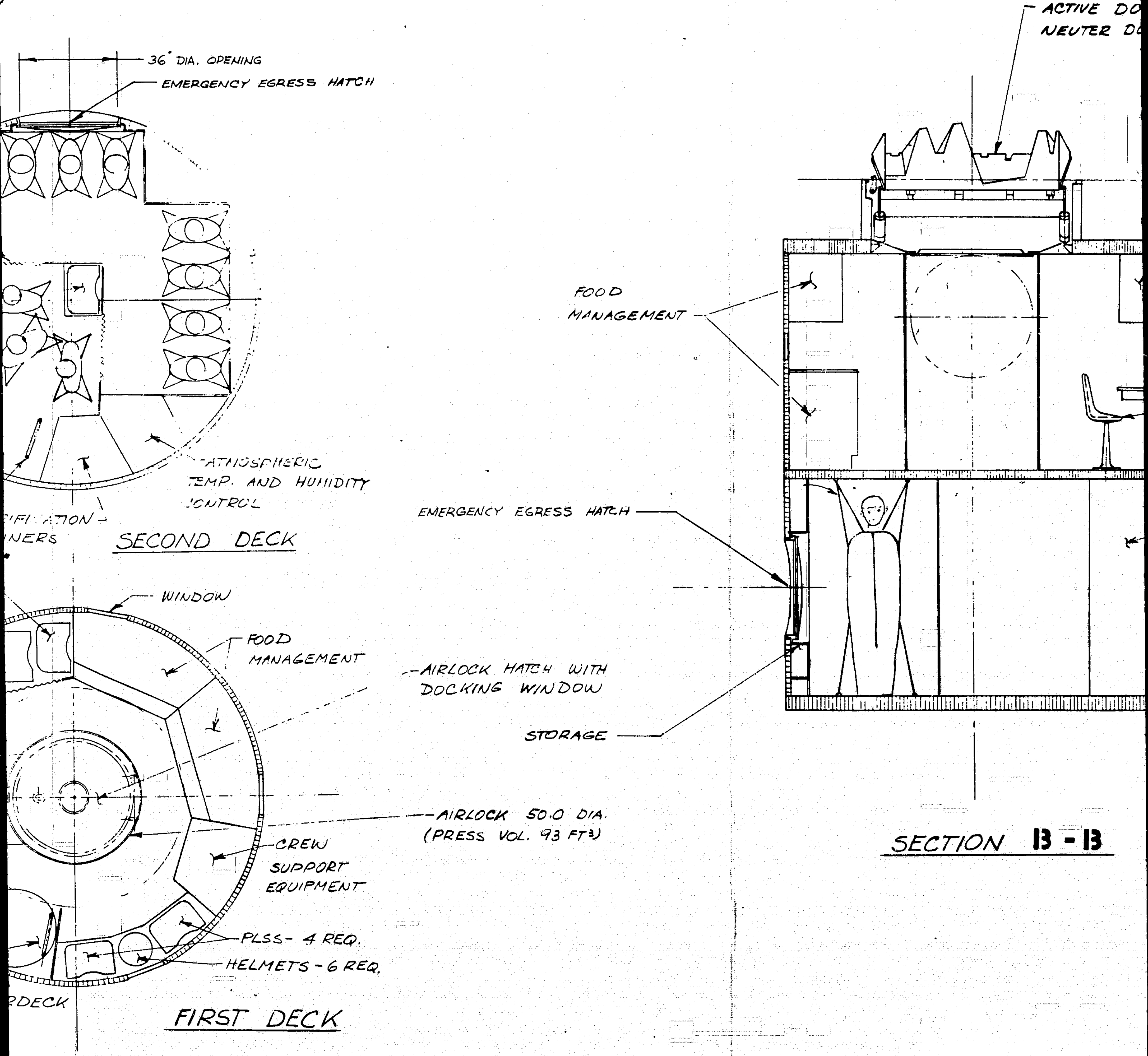

Figure 2-10. 12- and 15-Foot-Diameter Crew M
12 Man Rescue Vehicle

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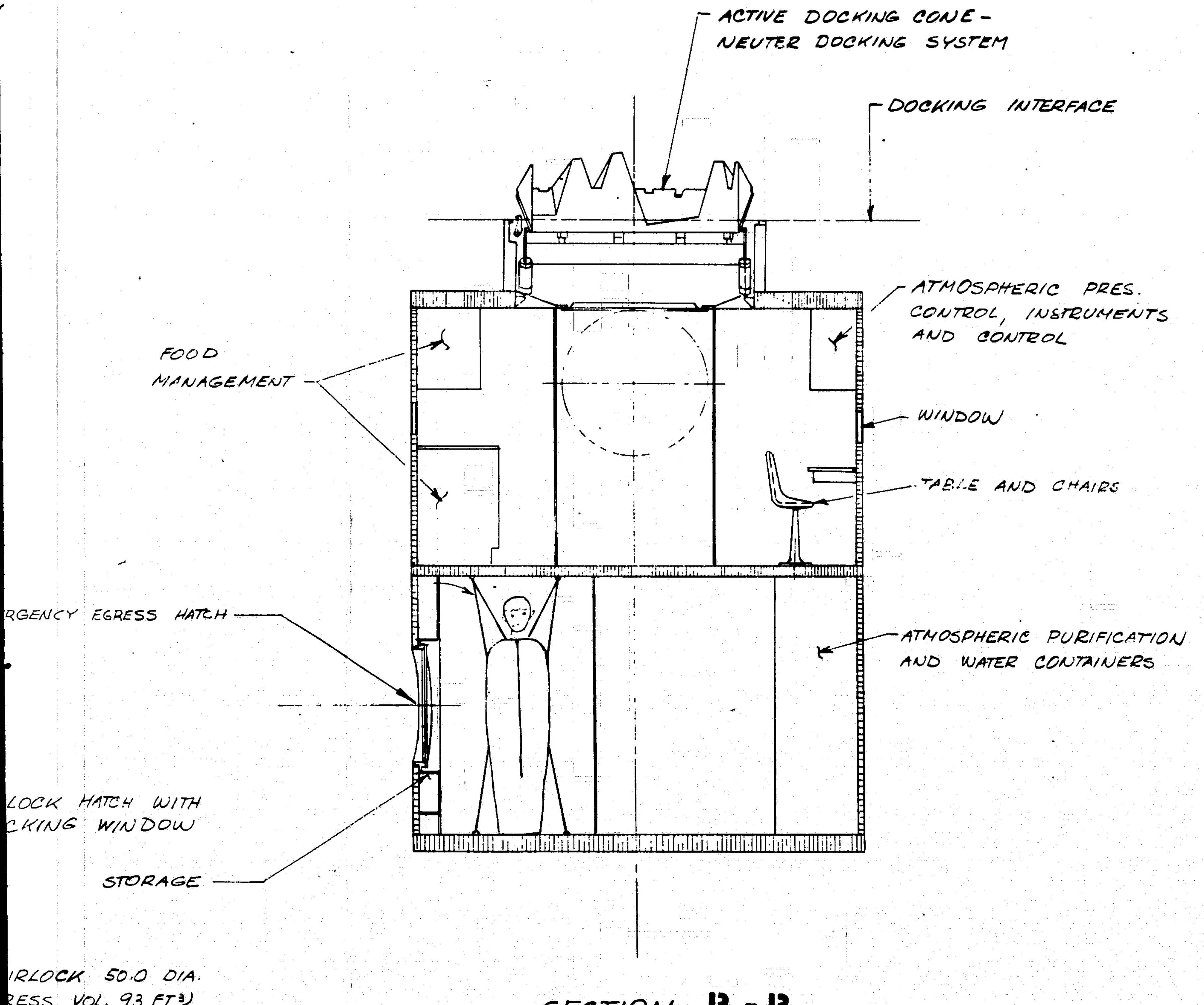

SECTION B - B

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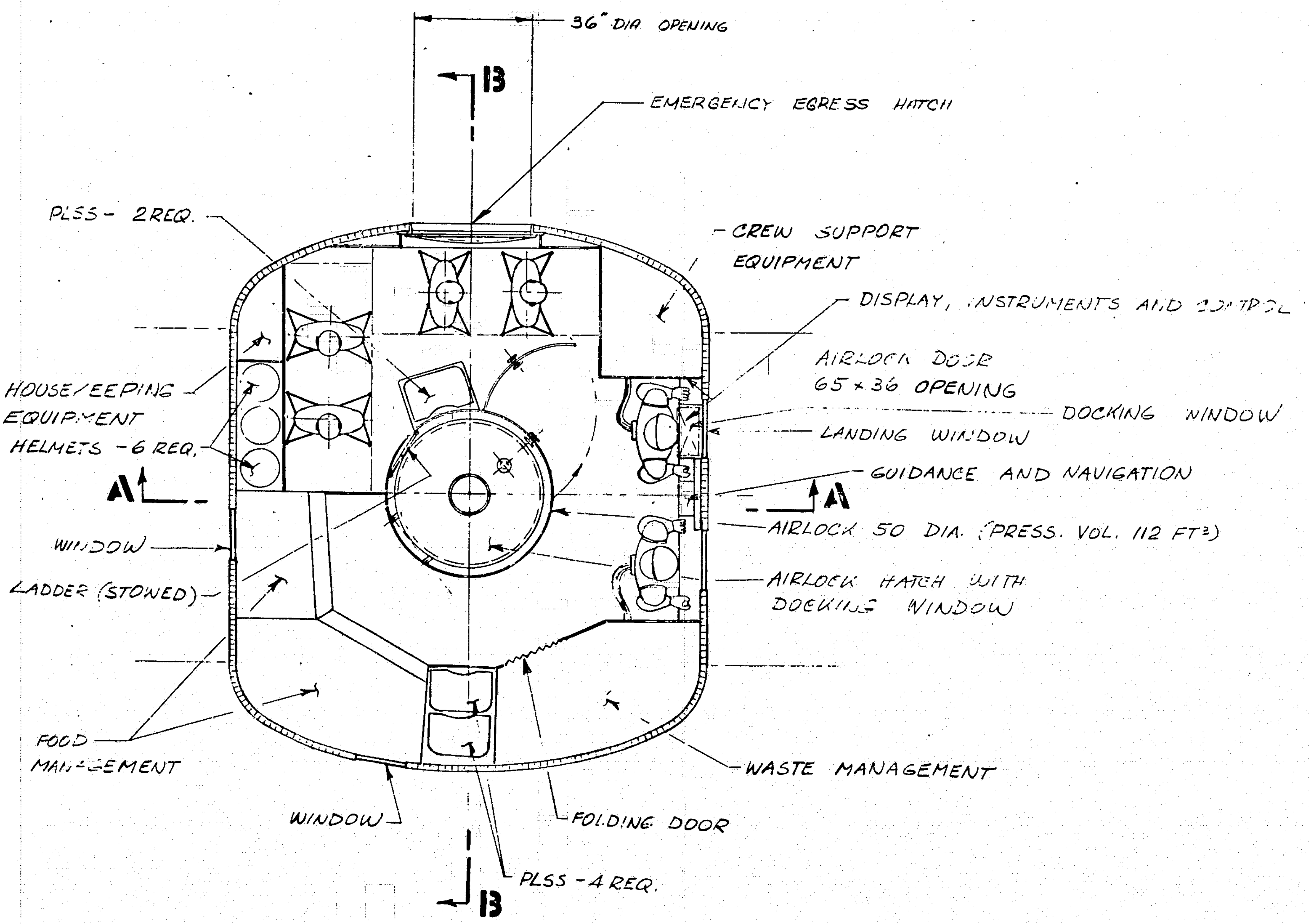
SHT 3 OF 4

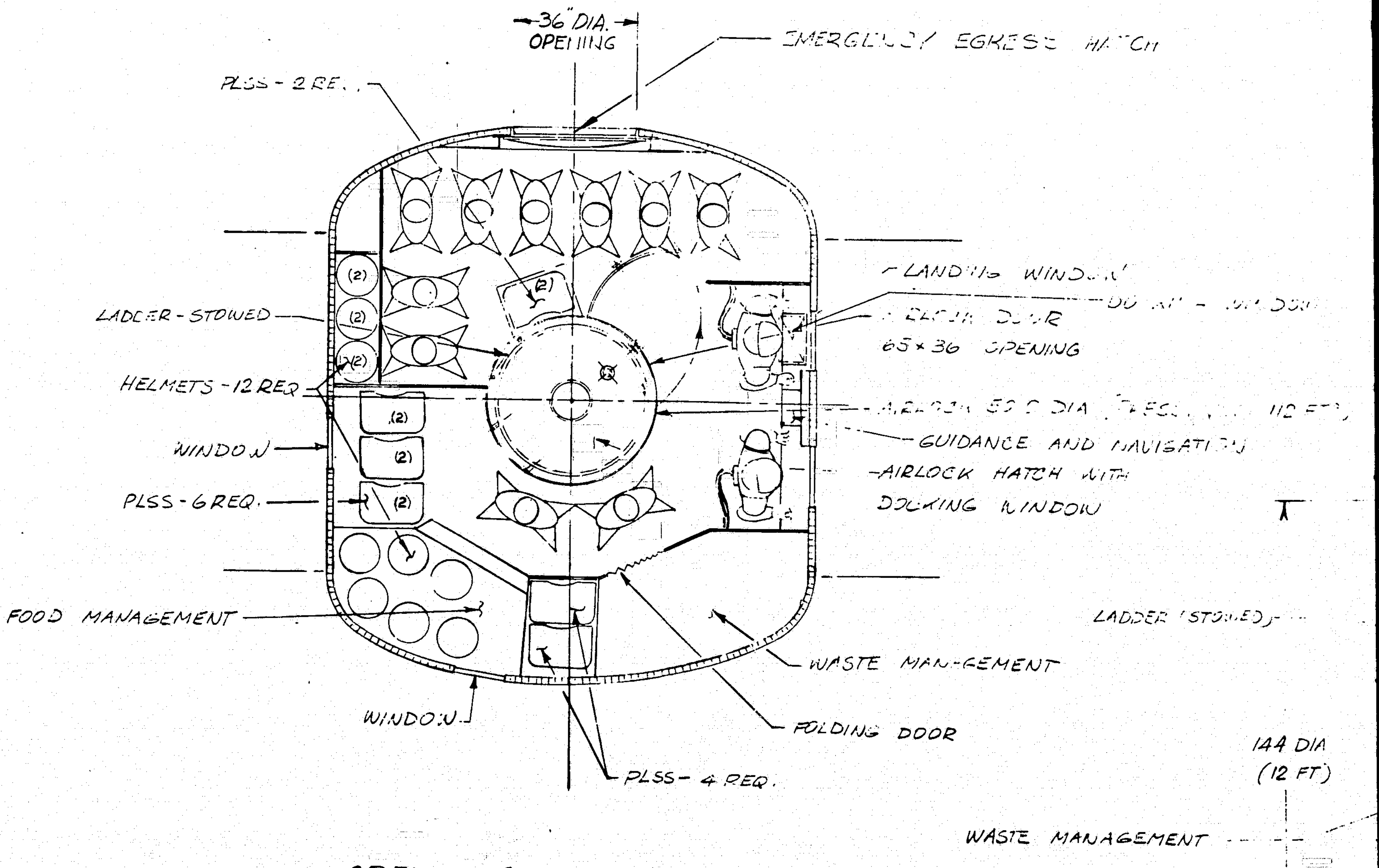
Figure 2-10. 12- and 15-Foot-Diameter Crew Modules, 6 Man 7 Days - 12 Man Rescue Vehicle (Sheet 3 of 5)



CREW MODULE - 12 FT. DIA
6 MEN, 7 DAYS - VEHICLE
GROSS PRESS. VOL. 1055 FT^3

2283-13C
SHT 4 OF 4

Figure 2-10. 12- and 15-Foot-Diameter Crew Modules, 6 Man 7 Days - 12 Man Rescue Vehicle (Sheet 4 of 5)



CREW MODULE - 12 FT. DIA
12 MEN - RESCUE VEHICLE
GROSS PRESS VOL. 1055 FT³

SECTION B - B
ROTATED 90°

PASSIVE DOCKING
NEUTER DOCKING

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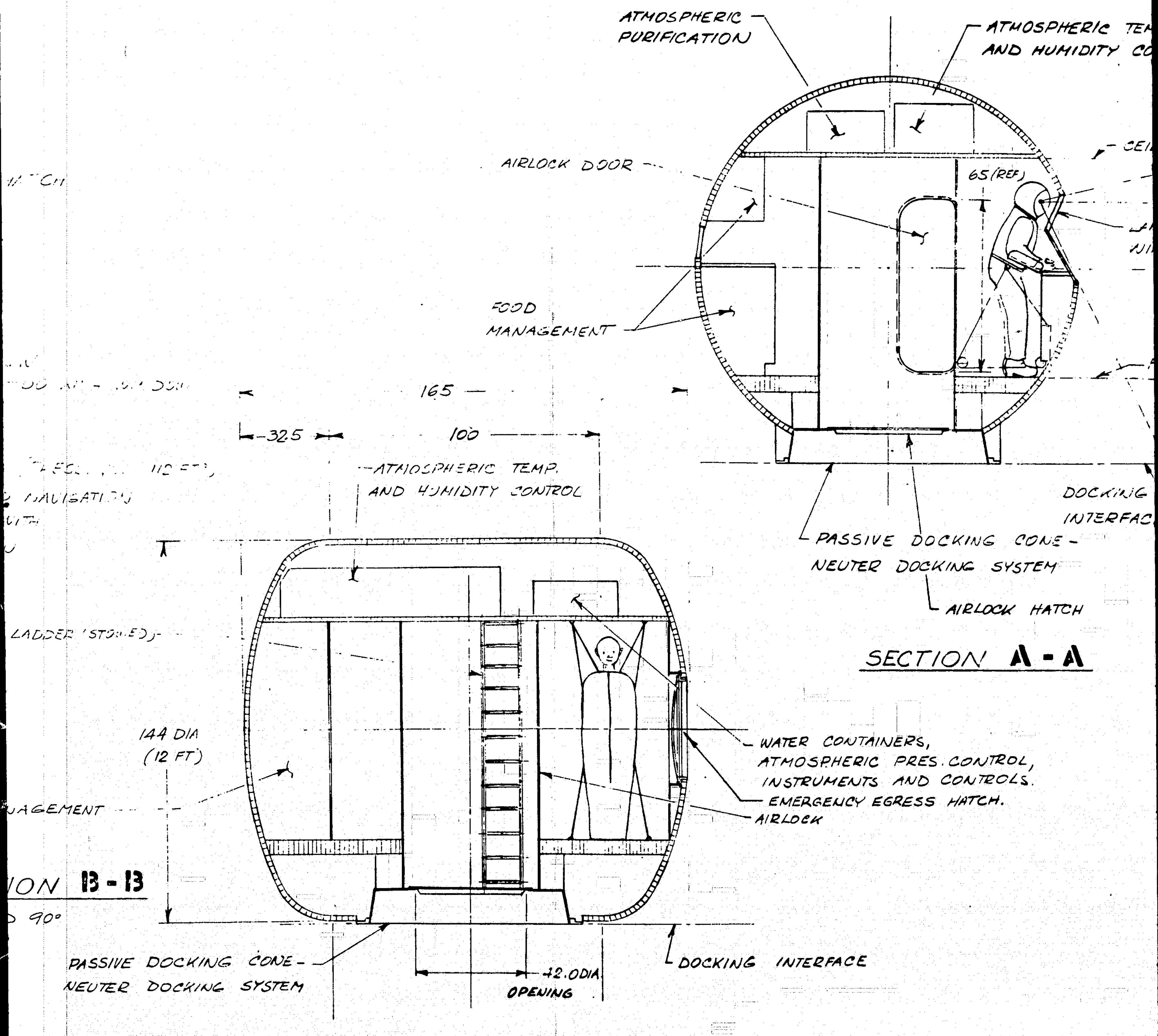


Figure 2-10. 12- and 15-Foot-Diameter Crew Modules, 6 Man 7 D
12 Man Rescue Vehicle (Sheet 5 of 5)

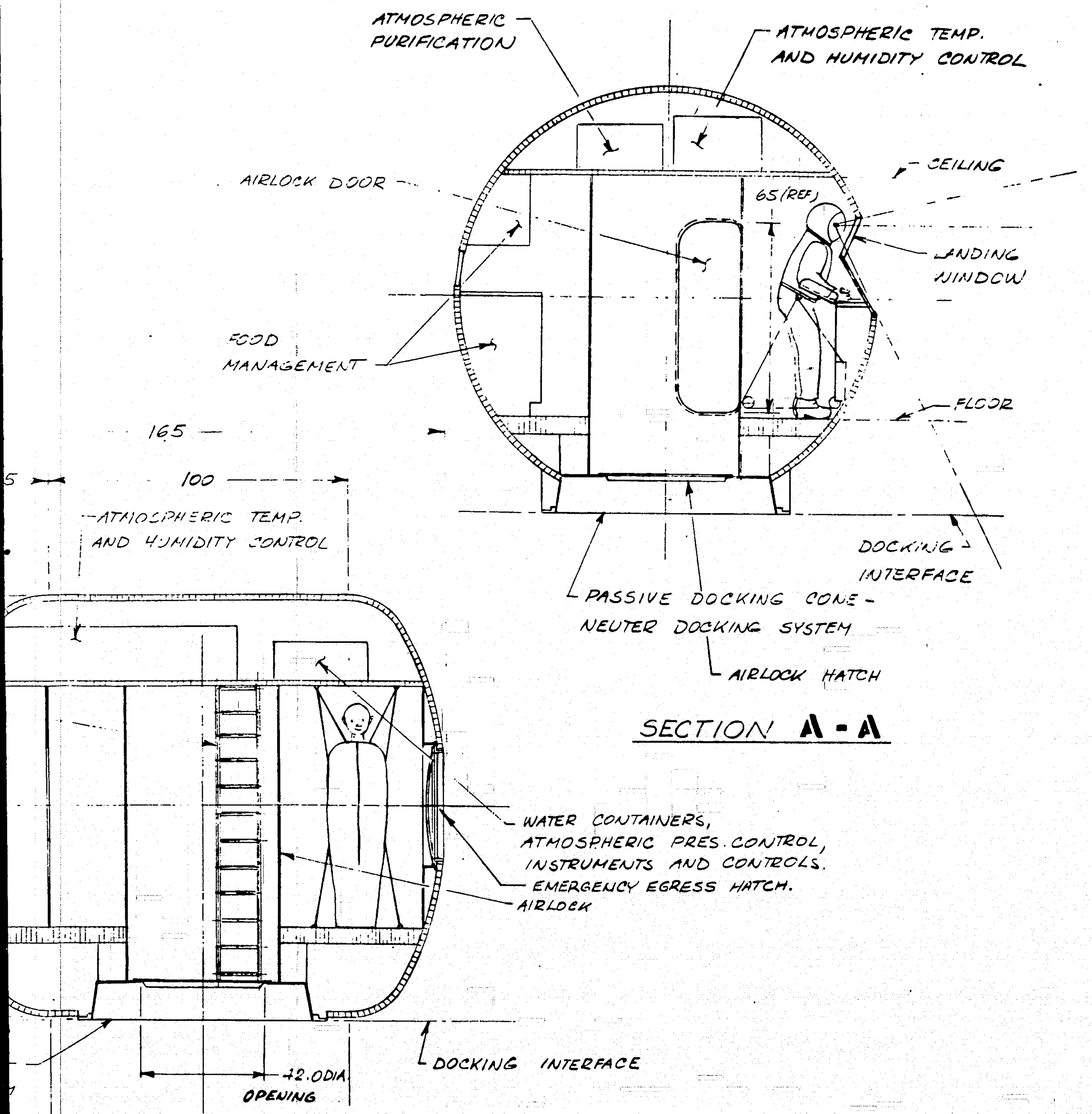


Figure 2-10. 12- and 15-Foot-Diameter Crew Modules, 6 Man 7 Days - 12 Man Rescue Vehicle (Sheet 5 of 5)

volume in the vertical module is 93 cubic feet (2.63 cubic meters) and 112 cubic feet (3.17 cubic meters) in the horizontal module. The pressurized volume of the 15-foot (4.57-m) module is 1125 cubic feet (31.85 cubic meters). For the 12-foot (3.66-m) vertical module it is 1460 cubic feet (41.34 cubic inches), and for the 12-foot (3.66-m) horizontal module it is 1055 cubic feet (29.87 cubic meters). The most volume is in the 12-foot (3.66-m) vertical module; however, this module contains two desks and offers little aisle area for movement from control station to work stations. The horizontal module requires that floors be put in the cylinder, which lessens the available storage area and gives one the feeling of closeness when viewed from the cylinder end. This module also uses elliptical bulkheads on the end of cylinder to offer a maximum volume, while still remaining within the 15-foot-diameter (4.57-m) limitation.

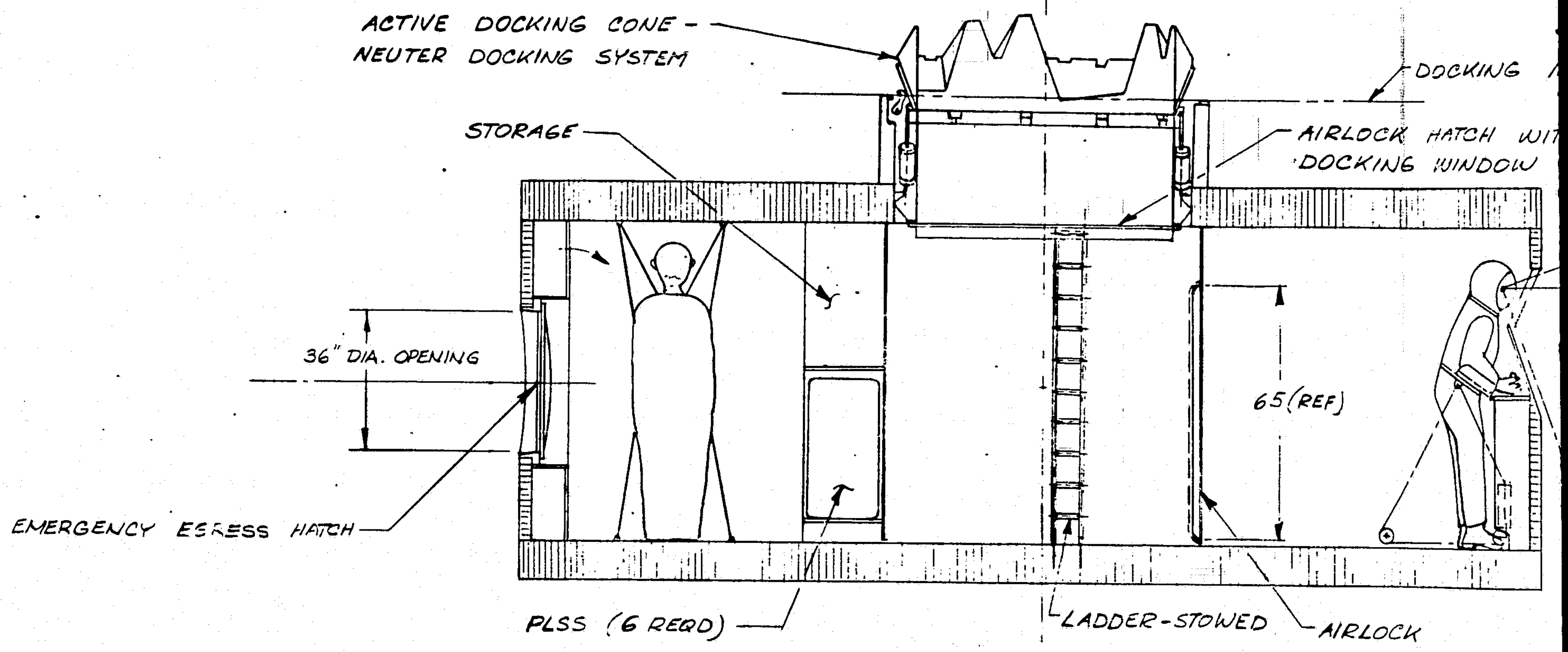
Of the three modules configured, the 15-foot (4.57-m) modules appears to be optimum. The 12-foot (3.66-meters) vertical required two decks, and the horizontal required floors, as well as overhead and beneath ceiling and floor storage. It also offered less floor area for standing and working. Before the 15-foot-diameter (4.57-m) module was adopted, though, further investigations were performed of other configurations, 22-foot-diameter (6.7-m) and arrangements for other longer-duration missions of four men for 28 days on the lunar surface.

22-Foot (6.7-Meter) Modules

The 22-foot-diameter (6.7-m) module is shown in Figure 2-11. This module is a vertical cylinder pressure vessel with flat bulkheads. The central air lock is 84 inches (2.13 meters) inside diameter with a pressurized volume of 263 cubic feet (7.45 cubic meters). The module internal pressurized volume is 2470 cubic feet (69.94 cubic meters) and is much more than is required for even the longest duration missions. As evidenced by the floor area plan, there is an over abundance of area for all functions within this module. If this area could be shared effectively with some payload, then the concept would be economically feasible. However, as only a crew shelter and work station, this concept is overdesigned. For this reason, as well as the 22-foot-diameter's (6.7 meters) incompatibility with the 15-foot-diameter (4.57-m) EOS cargo bay, this concept was carried no further.

Lunar Shelter Version

The last configuration prepared is shown in Figure 2-12. This concept is a 15-foot-vertical (4.57-m) vertical cylinder configured for four men 28 days. The arrangement is similar to that shown for this diameter and demonstrates that there is little difference between the configurations for four men for 28 days and six men for seven days. The most noticeable



SECTION A-A
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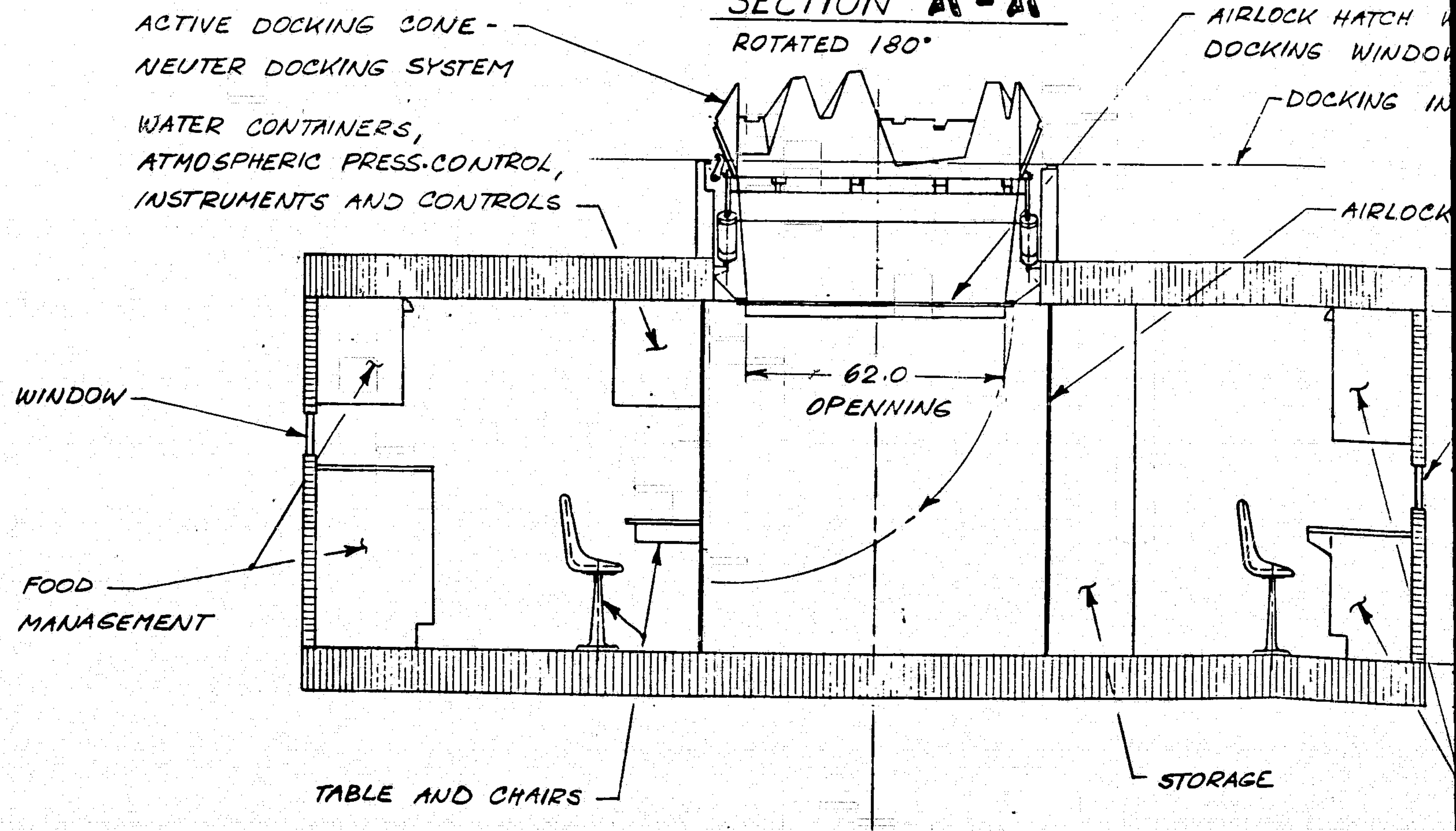


SECTION C-C
ROTATED 90°

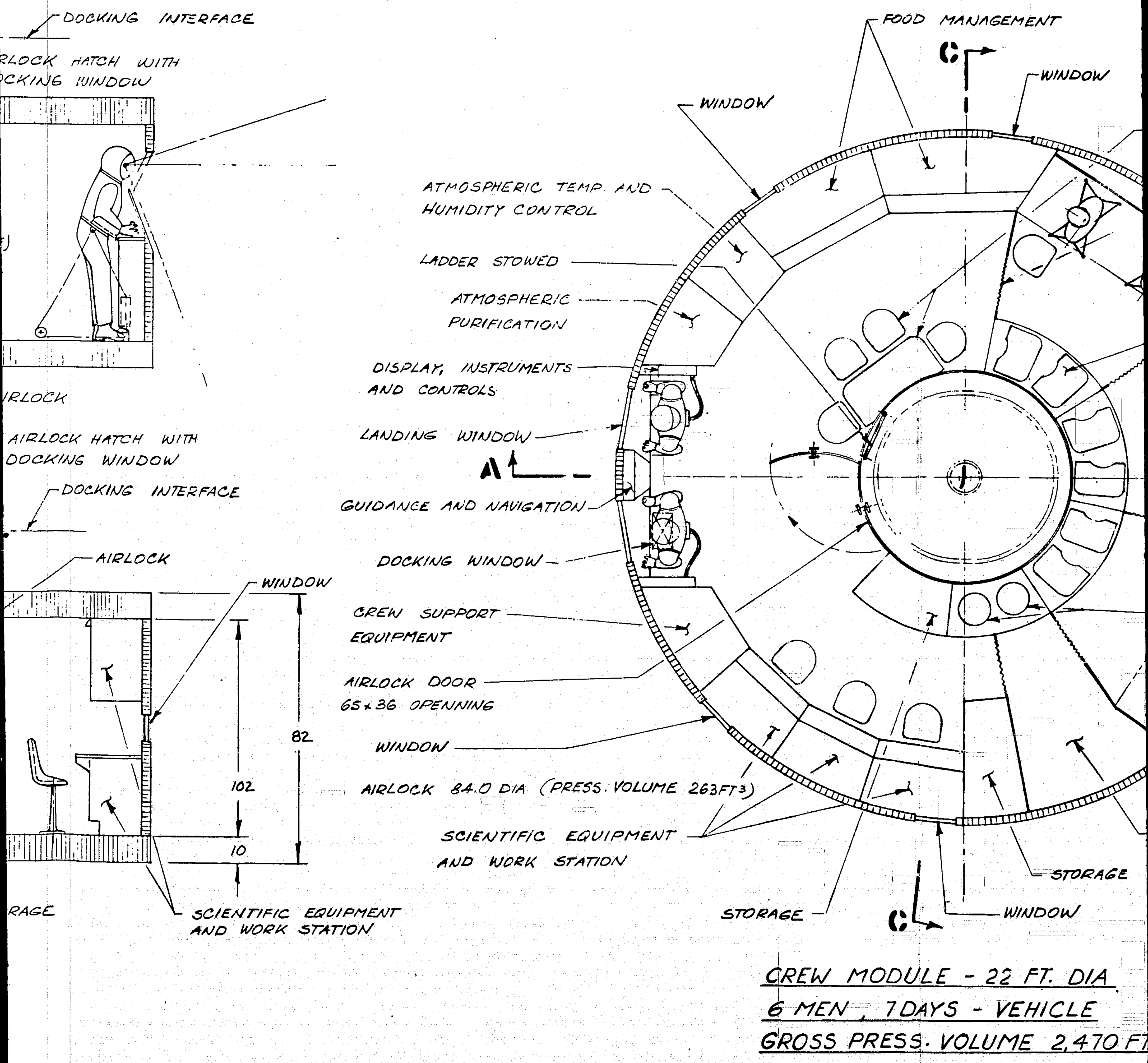


Figure 2-11. 22-Foot-Diam

FOLDOUT FRAME 2

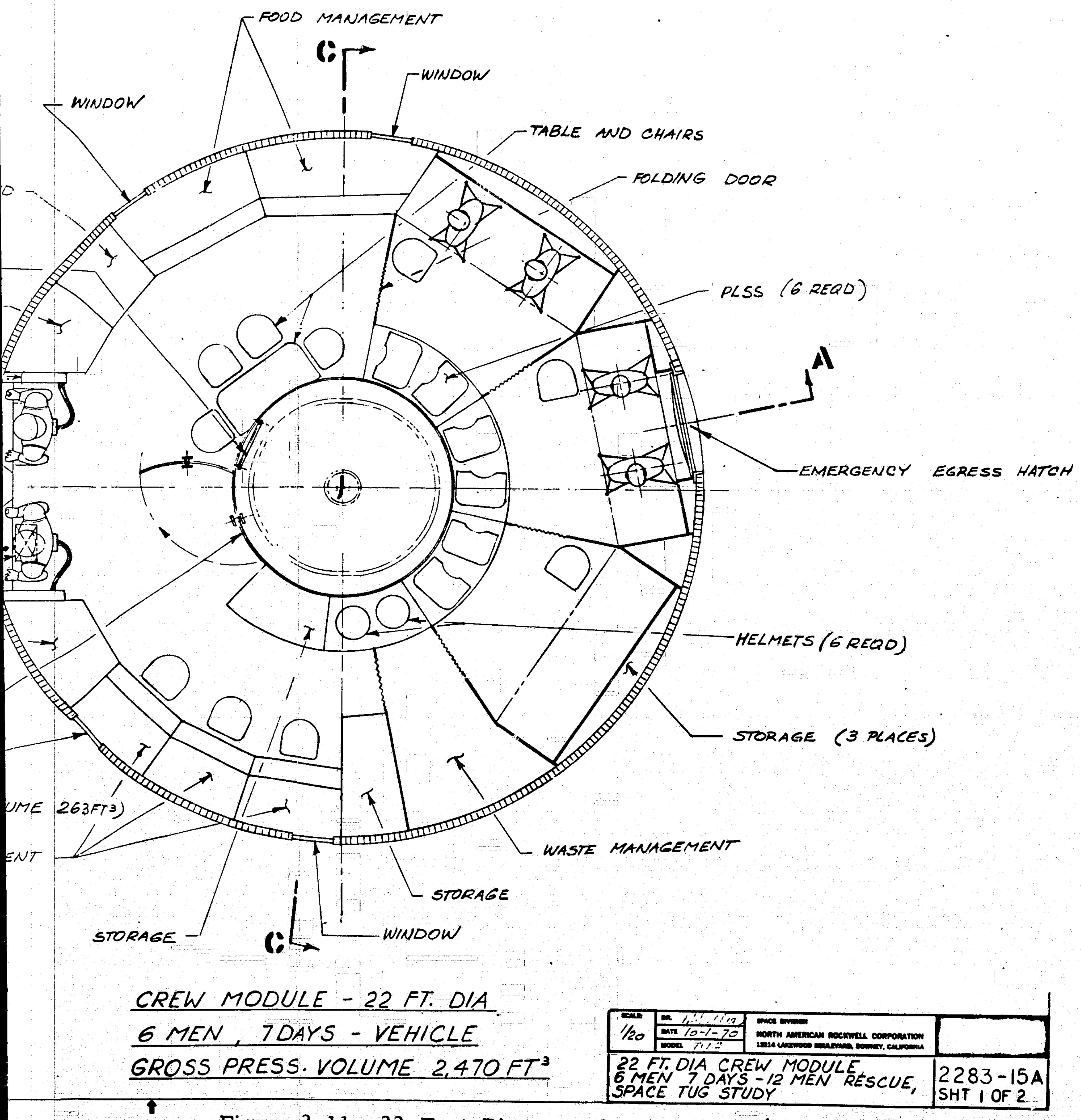


Figure 2-11. 22-Foot-Diameter Crew Module, 6 Man 7 Days - 12 Man Rescue Vehicle (Sheet 1 of 2)

2 FOLDOUT FRAME 3

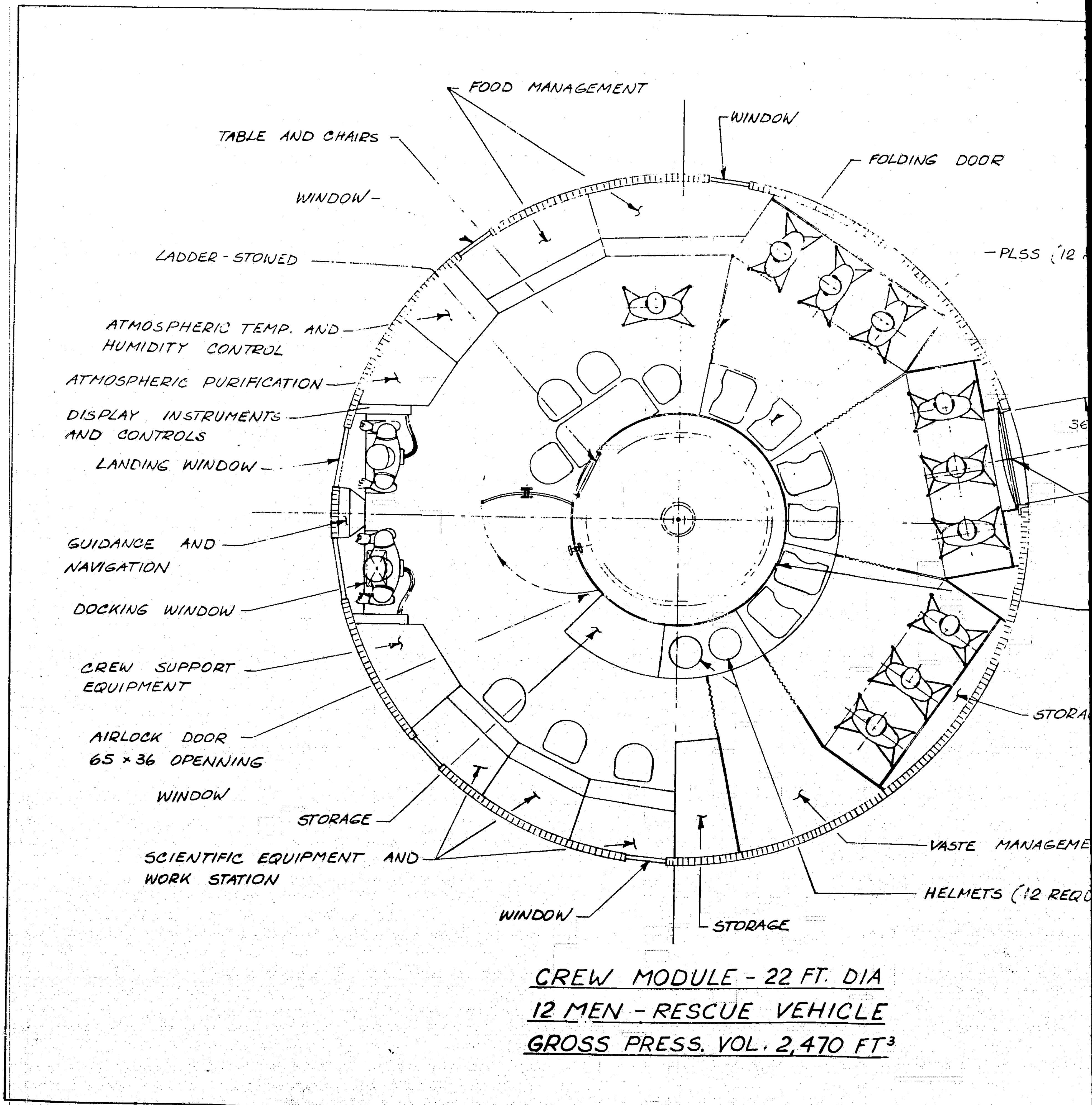


Figure 2-11. 22-Foot-Diameter Crew Module, 6 Ma
Rescue Vehicle (Sheet 2

FOLDOUT FRAME

FOLDOUT

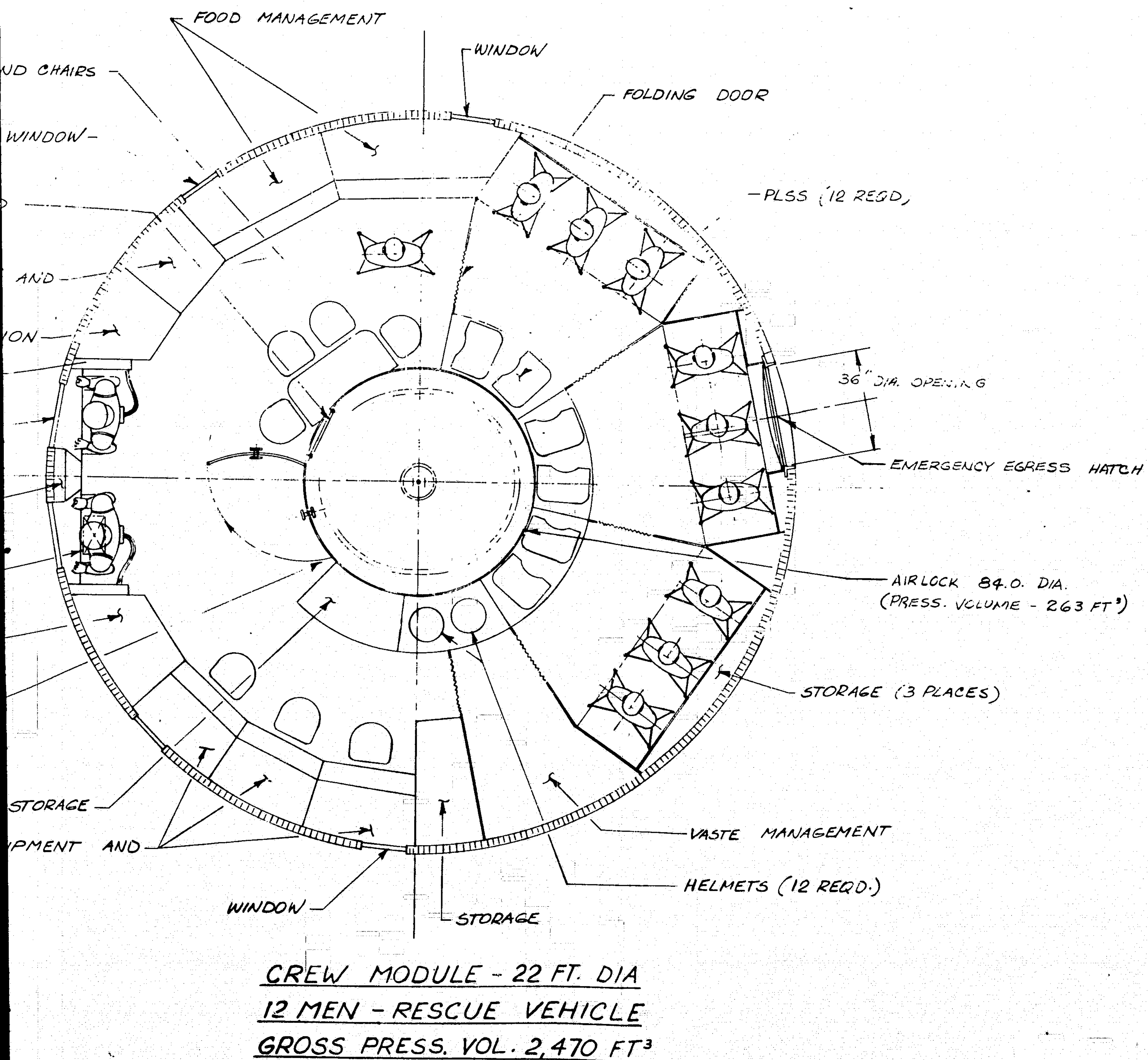
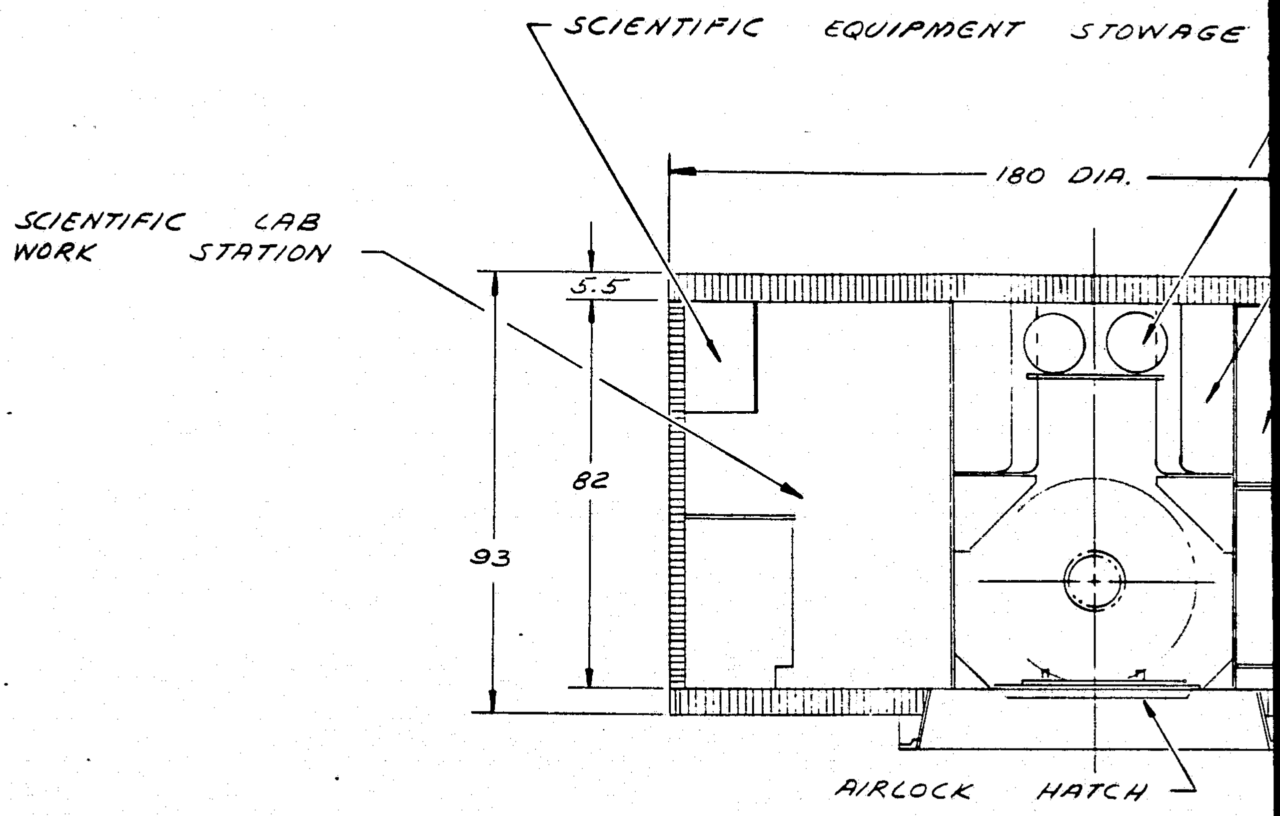
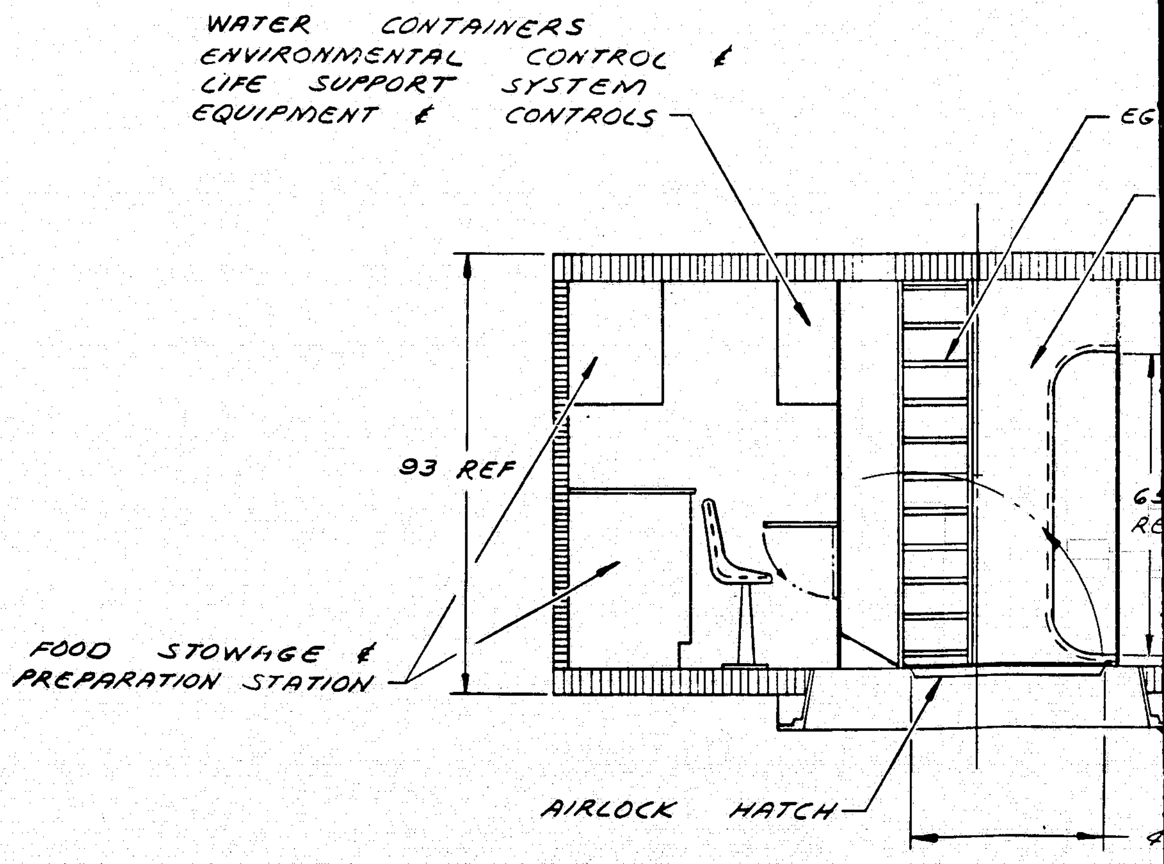


Figure 2-11. 22-Foot-Diameter Crew Module, 6 Man 7 Days - 12 Man Rescue Vehicle (Sheet 2 of 2)

FOLDOUT FRAME 2



SECTION A – A ROT



SECTION B – B

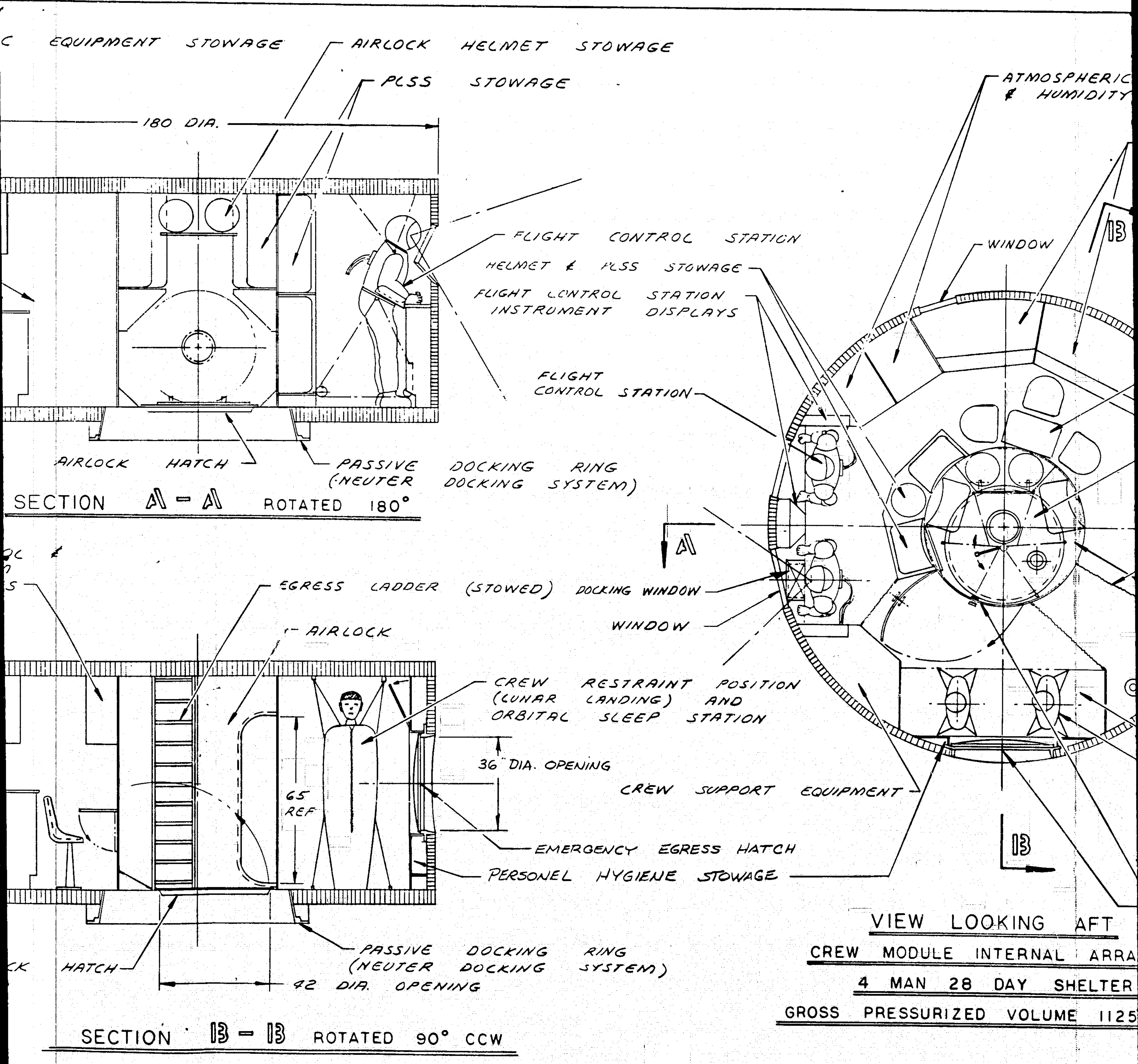


Figure 2-

FOLDOUT FRAME 2

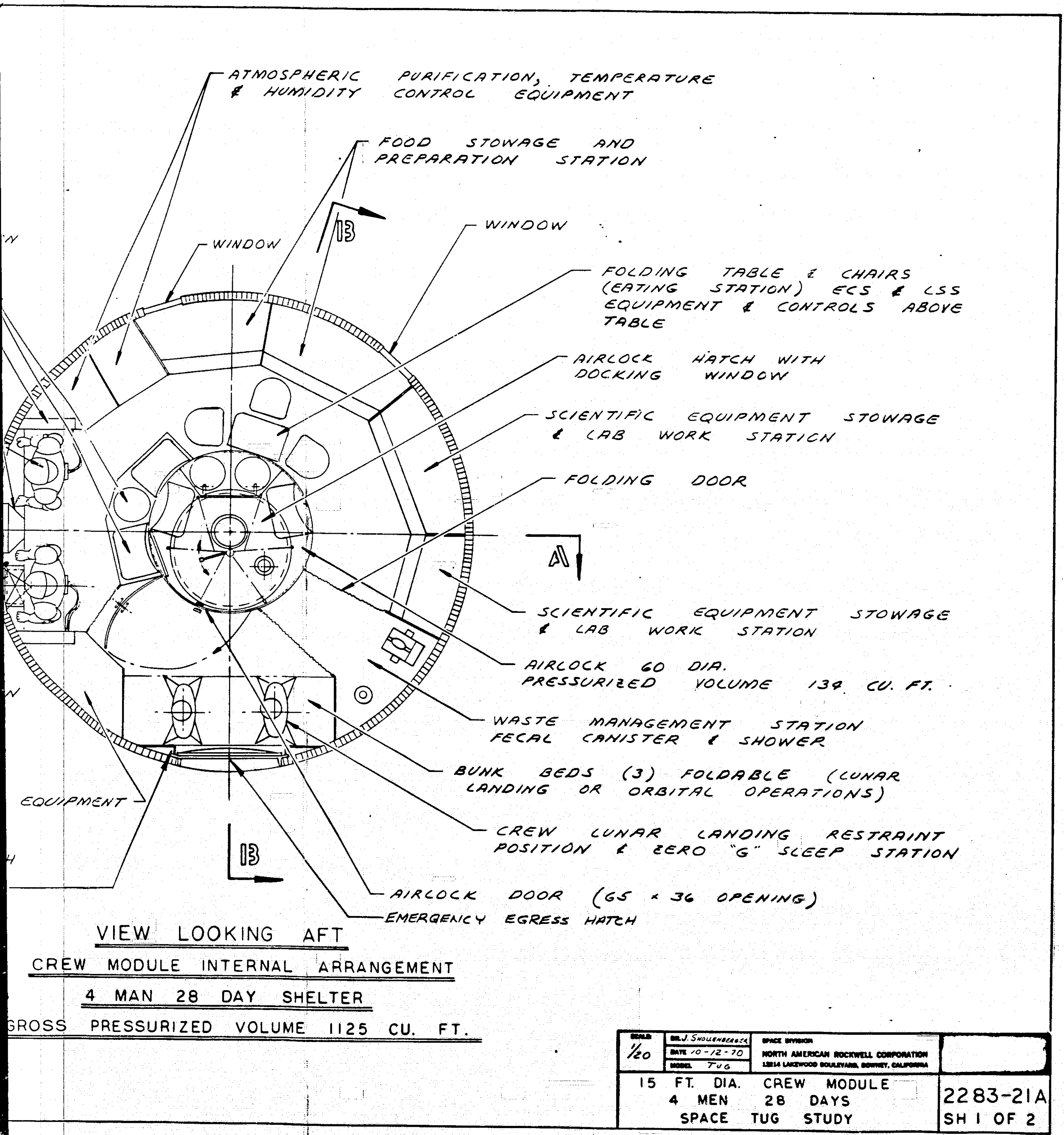
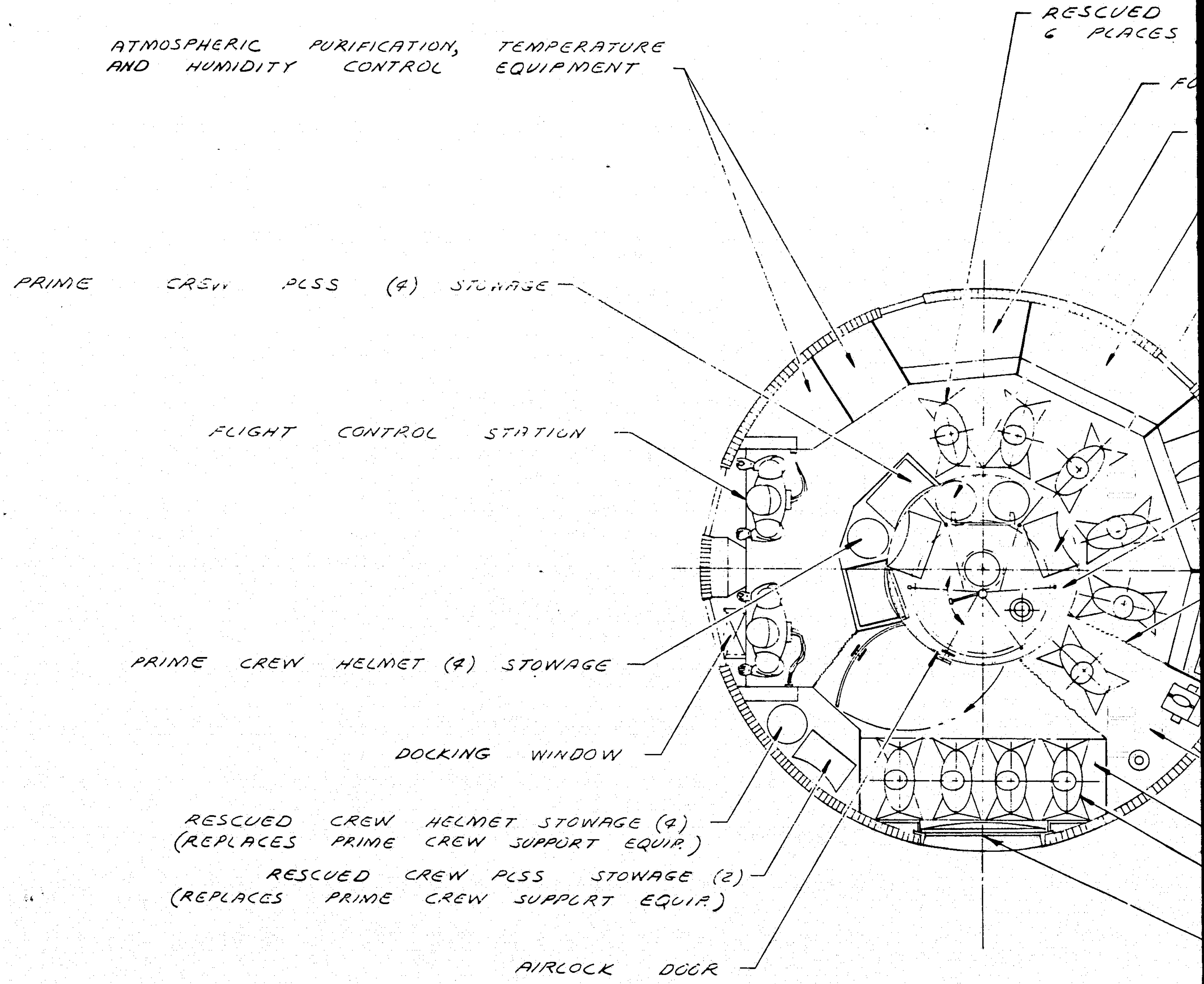


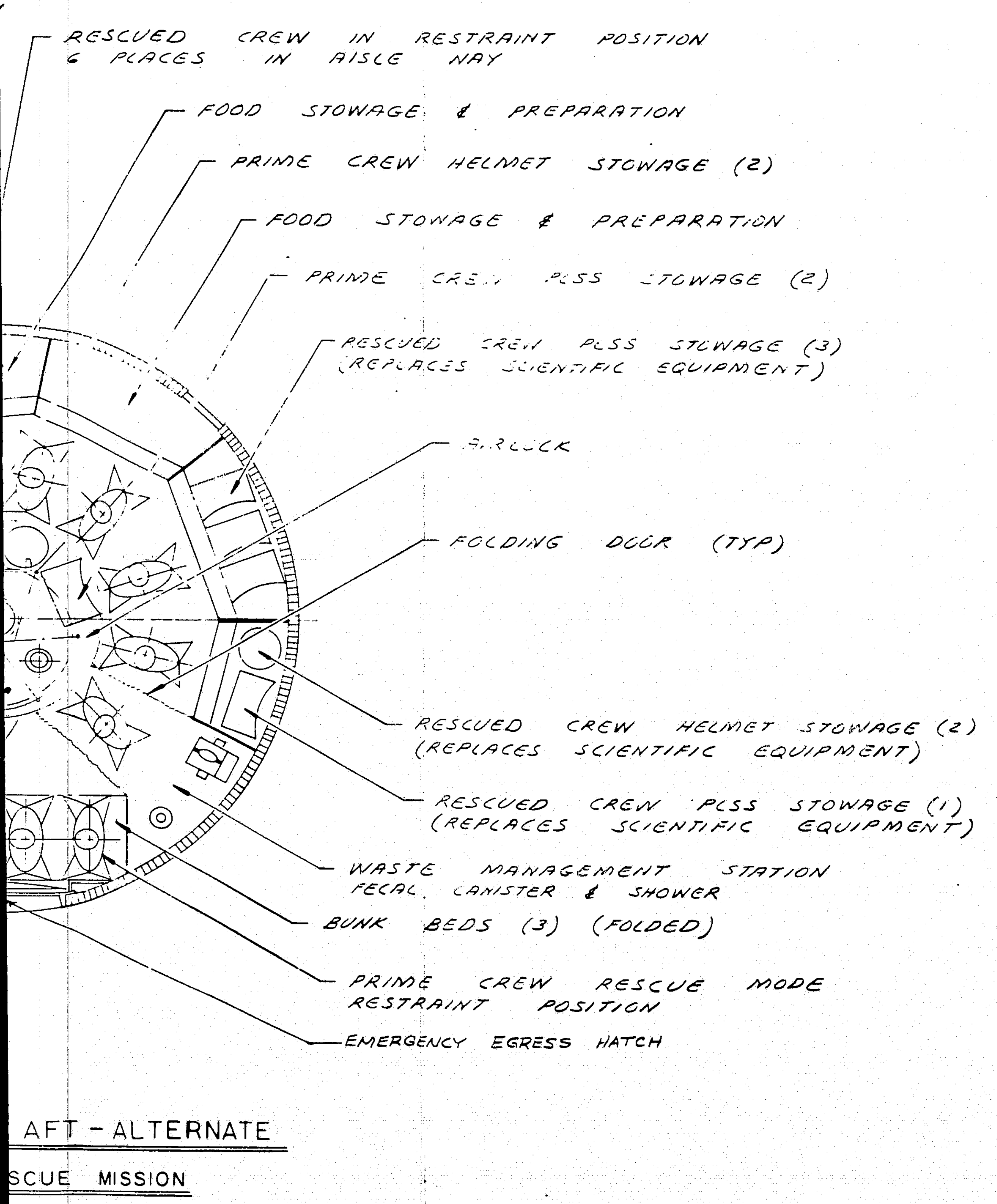
Figure 2-12. 15-Foot-Diameter Crew Module - 4 Man 28 Days (Sheet 1 of 2)



VIEW LOOKING AFT - ALTERNA

12 MAN RESCUE MISSION

FOLDOUT FRAME 1



2283 - 21A
SHEET 2 OF 2

Figure 2-12. 15-Foot-Diameter Crew Module - 4 Man 28 Days (Sheet 2 of 2)

difference is the removal of bunks in the four-man version and the addition of work stations. Since this version of the crew module is used as a lunar shelter, addition of work area is desirable. As demonstrated on the drawing, this configuration can be used as a 12-man rescue vehicle. For the rescue mode, the folding eating table and chairs are removed or folded and the crew members are restrained while standing in the aisle.

A passive docking ring has been shown on the aft end of the crew module. This arrangement is for an aft-located crew module configuration. For other vehicle configurations, this docking ring can be put on the forward end of the module or could be replaced with an active neuter system at either location. The overall visibility of the 15-foot-diameter (4.57-meter) vertical cylinder crew module to be compatible with all identified manned tug missions dictates that it be adopted as a baseline module for manned tug vehicle design configurations.

2.2.5 15-FOOT-DIAMETER (4.57-METER) CANDIDATE TUG VEHICLE CONCEPT DEFINITION

Single-Stage Concepts 1 and 2

In the first half of the study program, several basic tug candidate multipurpose approaches were identified. Different design optimization assumptions were used, as listed on Table 2-4. It was decided that the following concepts would be configurationally studied and defined: Concepts 1, 2, 3, 4, 5, 6, and 11. These concepts appeared at the time to satisfy the multipurpose approach best for completing each of the identified missions and were the primary candidates for midterm evaluation. The concepts may be grouped into similar configurations based on their tank and structural arrangements. The first such grouping is Concepts 1 and 2. They are defined in Figures 2-13, 2-14, and 2-15.

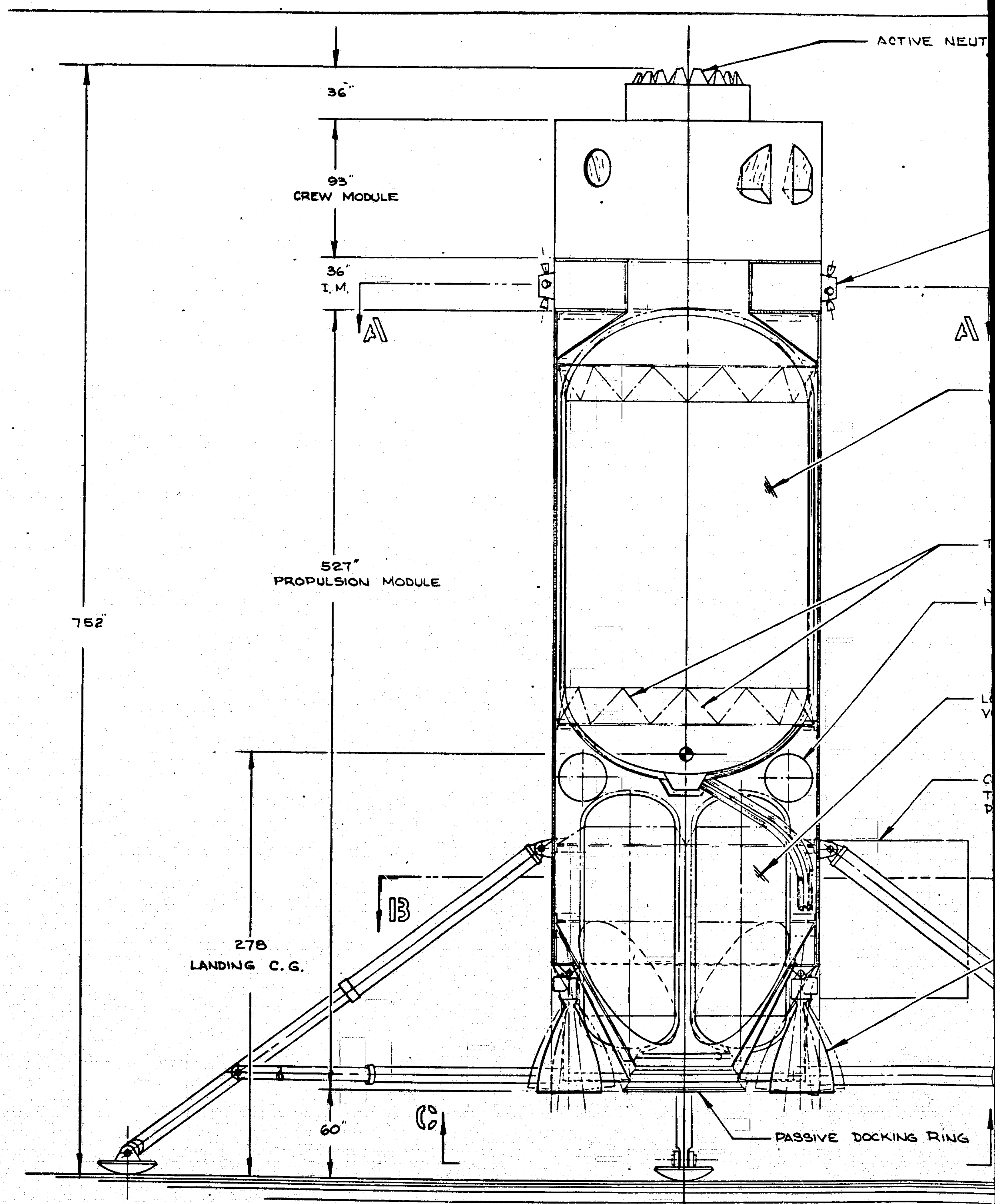
Figure 2-13 defines a 60,000- and 80,000-pound (27,215- and 36,287-kilogram) single-stage lunar lander, a size range that appeared to be representative. In each concept efforts were made to have common elements. One of commonality was in the IM. Each concept used a separable IM with swing-out ACS pods. Equipment arrangement was identical in all concepts. The CM used was a 15-foot-diameter (4.57-meter) vertical cylinder with the docking provision at either the aft or forward end, depending on the location of the CM in the vehicle. The exception is the last configuration. It used a 12-foot-diameter (3.66-meter) horizontal CM to obtain a comparison of the two-crew module integration approaches. In each concept the basic structural approach and refueling drogue assembly are also common elements. The first concept shown in a manned single-stage lunar lander with a PM having an 80,000-pound (36,287-kilogram) usable propellant capacity. The propulsion section has four engines based on an

Table 2-4. Propulsion Module Matrix

CONCEPT	PROPELLANT LOADING, 1000 LB (1000 Kg)	STAGING ARRANGEMENTS				
		GEOSYNCH	LUNAR LANDING	LOW EARTH ORBIT	PLANETARY	
					INNER	OUTER
1	80 (36)	SINGLE STAGE	SINGLE STAGE	SINGLE STAGE	SINGLE STAGE	SINGLE STAGE
2	52 (24)	TWO STAGE OR STAGE & TS	SINGLE STAGE (MODE A)	SINGLE STAGE	TWO STAGE	SINGLE STAGE
3	45 (20)	STAGE & TS	STAGE & TS	SINGLE STAGE	TWO STAGE	TWO STAGE (SECOND EXP)
4	41 (19)	TWO STAGE	SINGLE STAGE (MODE B)	SINGLE STAGE	TWO STAGE	TWO STAGE (SECOND EXP)
5	36 (16)	TWO STAGE	SINGLE STAGE (MODES C & D)	SINGLE STAGE	TWO STAGE	TWO STAGE (SECOND EXP)
6	31 (14)	TWO STAGE (SECOND REC)	STAGE & TS	SINGLE STAGE	SINGLE STAGE	TWO STAGE (SECOND EXP)
7	27 (12)	SINGLE STAGE	STAGE & TS	SINGLE STAGE	SINGLE STAGE	TWO STAGE (SECOND EXP)
8	23 (10)	TWO STAGE (SECOND EXP)	STAGE & TS (MODES B, C, & D)	SINGLE STAGE	SINGLE STAGE	—
11	9/48 (4/22)	SMALL STAGE & TS (TS EXPENDED)	SMALL STAGE & TS (MODE A)	SMALL STAGE	SMALL STAGE & TS (TS EXPENDED)	SMALL STAGE & TS (BOTH EXPENDED)

MISSION FROM WHICH CONCEPT ORIGINATED

PARTIALLY OR FULLY EXPENDED IN ACCOMPLISHING MISSION



LUNAR LANDING TUG
(80K PROPELLANT CAP.)

FOLDOUT FRAME 1

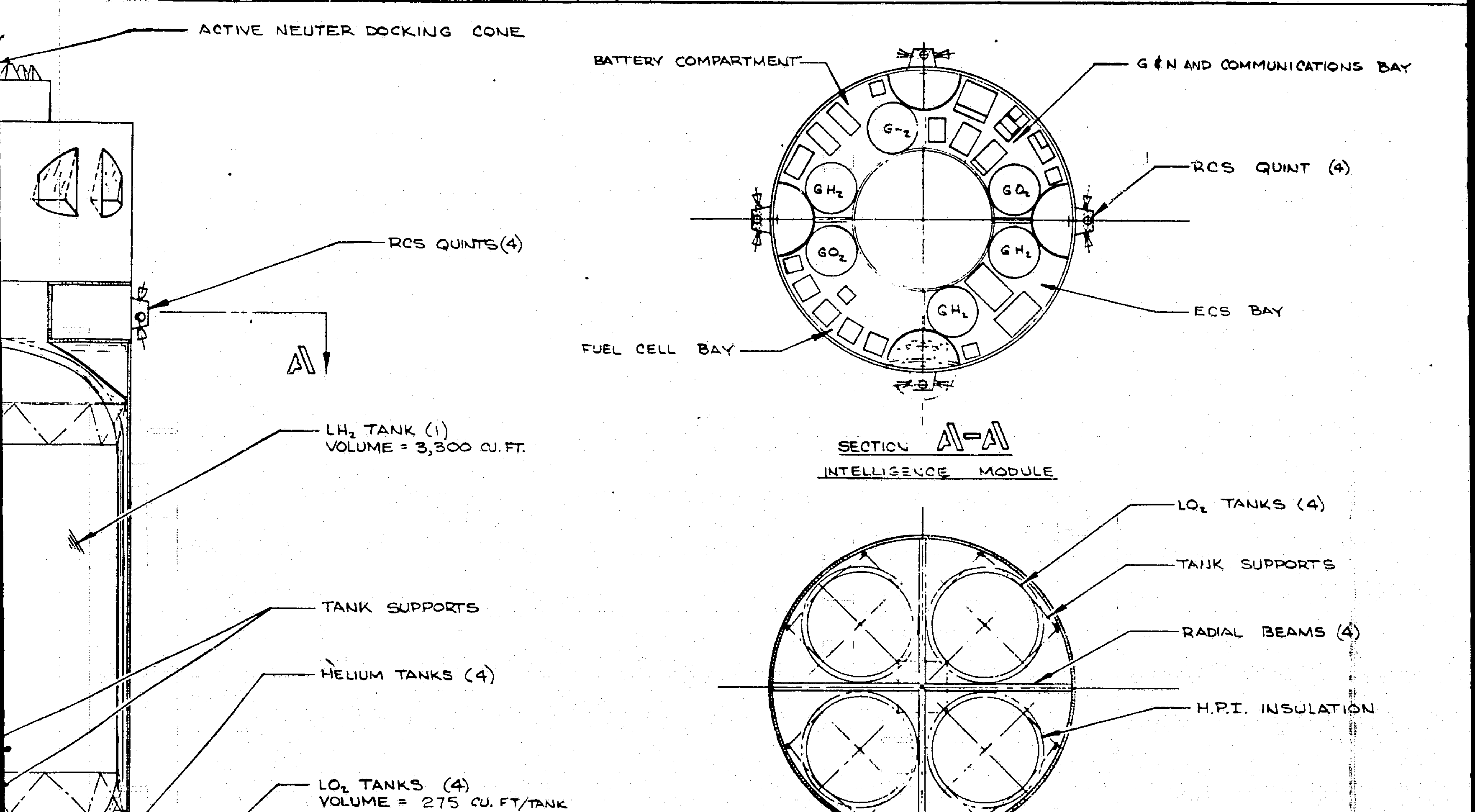


Figure 2-13. 60K and

FOLDOUT FRAME 2

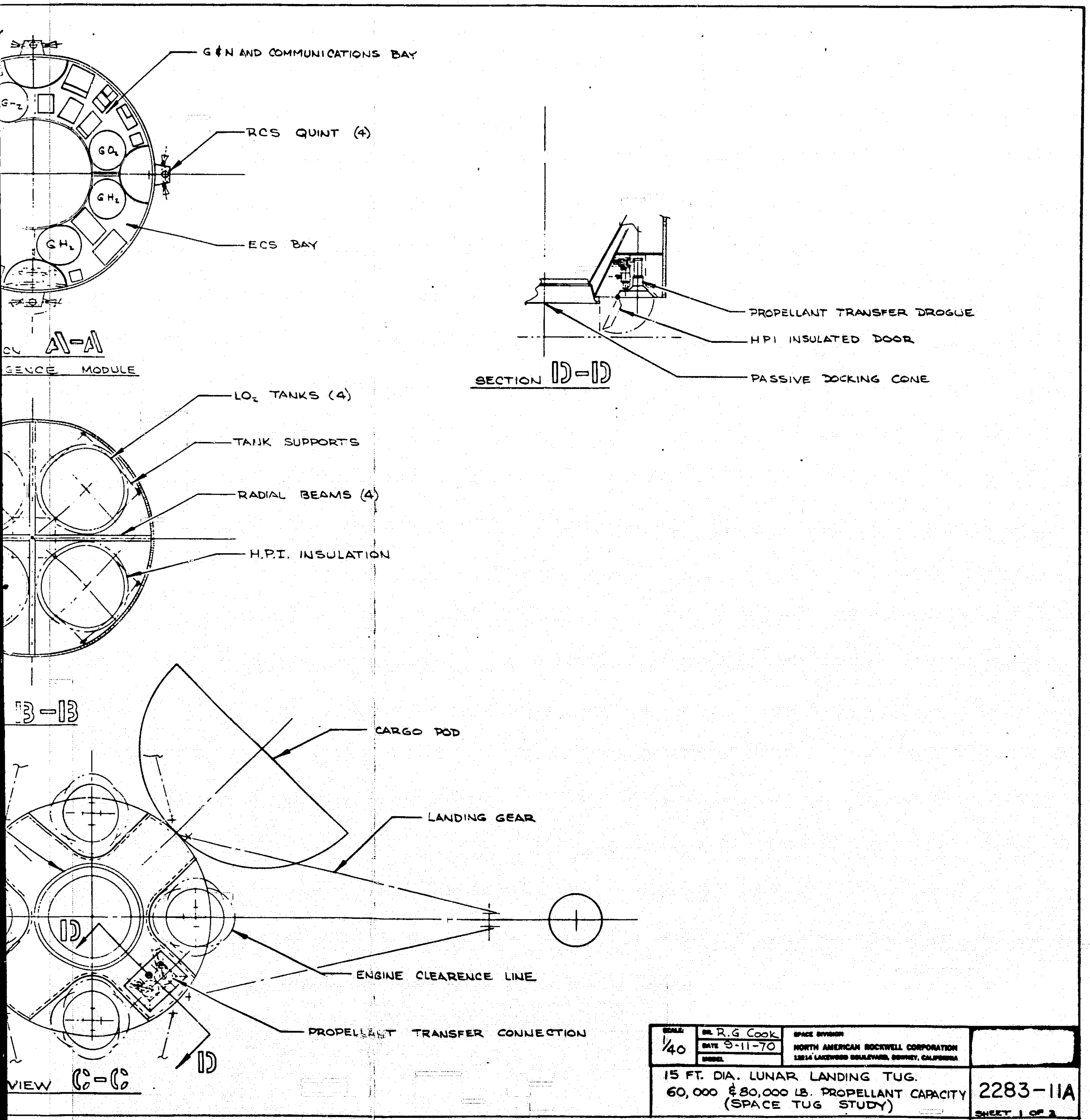
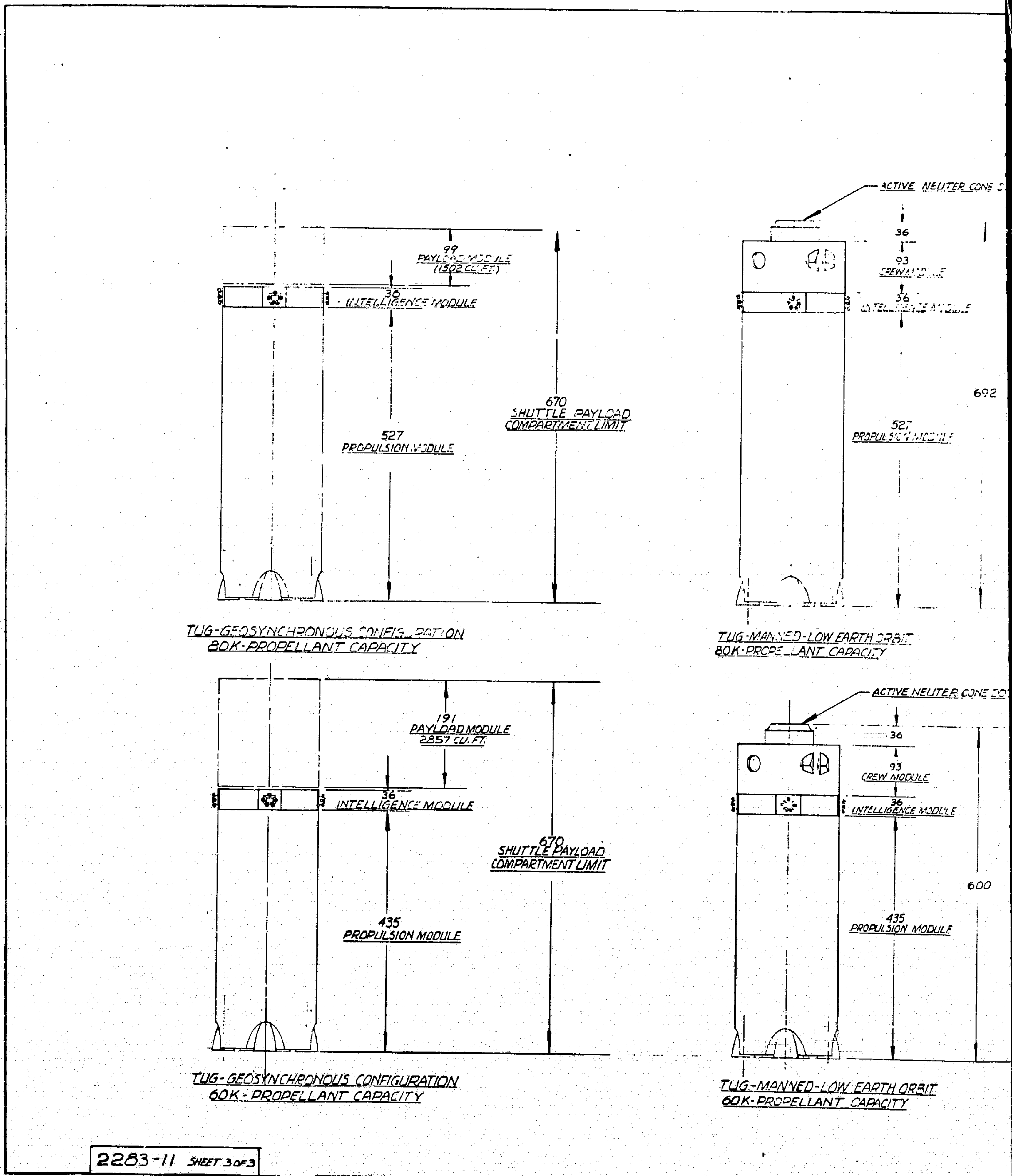


Figure 2-13. 60K and 80K Pound Propellant Capacity 15-Foot-Diameter Lunar Landing Tug (Sheet 1 of 2)

FRAME 2

FOLDOUT FRAME 3



FOLDOUT FRAME 1

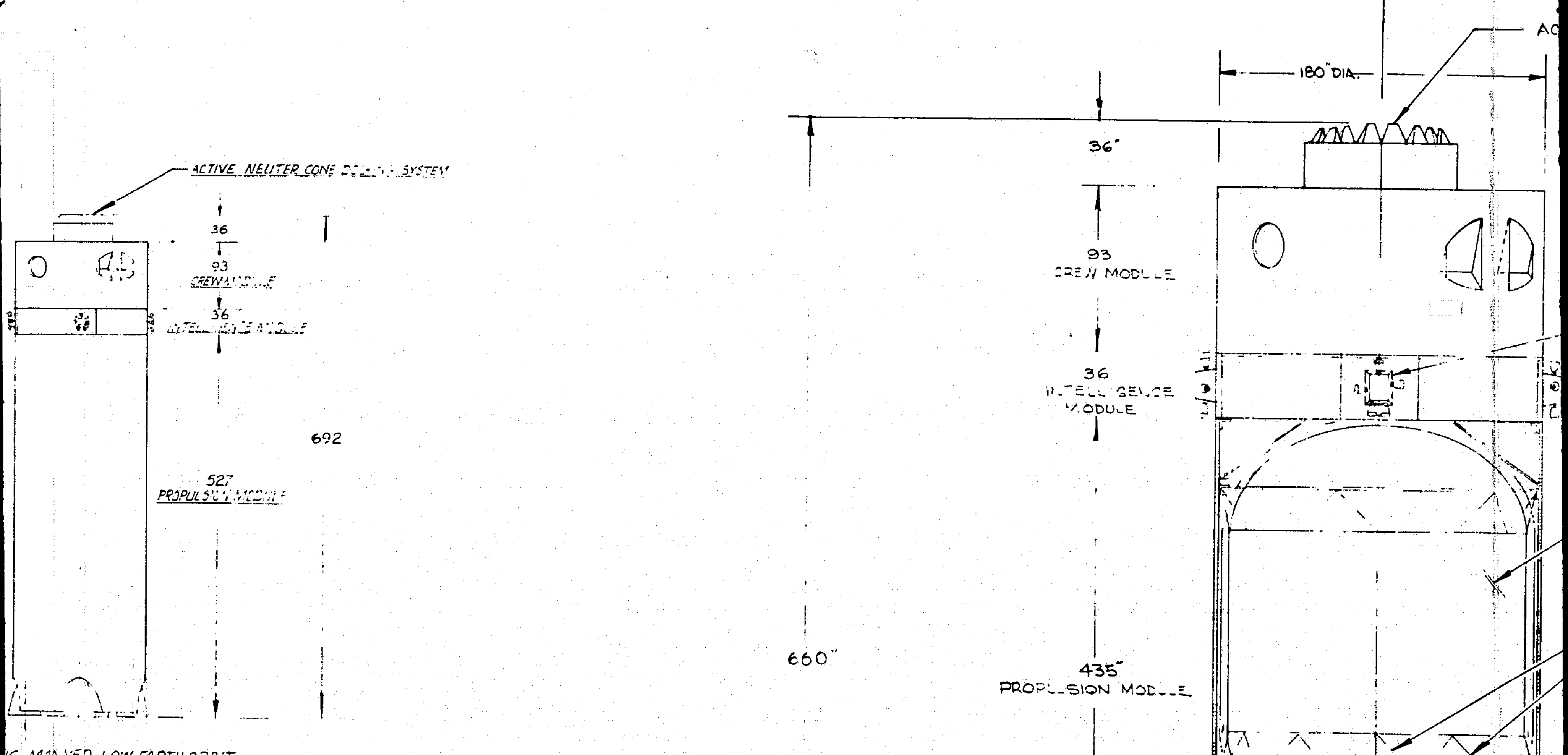


Figure 2-13. 60

FOLDOUT FRAME

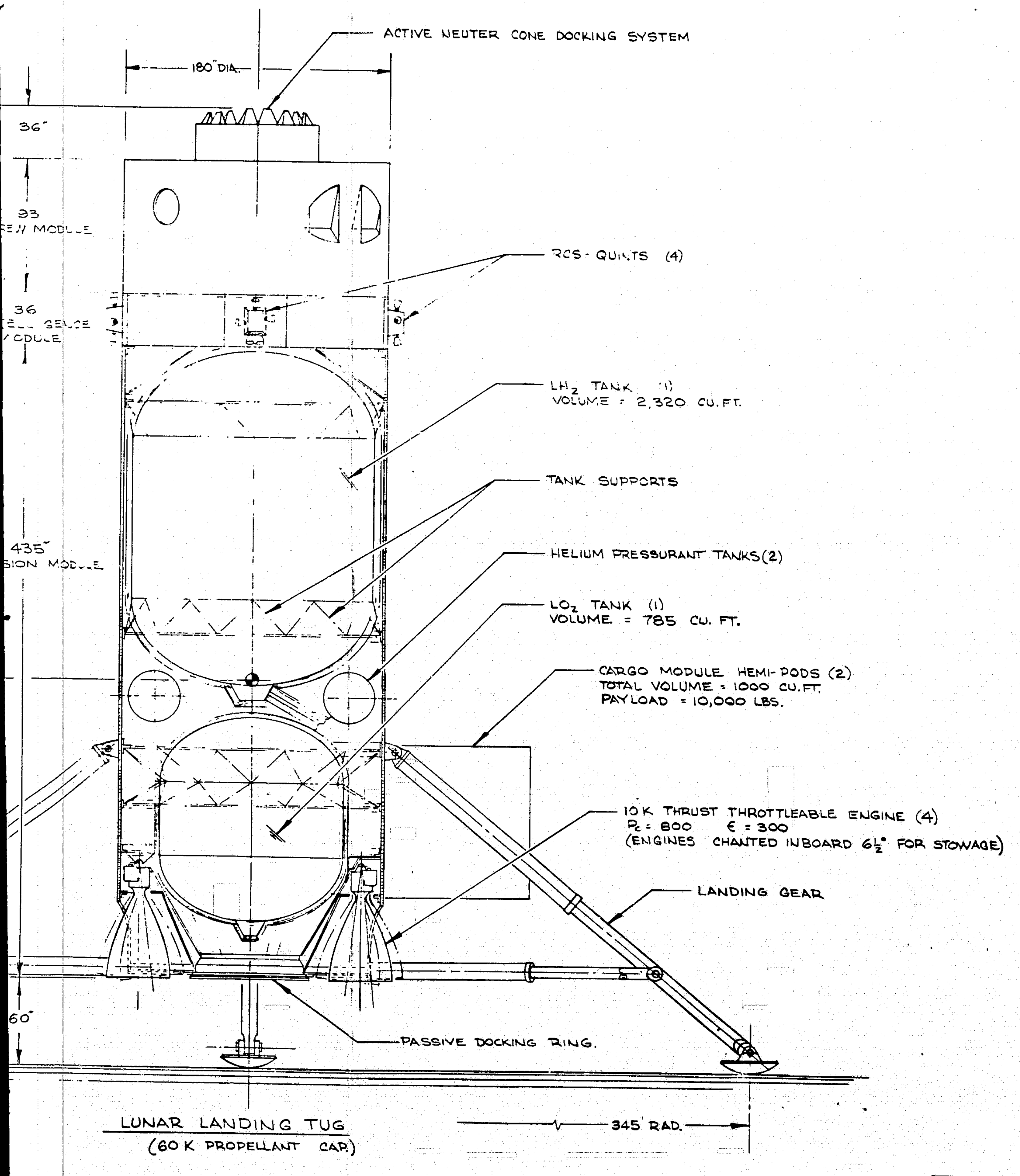
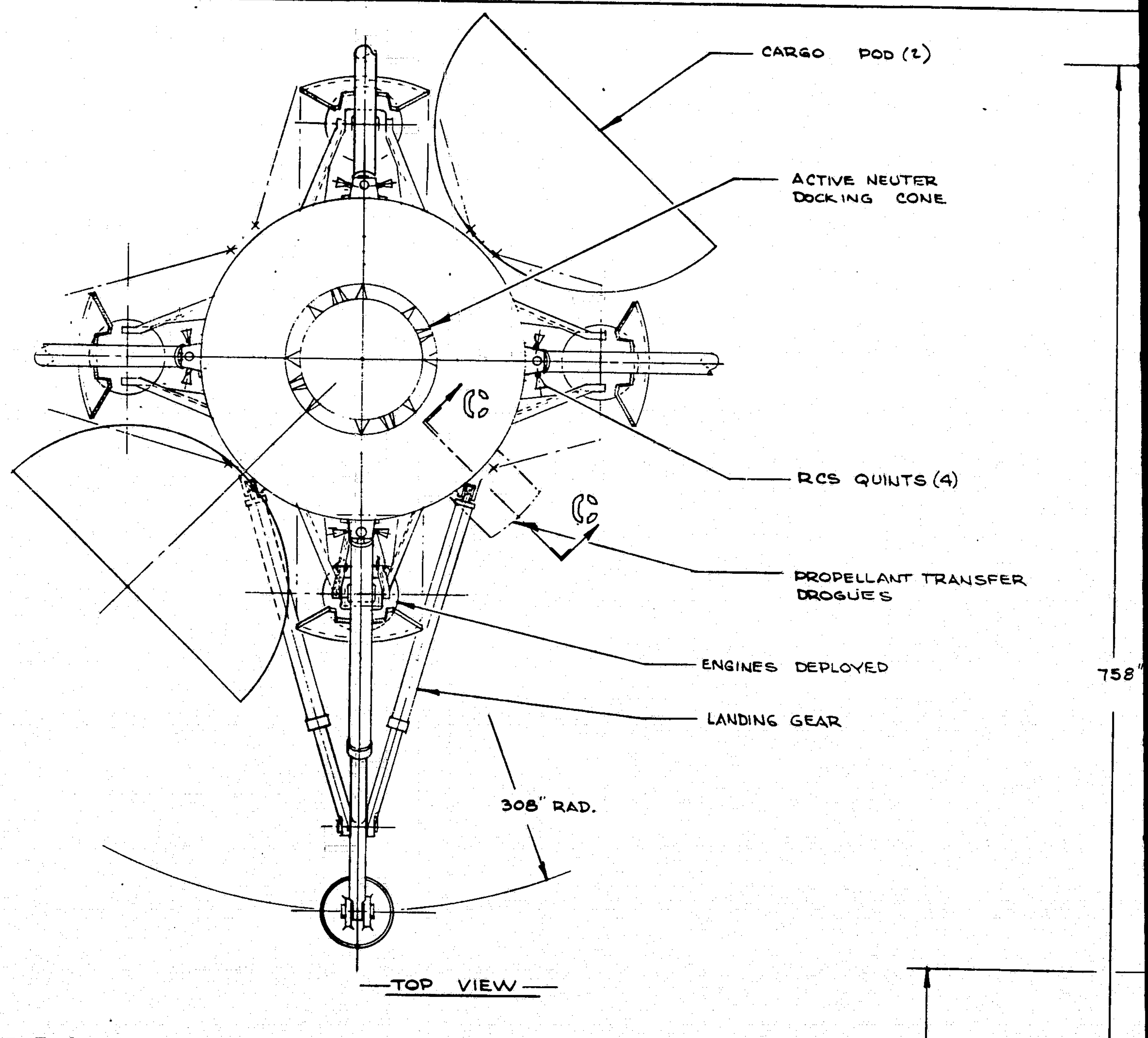
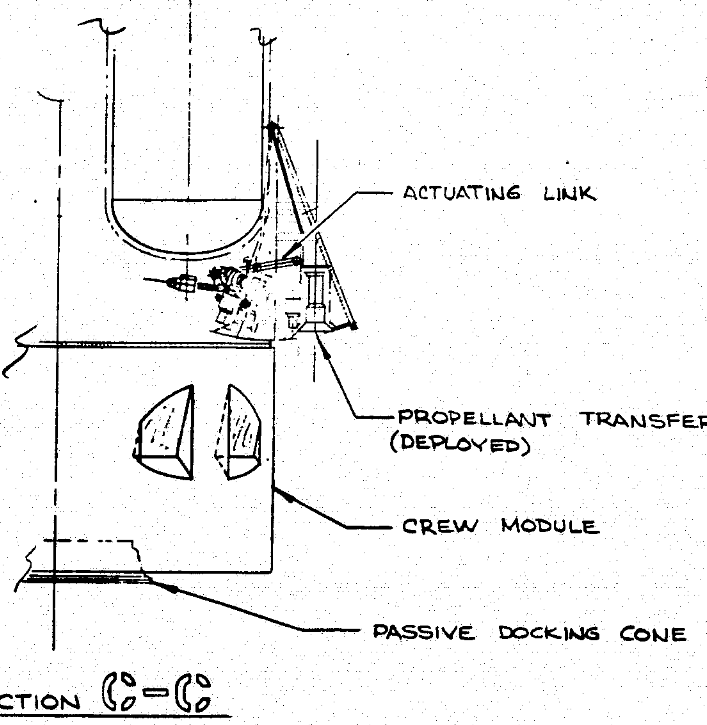


Figure 2-13. 60K and 80K Pound Propellant Capacity 15-Foot-Diameter Lunar Landing Tug (Sheet 2 of 2)



—TOP VIEW—



SECTION C–C

2283-12 SHT. 2 OF 3

FOLDOUT FRAME /

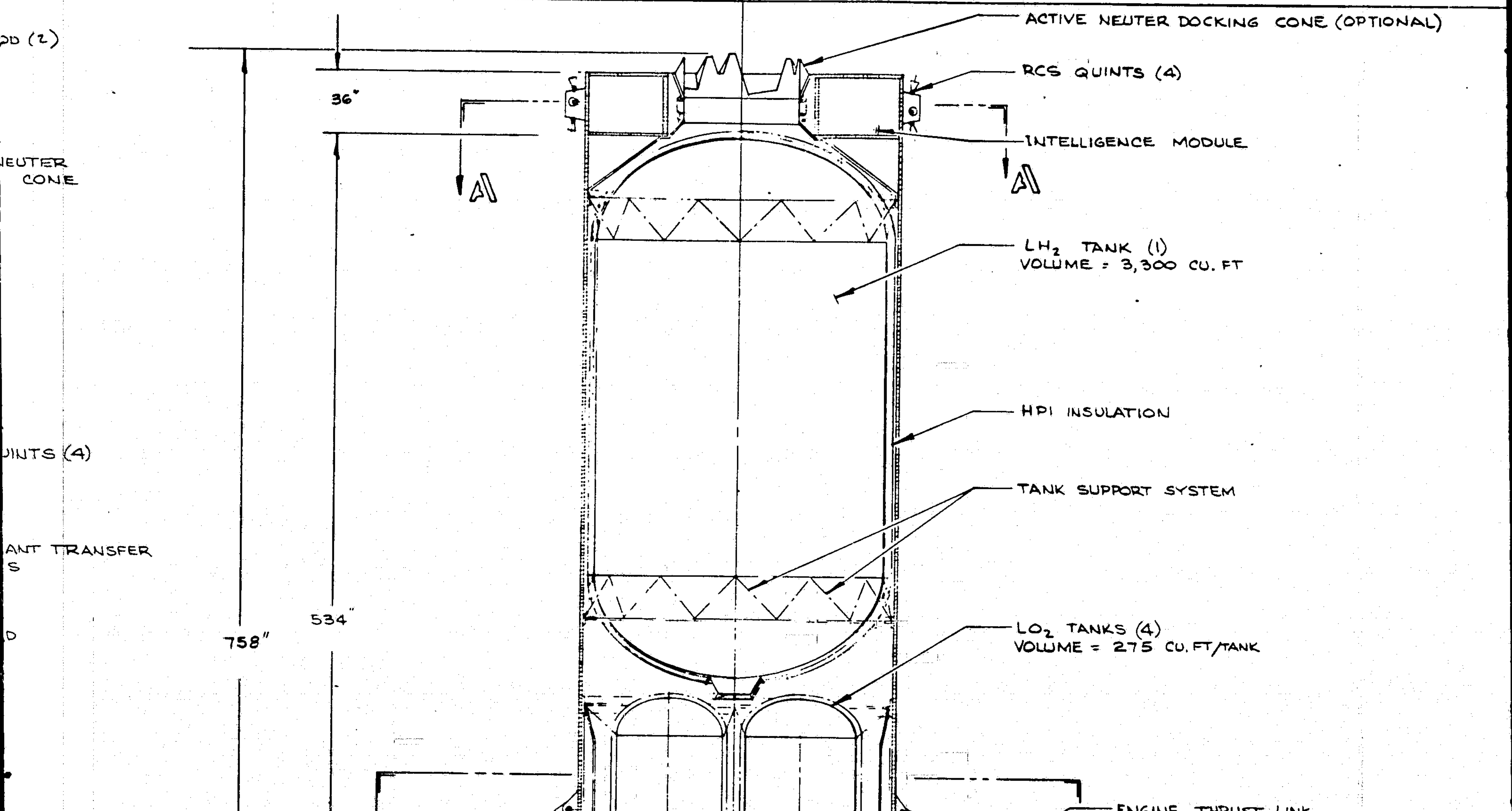


Figure 2-14. Deplo
Capacity

FOLDOUT FRAME 2

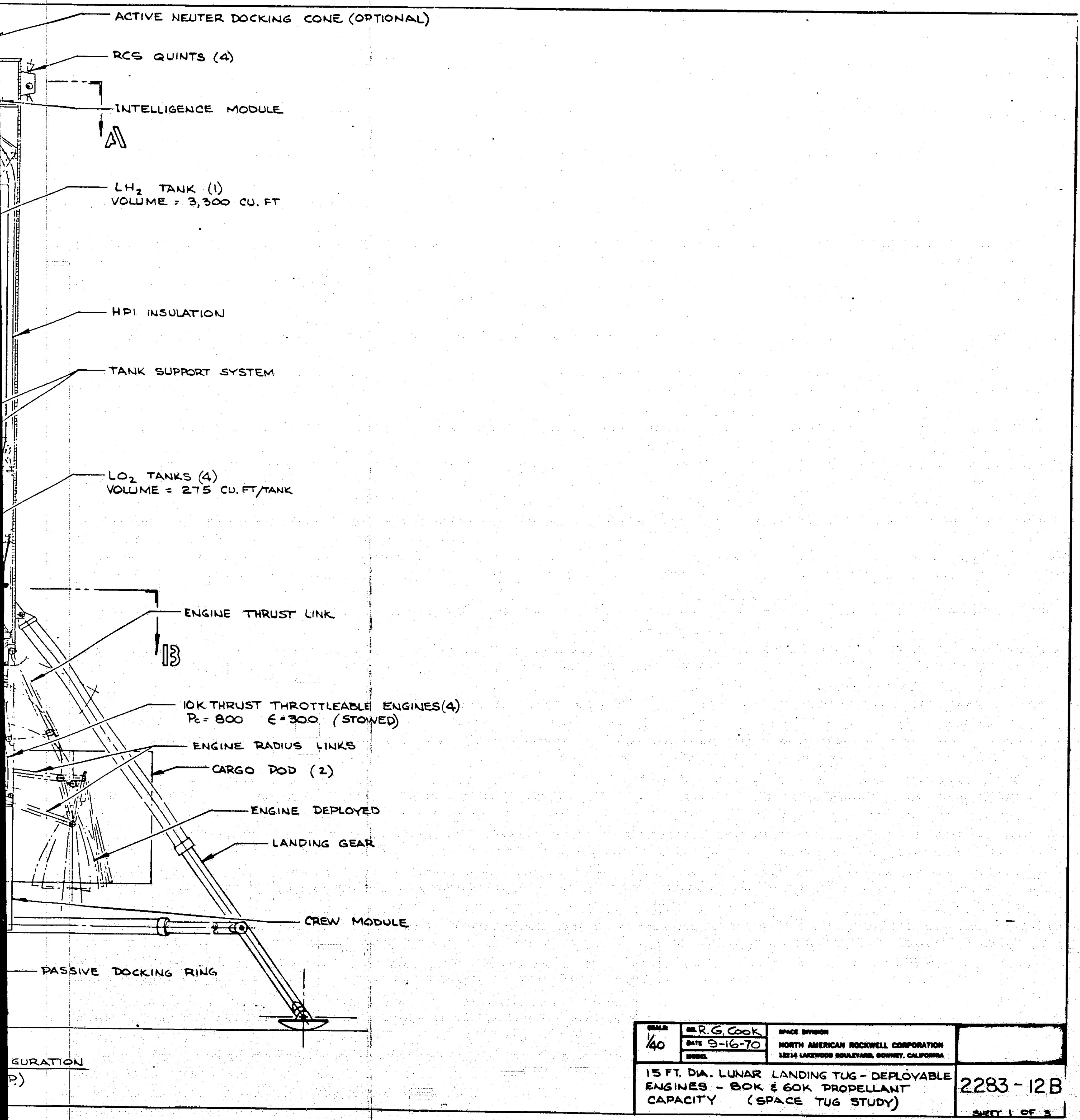
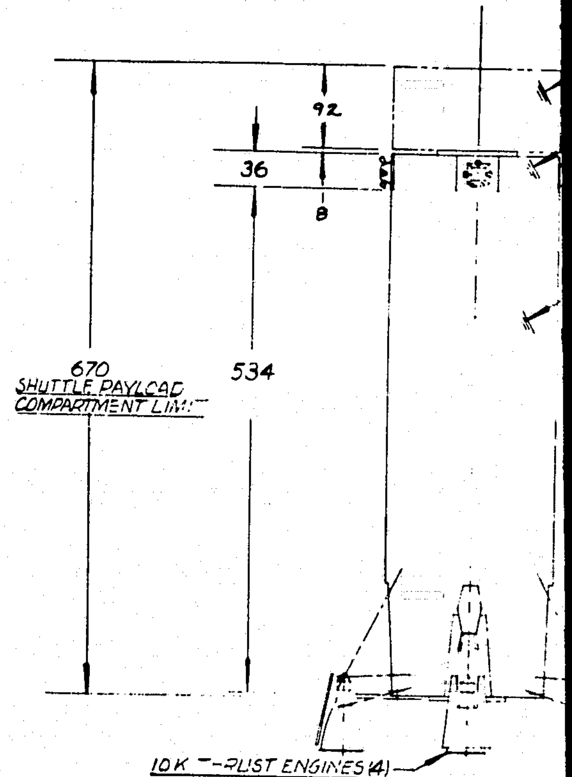
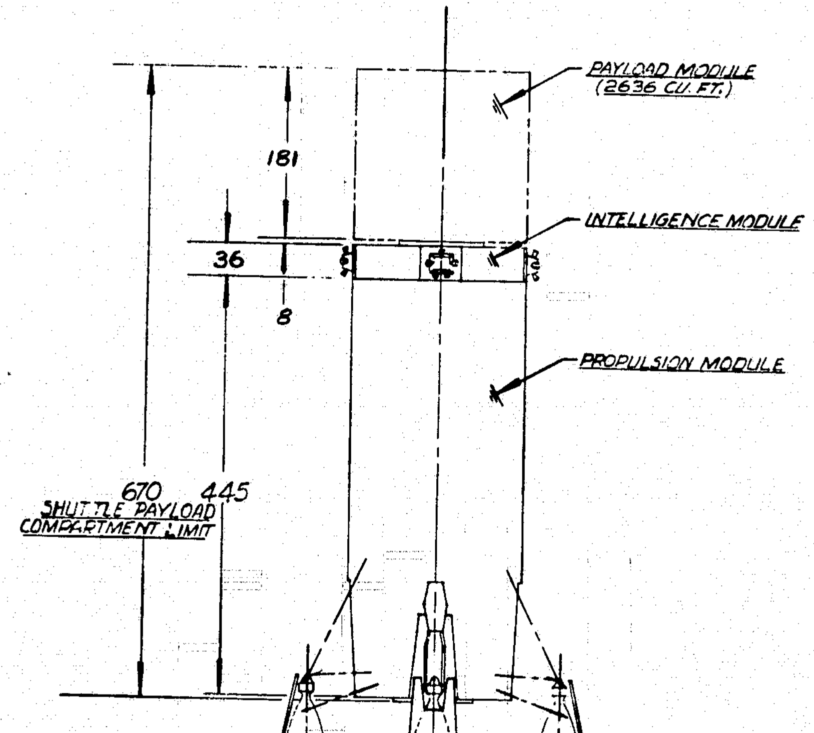


Figure 2-14. Deployable Engines - 80K and 60K Pounds Propellant Capacity 15-Foot-Diameter Lunar Landing Tug (Sheet 1 of 2)



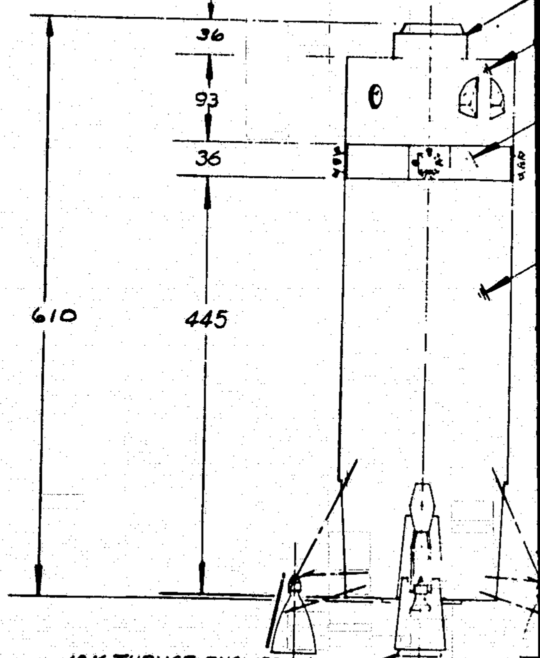
10K THRUST ENGINES (4)

UNMANNED LEO & GEO
80K PROPELLANT CAPAC
1/100 SCALE



10K THRUST ENGINES (4)

UNMANNED LEO & GEOSYNCHRONOUS CONFIGURATION
60K PROPELLANT CAPACITY
1/100 SCALE



10K THRUST ENGINES (4)

MANNED LOW EARTH ORBIT CON
60K PROPELLANT CAPACI
1/100 SCALE

2283-11 SHEET 3 OF 3

FOLDOUT FRAME 1

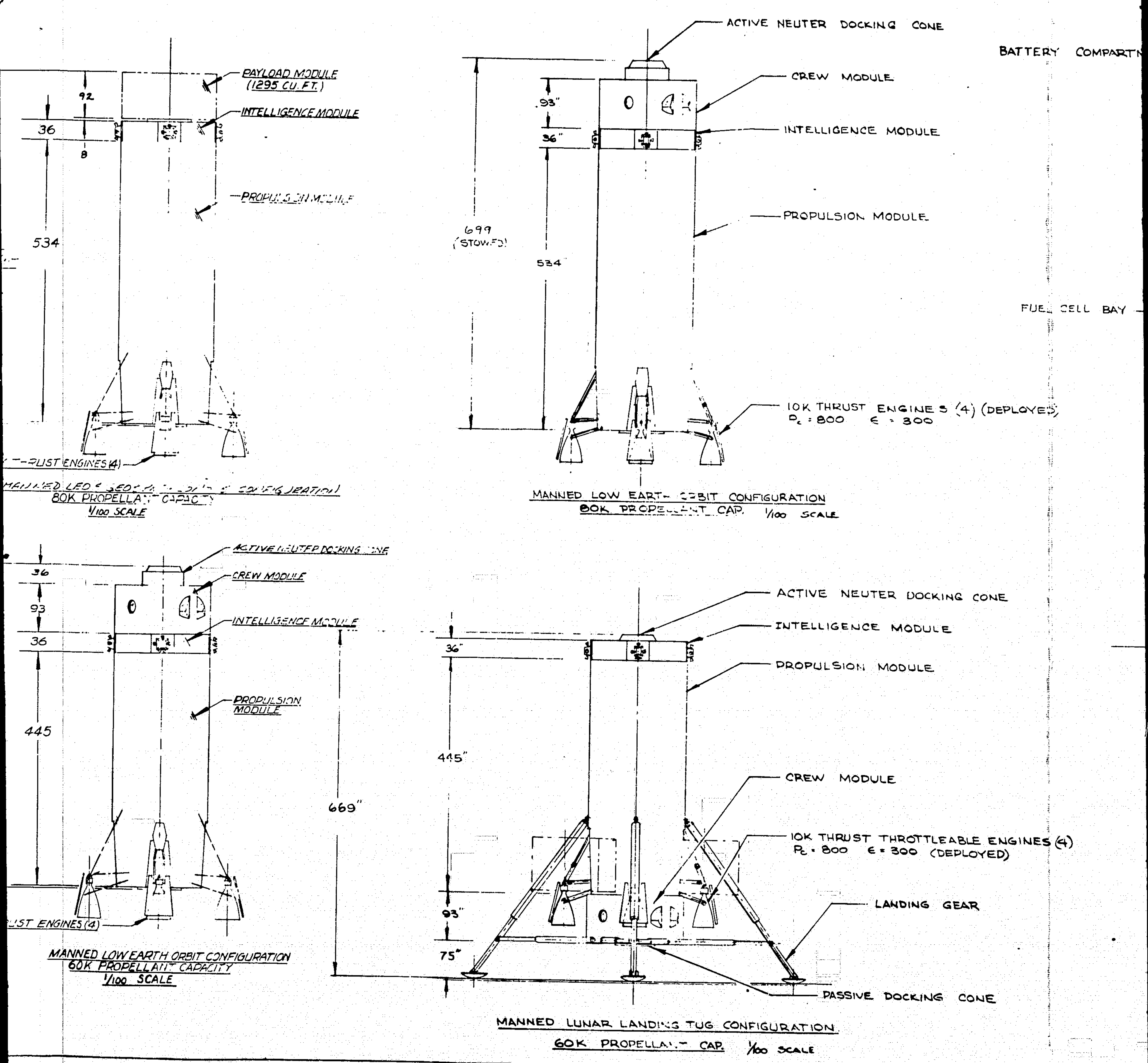


Figure 2-14. De
Capac

FOLDOUT FRAME 2

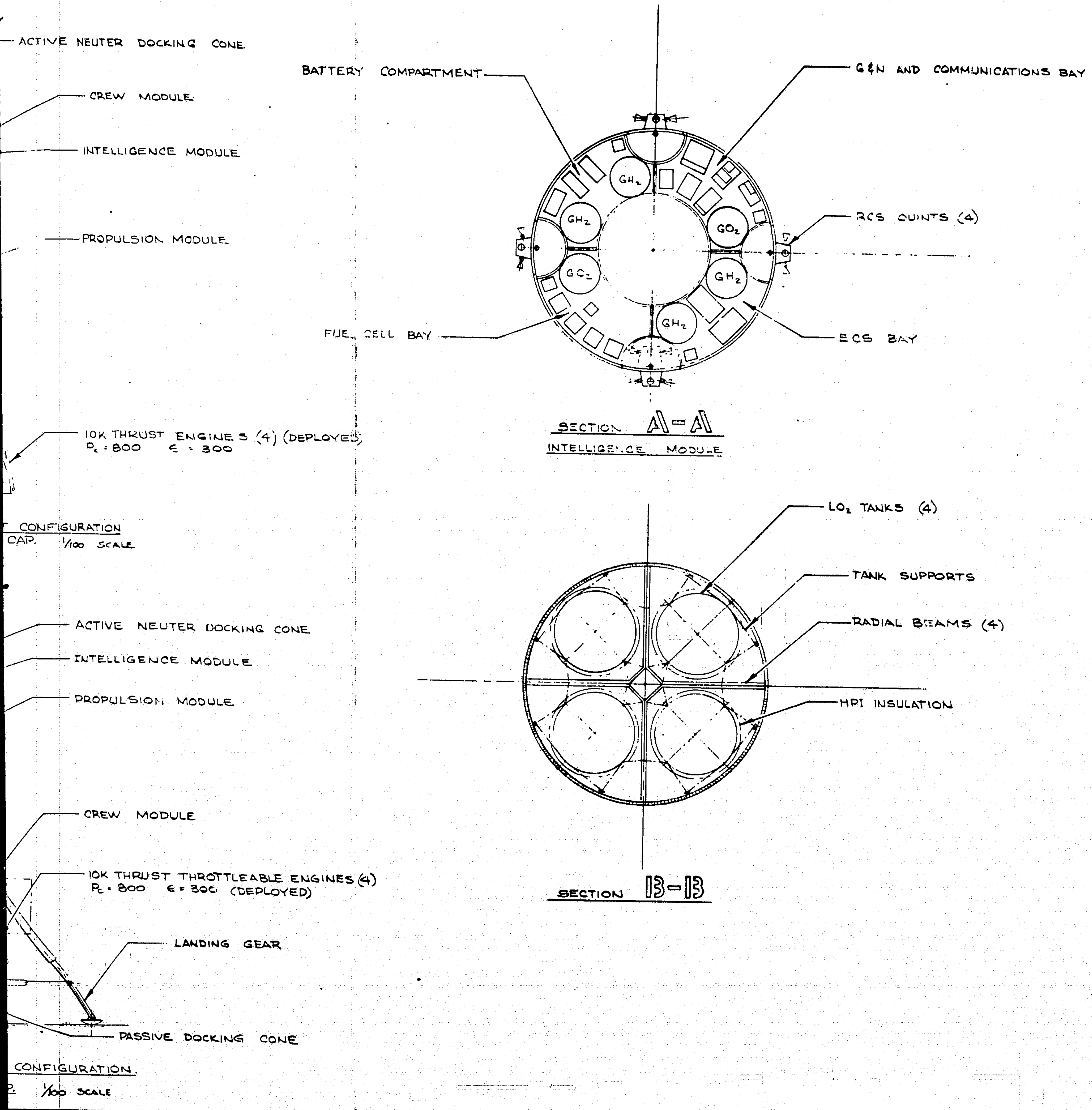
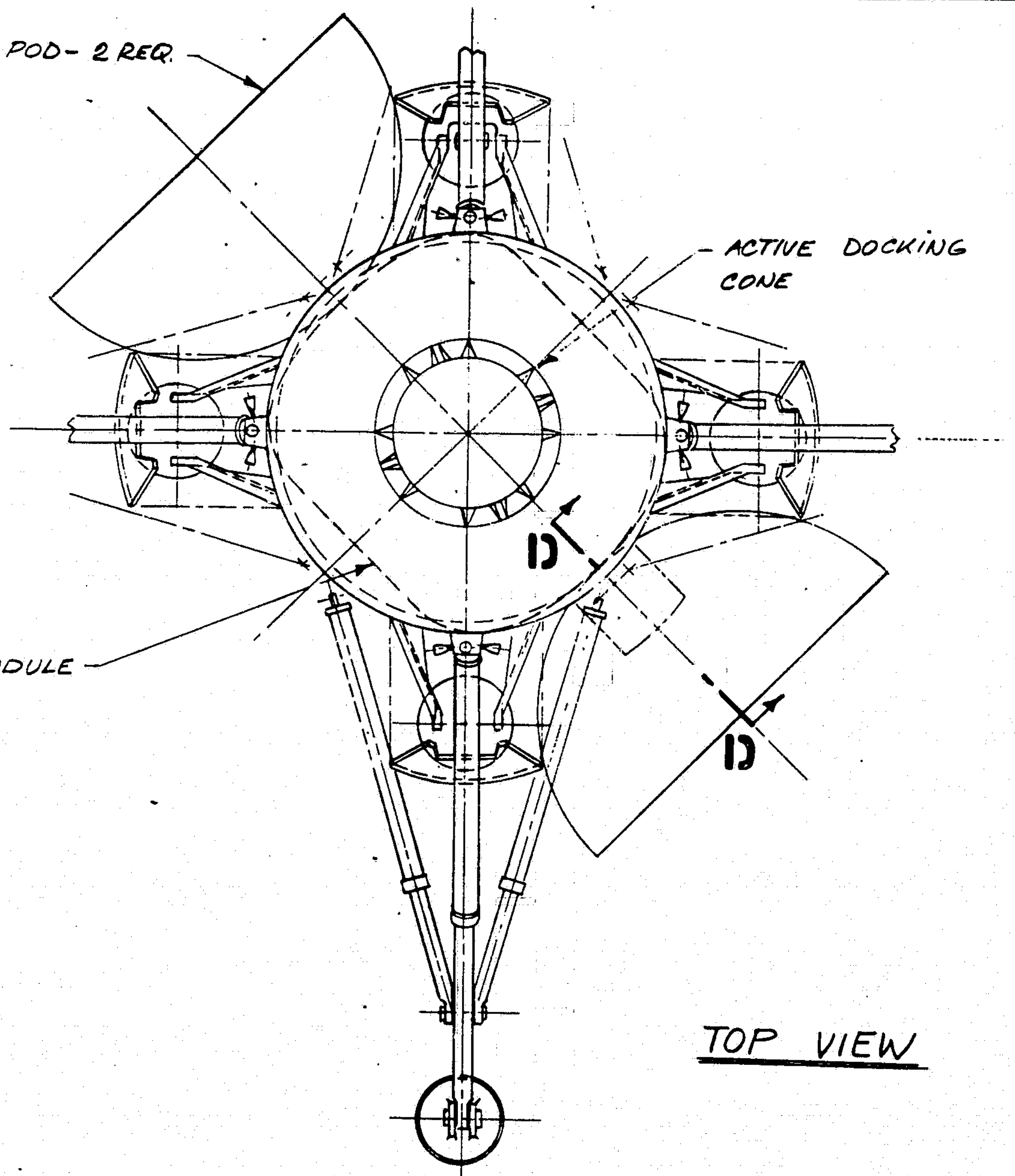
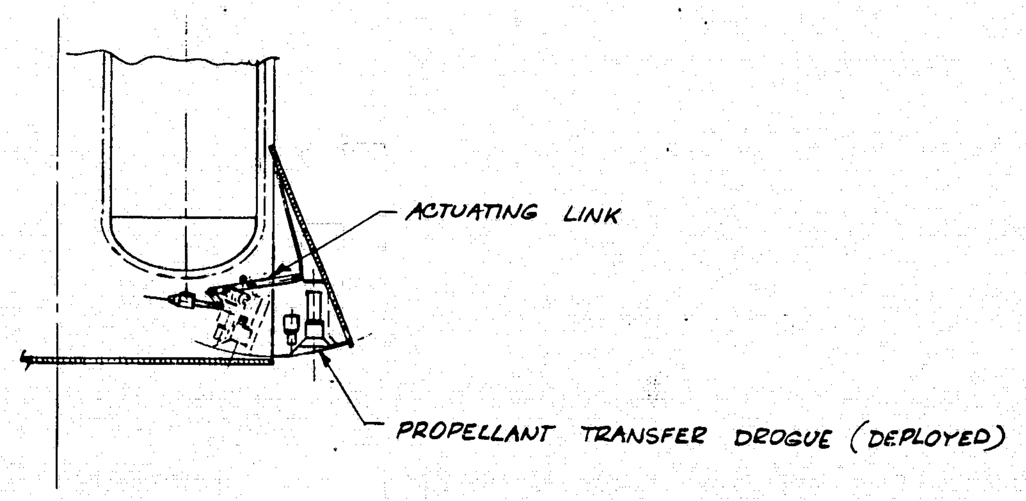


Figure 2-14. Deployable Engines - 80K and 60K Pounds Propellant Capacity 15-Foot-Diameter Lunar Landing Tug (Sheet 2 of 2)

FOLDOUT FRAME 3

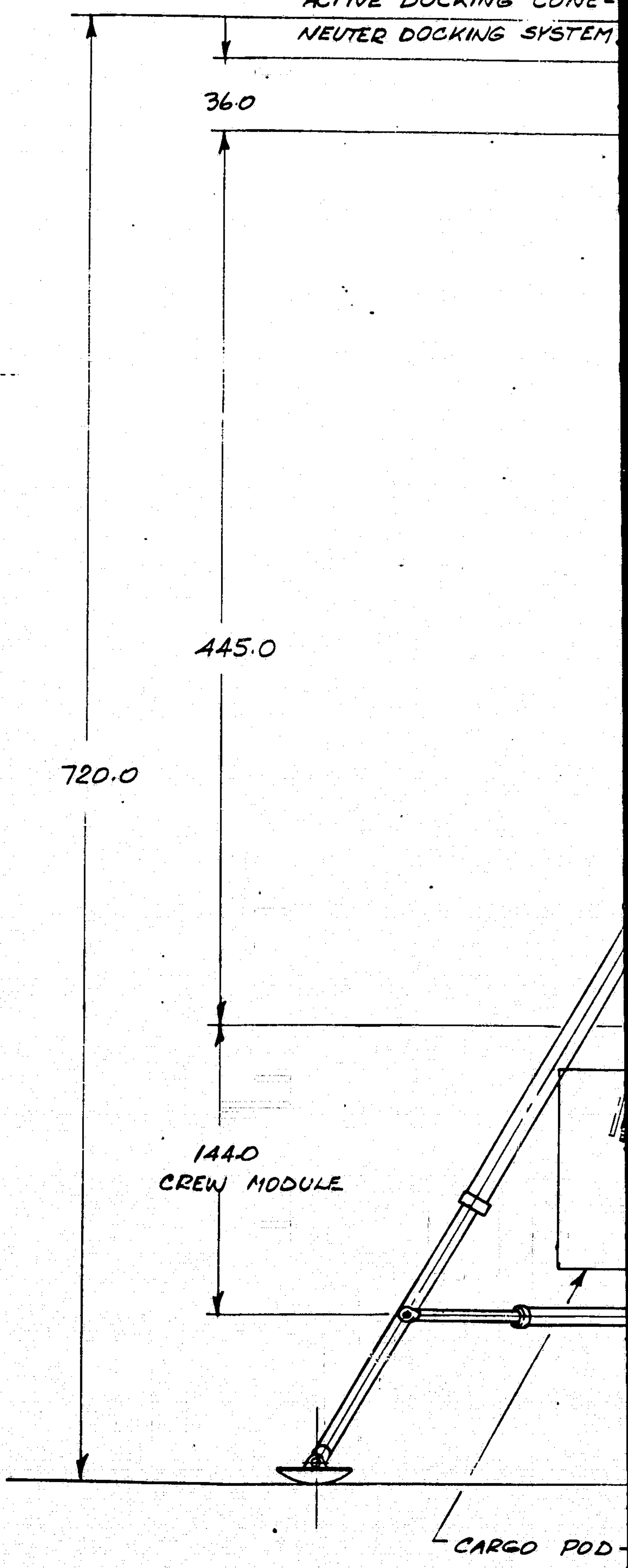


TOP VIEW



SECTION D-D

ROTATED 45°



FOLDOUT FRAME 1

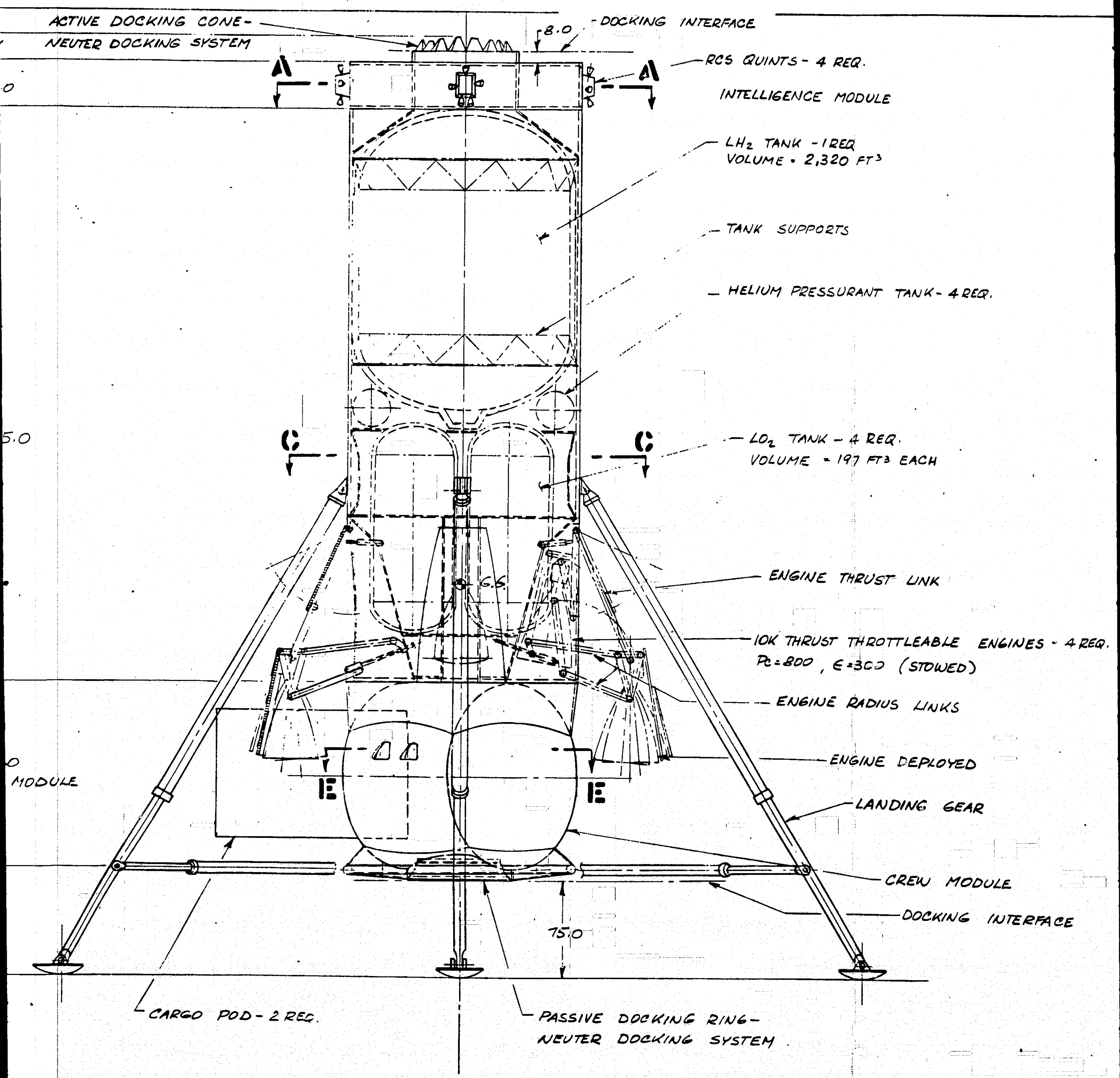


Figure 2-15. Horizontal

FOLDOUT FRAME 2

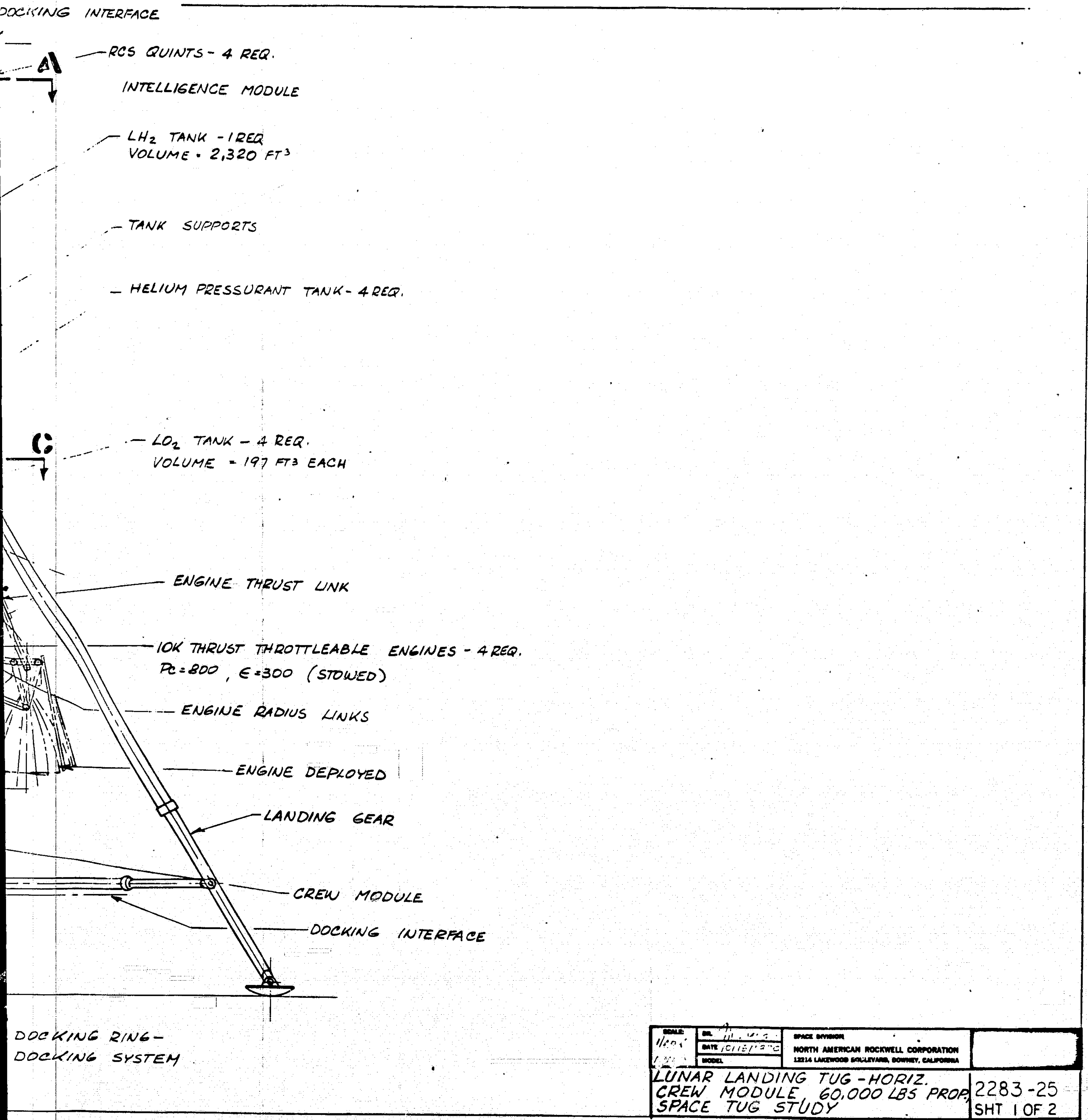
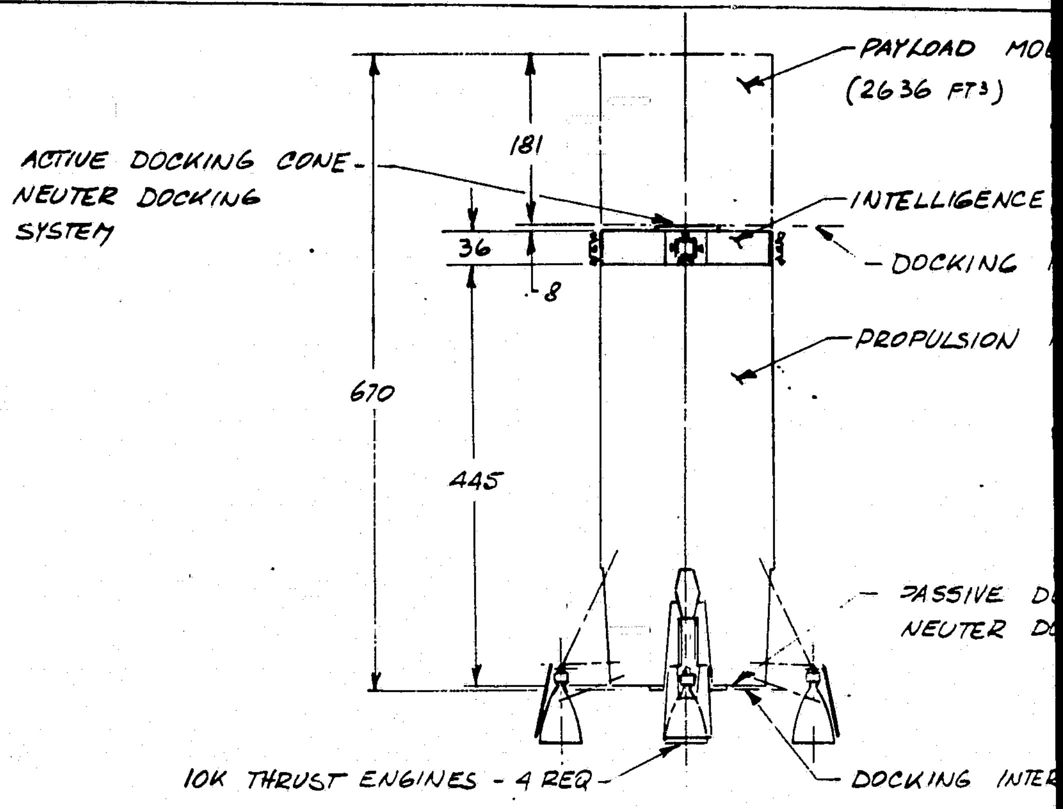
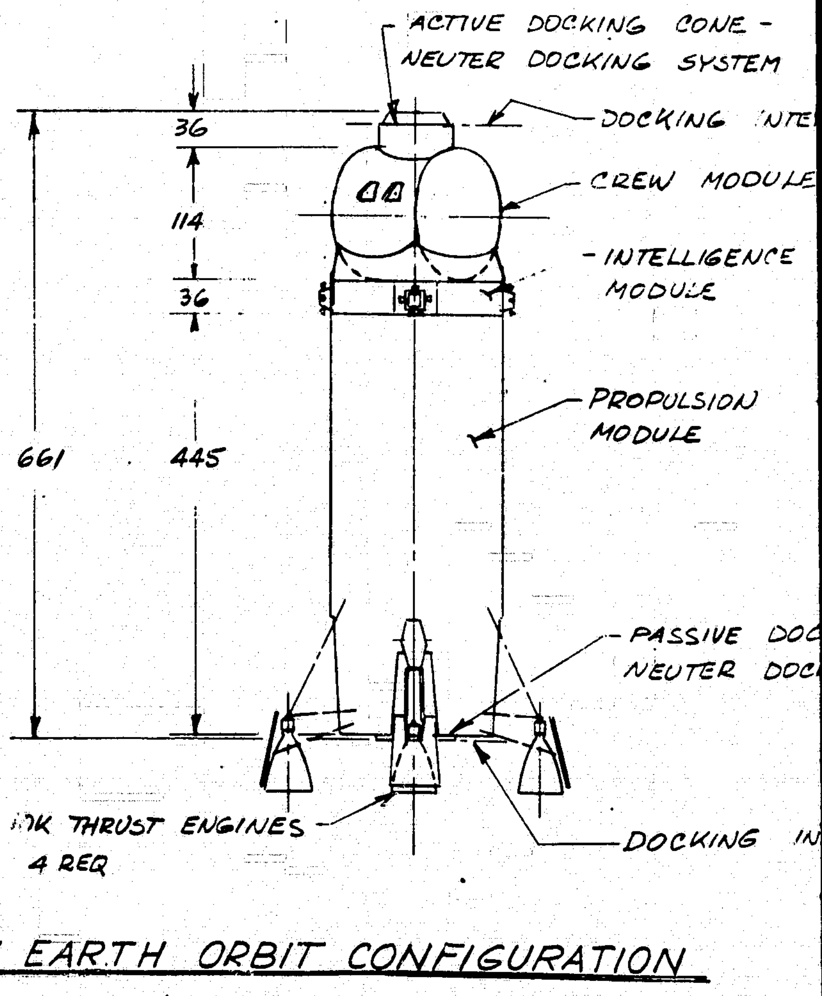


Figure 2-15. Horizontal Crew Module - 60K Pound Propellant Capacity Lunar Landing Tug (Sheet 1 of 2)



UNMANNED LEO AND GEOSYNCHRONOUS CONFIGURA

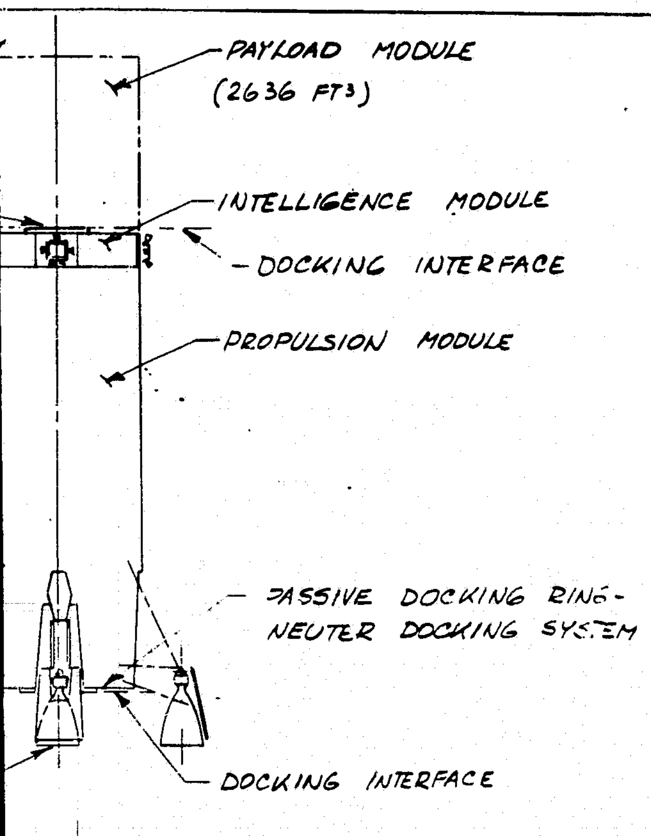
SCALE: 1/100



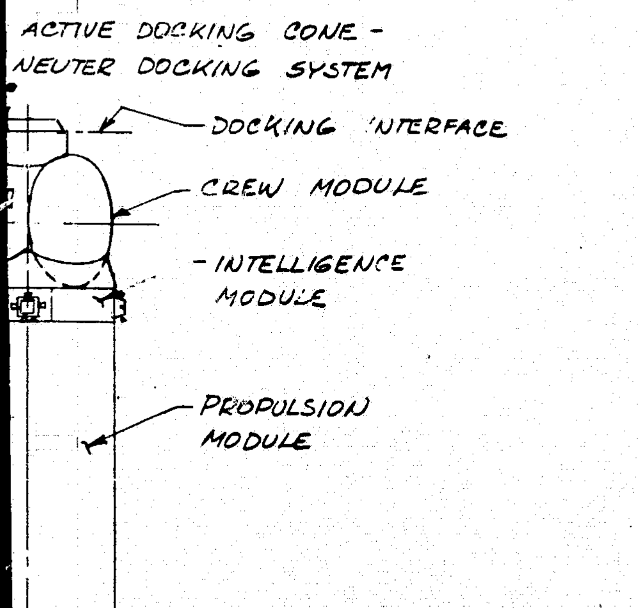
MANNED LOW EARTH ORBIT CONFIGURATION

SCALE: 1/100

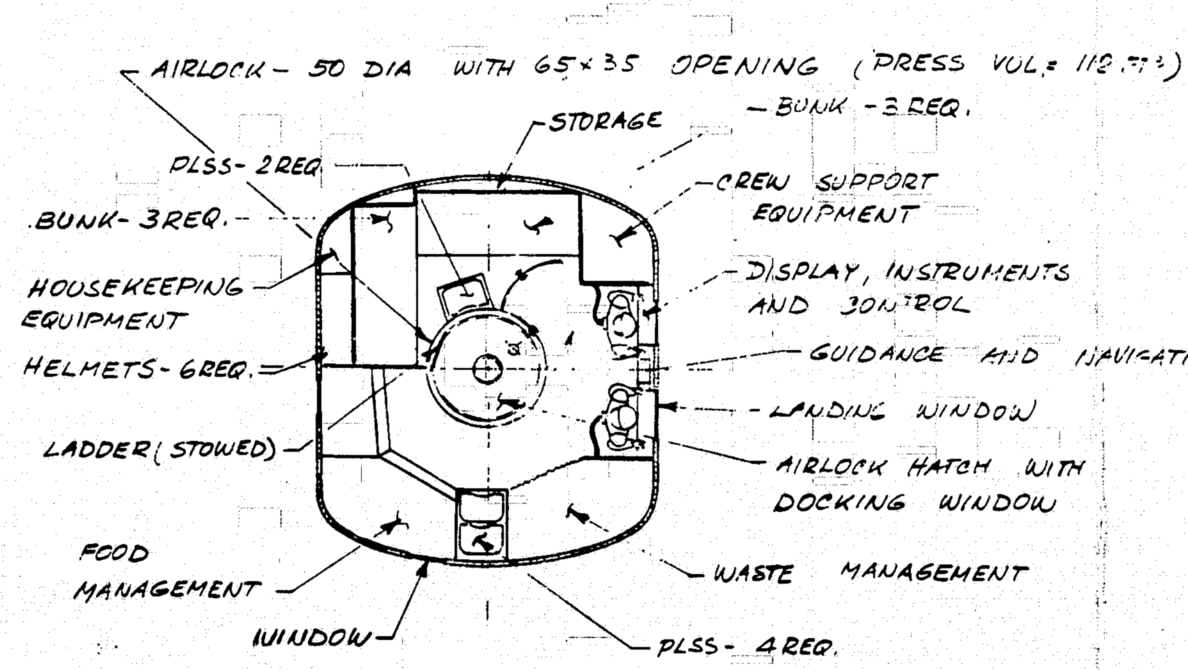
FOLDOUT FRAME



RONOUS CONFIGURATION



NFIGURATION

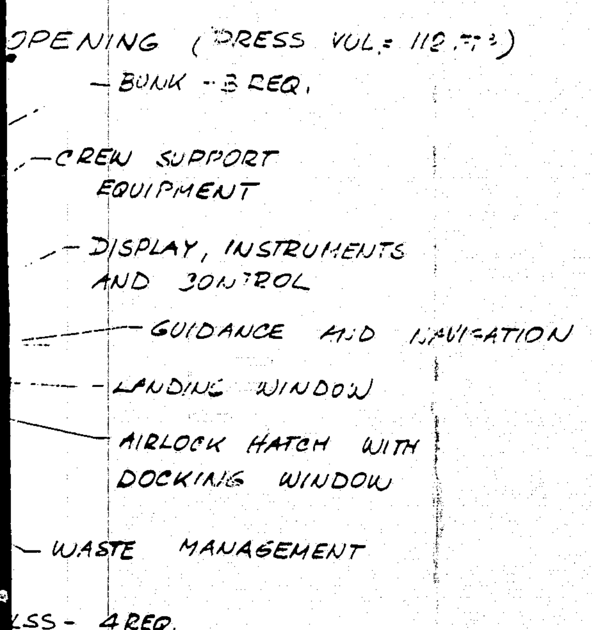


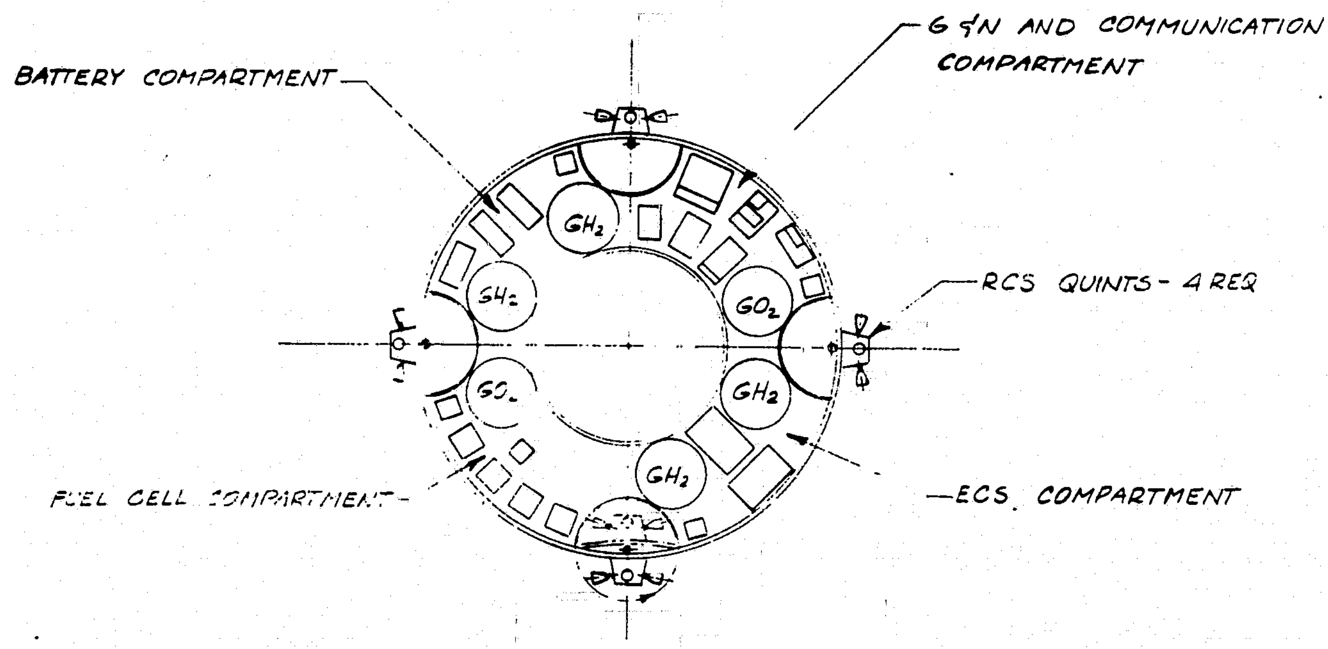
SECTION E - E
ROTATED 45°

Figure 2-15.

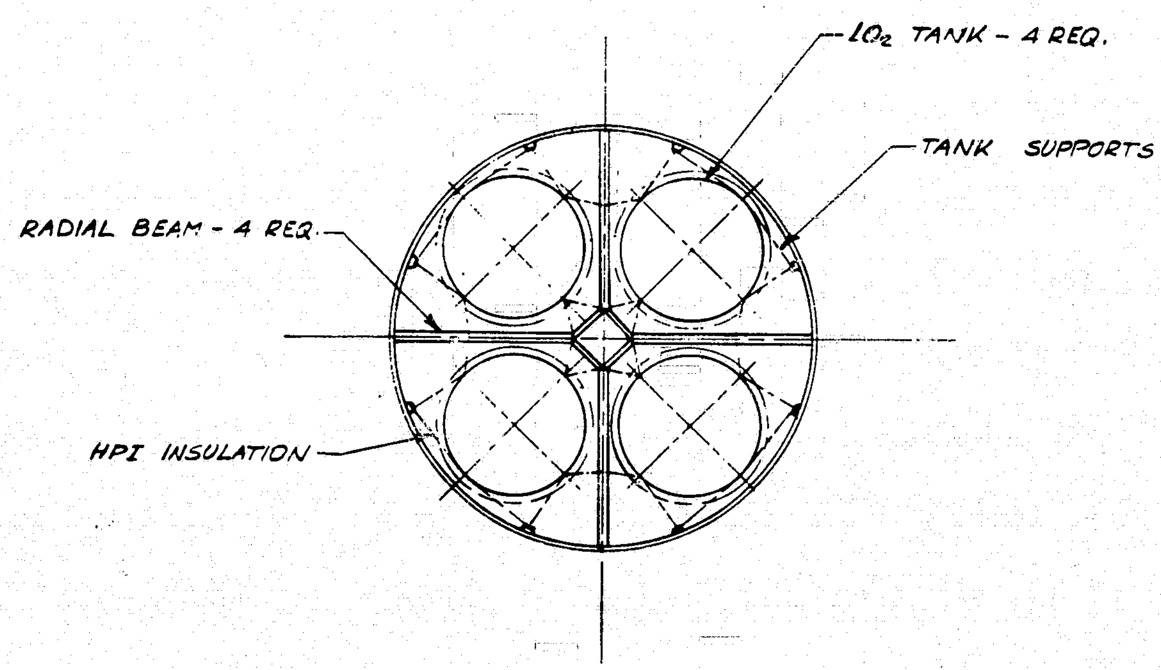
2

FOLDOUT FRAME





SECTION A-A



SECTION C-C

2283-25
SHT 2 OF 2

Figure 2-15. Horizontal Crew Module - 60K Pound Propellant Capacity Lunar Landing Tug (Sheet 2 of 2)

FOLDOUT FRAME 3

800-psi (552 N/cm^2) chamber pressure, 300-to-1 expansion ratio nozzle, and a specific impulse of 463 (4550 N-S/kilogram) seconds. The engines are canted inboard 6-1/2 degrees during stowage to prevent violating the 15-foot (4.57-meter) maximum diameter requirement for the EOS cargo compartment. The launch configuration of the basic vehicle has a length of 691 inches (17.6 meters), which is compatible with the EOS bay.

The basic vehicle comprises an active neuter docking cone at the top, the CM, the IM, a 3300-cubic foot, (93.4 cubic meter) H_2 tank, four 275-cubic foot (7.8 cubic meter) O_2 tanks, engine thrust structure, two hemipod cargo modules with a total volume of 1,000 cubic feet (28.3 cubic meters), four 10,000-pound-thrust (44,482-N) engines, and a passive docking cone on the bottom. The landing gear is a kit that is installed in space. The basic vehicle structure forms a cruciform about the lower end of the engines and supports the horizontal landing gear members and the propellant transfer system interface. The transfer system consists of two drogues—one for liquid hydrogen and one for liquid oxygen—and a quick disconnect for helium. This equipment is enclosed in high-performance insulation to minimize heat leaks.

For access to the transfer equipment by the orbiting propellant depot, a hatch on the tug, which covers the drogues, is opened, permitting the probes to be attached. Once the connections are completed, the transfer equipment is completely encased in the high-performance insulation, holding the boiloff losses to a minimum. The second concept shown is the same as the first except the usable propellant capacity is 60,000 pounds (27,215 kilograms) instead of 80,000 pounds, (36,287 kilograms) the liquid hydrogen tank capacity is 2,320 cubic feet (67.7 cubic meters) and the single liquid oxygen tank is 785 cubic feet (22.2 cubic meters). The vehicle for the launch configuration is 499 inches (12.7 meters) long, which is compatible with the EOS cargo bay.

Lunar-Optimized Concepts

Shown on Figure 2-14 is a manned single-stage lunar lander with a capacity of 80,000 pounds (36,287 kilograms) of usable propellant. The propulsion section has four translating (swing-out) engines stowed within the 15-foot-diameter (4.57-meter) moldline of the vehicle. The engines are supported on four-bar linkage systems stabilized by over-center folding thrust braces. The moldline shrouding is attached to the engines and the thrust braces. The engine is suspended in a mounting frame on the linkage by a gimbal, and the thrust vector control actuators are attached to the frame and the engine, permitting the engine thrust vector to vary ±5 degrees in two axes. As the engines are deployed from their stowed position, they move outboard a one-nozzle diameter beyond the outer moldline and down approximately 80 inches (2.0 meters) to within 28 inches (0.71 meters) of

the bottom plane of the crew compartment. With this engine deployment system, analysis of the crew module revealed no excessive plume impingement temperatures. In the launch configuration, the vehicle is 675 inches (17.1 meters) long.

The basic vehicle comprises of an active neuter docking cone at the top mounted in the center of the IM, a 3300-cubic feet (93.4 cubic meters) LH_2 tank, four 32-inch-diameter (0.81-meter) helium tanks with 10 cubic feet (0.3 cubic meters) per tank, four 275-cubic feet (7.8-cubic meters) LOX tanks, two hemipod cargo module with a total volume of 1000 cubic feet (28.3 cubic meters) four 10,000-pound-thrust (44,482-N) engines with an 800-psi (552 k/cm^2) chamber pressure, 300-to-1 expansion ratio nozzle, and specific impulse of 464 pound-seconds (4550 N-S/kilograms), and a passive, pressurized docking cone on the bottom for crew access to the lunar surface. The landing gear is a kit that is installed in space. The propellant transfer system on the tug consists of two drogues and is identical to that described on Figure 2-13.

The second concept is a manned low earth orbiting tug configuration with the crew compartment mounted on the upper end of the vehicle to provide better visibility. The vehicle has the same tanks and propulsion system as the first concept, and the stowage length is 703 inches (17.9 meters).

The last concept shown is a manned lunar landing tug. It is identical to the first concept except the usable propellant has been reduced to 60,000 pounds (27,215 kilograms). The stowage length of this concept is 584 inches (14.8 meters).

Concept 2 With Horizontal CM

The concept in Figure 2-15 is a manned lunar lander that has accommodated an aft-located horizontal CM. The basic configuration is similar to that in Figure 2-14. Swing-out or deployable main engines are used, and a single LH_2 tank and four LOX tanks are accommodated. The only difference between this concept and that shown earlier is the length of the propellant tanks and the integration of a horizontal CM.

The propellant capacity is 60,000 pounds (27,215 kilograms). The horizontal cylinder CM of 12 feet (3.66 meters) in diameter appears reasonable and used a passive docking ring at the aft end. The greatest problem encountered with this concept is the integration of the horizontal CM into the basic vehicle. This CM is higher (longer in the stack) than a vertical cylinder and offers less floor area. This CM concept also has integration problems when the CM is mounted at the forward end of the vehicle. In this instance a neuter docking system is required on the forward end. There does not appear to be a straightforward approach to integrating

the CM with the basic vehicle. Consequently, it was not felt that this concept should receive further consideration. It does not appear to offer sufficient advantages over a vertical cylindrical CM approach.

Concept 4 - NASA Mode B Lander

Concept 4 of Table 2-4 is the only other design configuration based on a lunar landing mission. This concept was sized for 41,000 pounds (18,597 kilograms) of propellant and operated as a single-stage lunar lander. It performed the Mode B lunar lander mission, which lands 20,000 pounds (9072 kilograms) of payload on the lunar surface and returns 10,000 pounds (4536 kilograms) to orbit. A single LH_2 tank with an internal volume of 1515 cubic feet (42.9 cubic meters) is used along with a single LOX tank with an internal volume of 533 cubic feet (15.1 cubic meters). The structural concept and refueling approach are as described.

The landing gear is an in-space attached kit. A separable IM has been incorporated as has a forward-mounted CM. Overall length of the configuration is 500 inches (12.7 meters). For a geosynchronous mission, two PM's with IM's are used in tandem. Both vehicles are then recovered. This concept converts to a shorter lunar lander vehicle than the previous vehicles but it can only return the CM, approximately 10,000 pounds (4536 kilograms) to orbit. It has the capability of landing a total payload of 20,000 pounds (9072 kilograms) made up of about 10,000 pounds (4536 kilograms) for a CM and 10,000 pounds (4536 kilograms) of payload. To operate in landing Mode A, this configuration would require a tank set with about 20,000 pounds (9072 kilograms) of additional propellant.

Multiple-Stage/Tank Concepts

Three concepts were configured (optmized as to propellant capacity) for a geosynchronous mission with a payload of 10,000 pounds (22046 kilograms) and maybe grouped. These are Concepts 3, 5 and 6. They have many common elements. In each a single LH_2 and single LOX tank has been used in either a PM or tank set. A separable IM is used with the same general arrangement as discussed. Refueling drogues are provided at the aft end of the vehicles and are insulated from the space environment to minimize propellant loss. Each PM accommodates four 10,000-pound-thrust (44,482-N) engines. The engines are supported by a girth ring assembly. The aft located passive docking ring is attached to this same ring through a conical skin-stringer structure. In this fashion the docking and engine loads are distributed to the outer shell (skin-stringer) of the vehicle.

In each configuration the LH_2 and LOX tanks are suspended from the outer shell through girth rings and tank support straps. Each tank is fully

insulated with layers of high-performance insulation. All of the LH_2 tanks are made up of cylindrical center sections with elliptical bulkheads of a 1.4-to-1 ratio. The LO_2 tanks are similarly constructued in the lower volume concepts and are made up of only bulkheads in the smaller volumes.

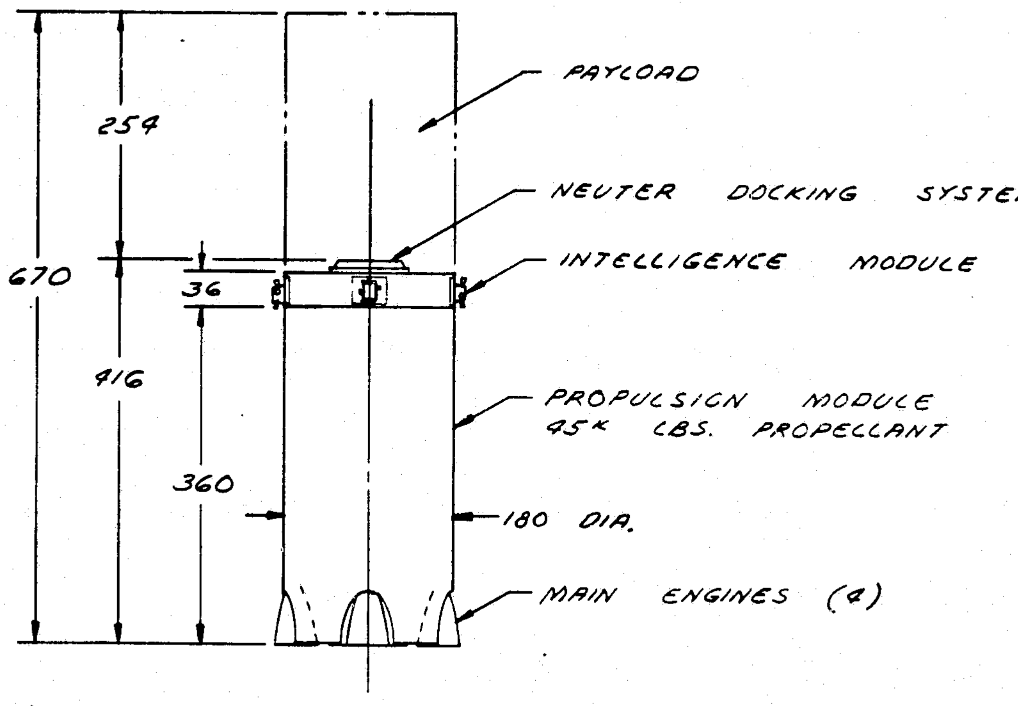
Concept 3 is shown in Figure 2-16. This concept uses a PM, tank set and an IM to perform the geosynchronous mission as a single stage. In this mode, the entire vehicle is recoverable. The propulsion module and tank set contain the same amount of usable propellant, 45,000 pounds (20,412 kilograms). The tank set is loosely described as a propulsion module without the engines and docking provisions. The intelligence module is located at the forward end of the tank set and accommodates an active neuter docking system for payload attachment.

The LH_2 tanks have an internal volume of 1710 cubic feet (48.2 cubic meters) each, and the LOX tanks have an internal volume of 590 cubic feet (16.7 cubic meters). The geosynchronous mission configuration is 750 inches (19 meters) long. The PM with IM operating as an unmanned low earth orbit vehicle is 416 inches (10.6 meters) long. The addition of a CM for manned operations near earth adds 109 inches (2.8 meters) to the configuration for a total length of 525 inches (13.3 meters). Two adaptations for lunar lander configurations are shown schematically in Figure 2-16. The first is a single-stage lander with a CM on top of the tank set. In this concept the tank set is required to deliver only 22,000 pounds (9979 kilograms) of propellant, which means it is only half full. This vehicle is extremely long, 859 inches (21.8 meters) for a lander and would require a large and heavy landing gear kit.

The other approach to the lunar lander is the parallel operation of two PM's with a CM and payload module between the two. This concept is only 396 inches (10 meters) long and requires each PM to deliver 33,500 pounds (15,195 kilograms) of propellant. The cargo module (CAM) could be configured from a CM shell with an egress tunnel in its center. The landing gear would be attached to the CAM and thus reduce scar weight on the PM and CM.

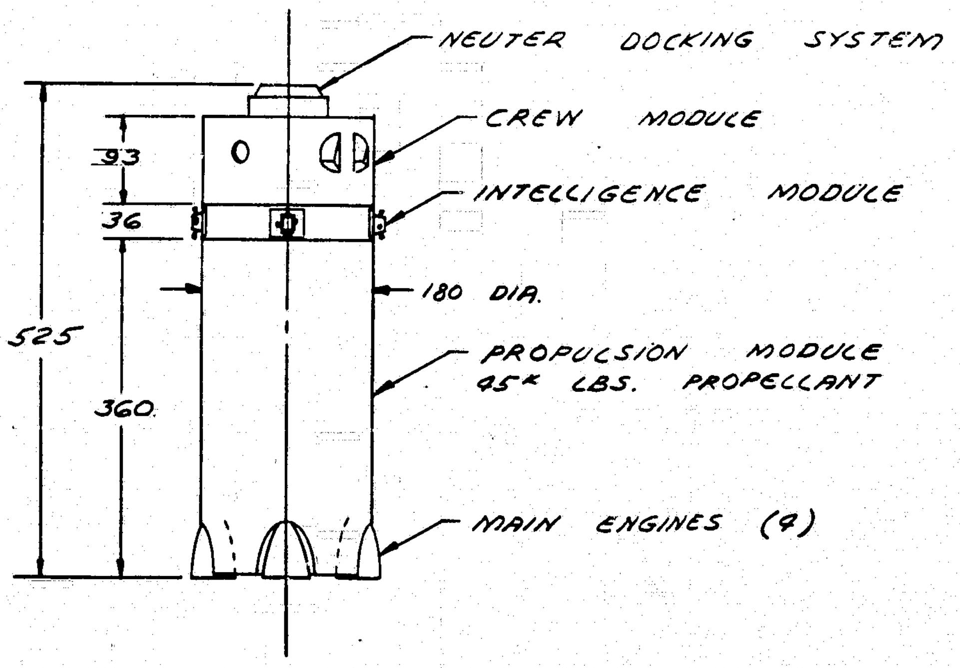
A third lunar lander was configured to accommodate the landing Mode B (Figure 2-17). This concept features a PM, an IM and a forward-located CM. The PM delivers 40,000 pounds (18,144 kilograms) of propellant. The CAM's are left on the lunar surface since Mode B missions do not return payload to orbit. Overall the configuration is 525 inches (13.3 meters), which is the same as the manned low earth orbit configuration. In each configuration the landing gear is attached as a kit.

The configuration shown on Figure 2-18 is Concept 5. The basic vehicle is configured as an optimized two-stage geosynchronous mission



UNMANNED LOW ORBIT CONFIGURATION

SCALE 1/100



MANNED LOW ORBIT CONFIGURATION

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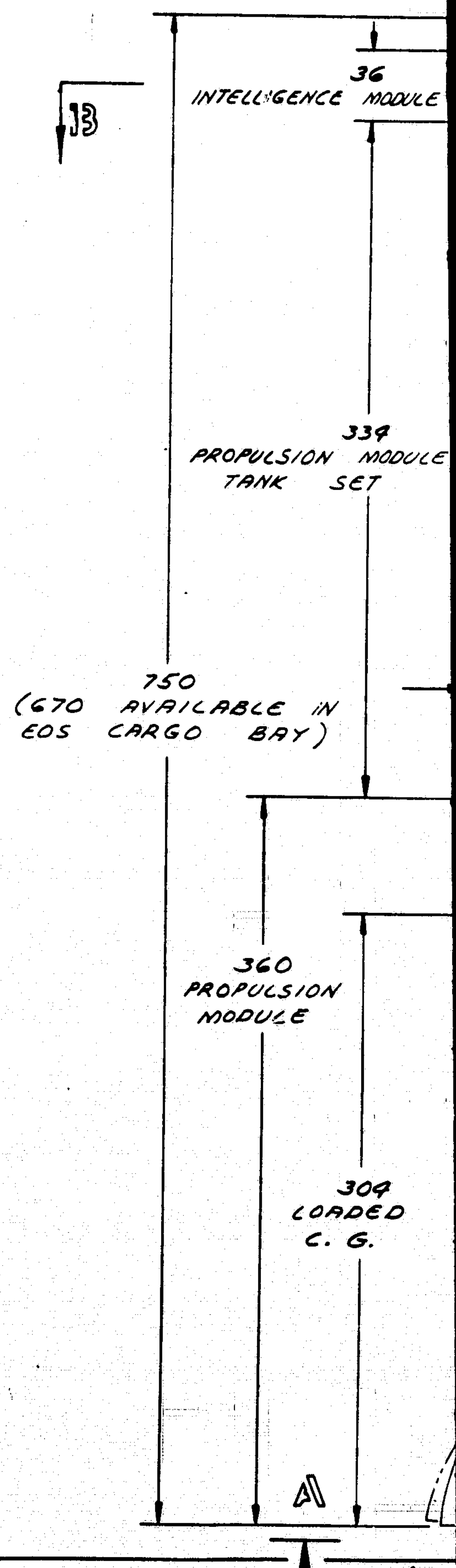

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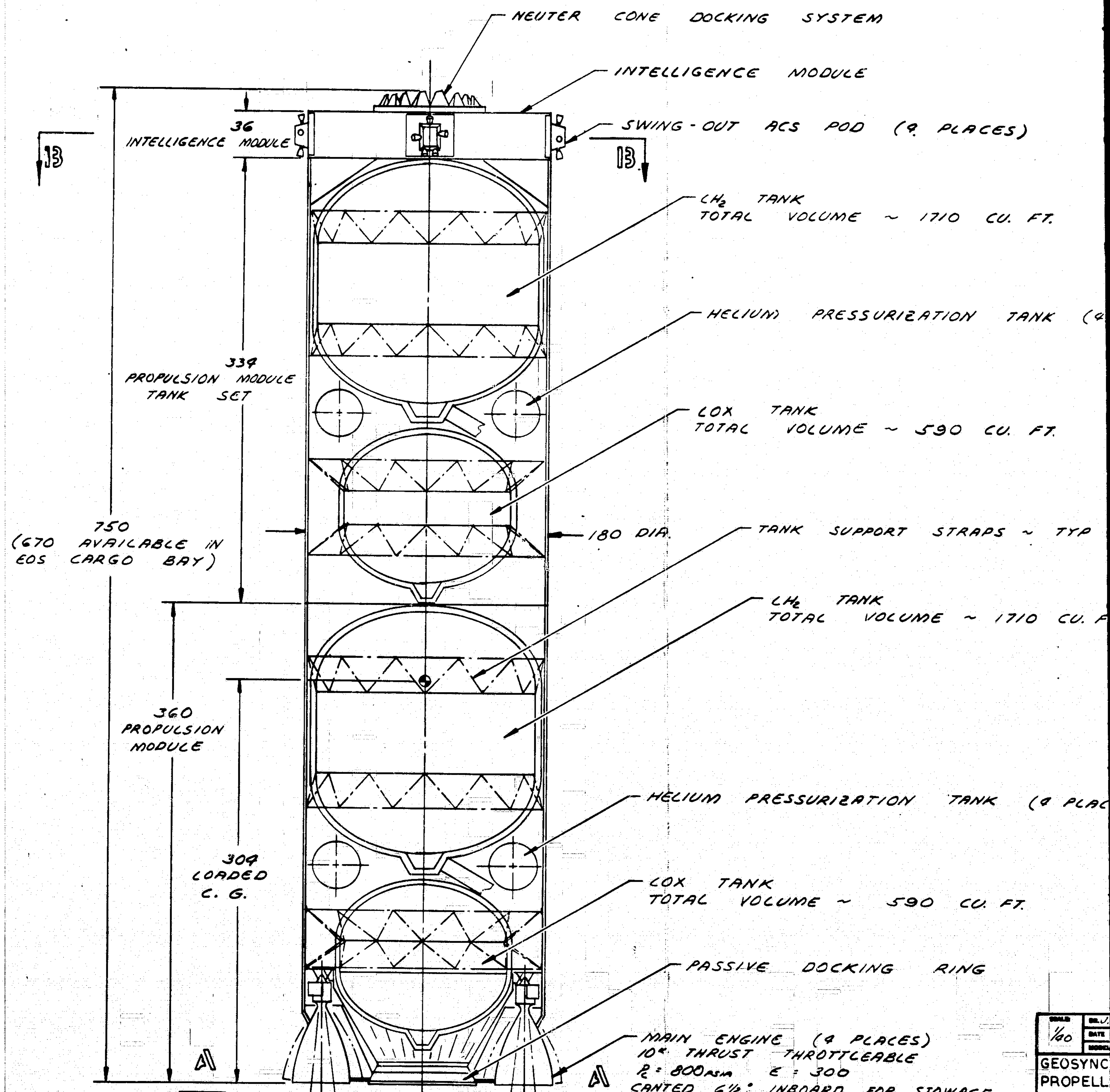

Figure 2-16. 45K Pound Propellant Pro
Single Stage Geosynchron

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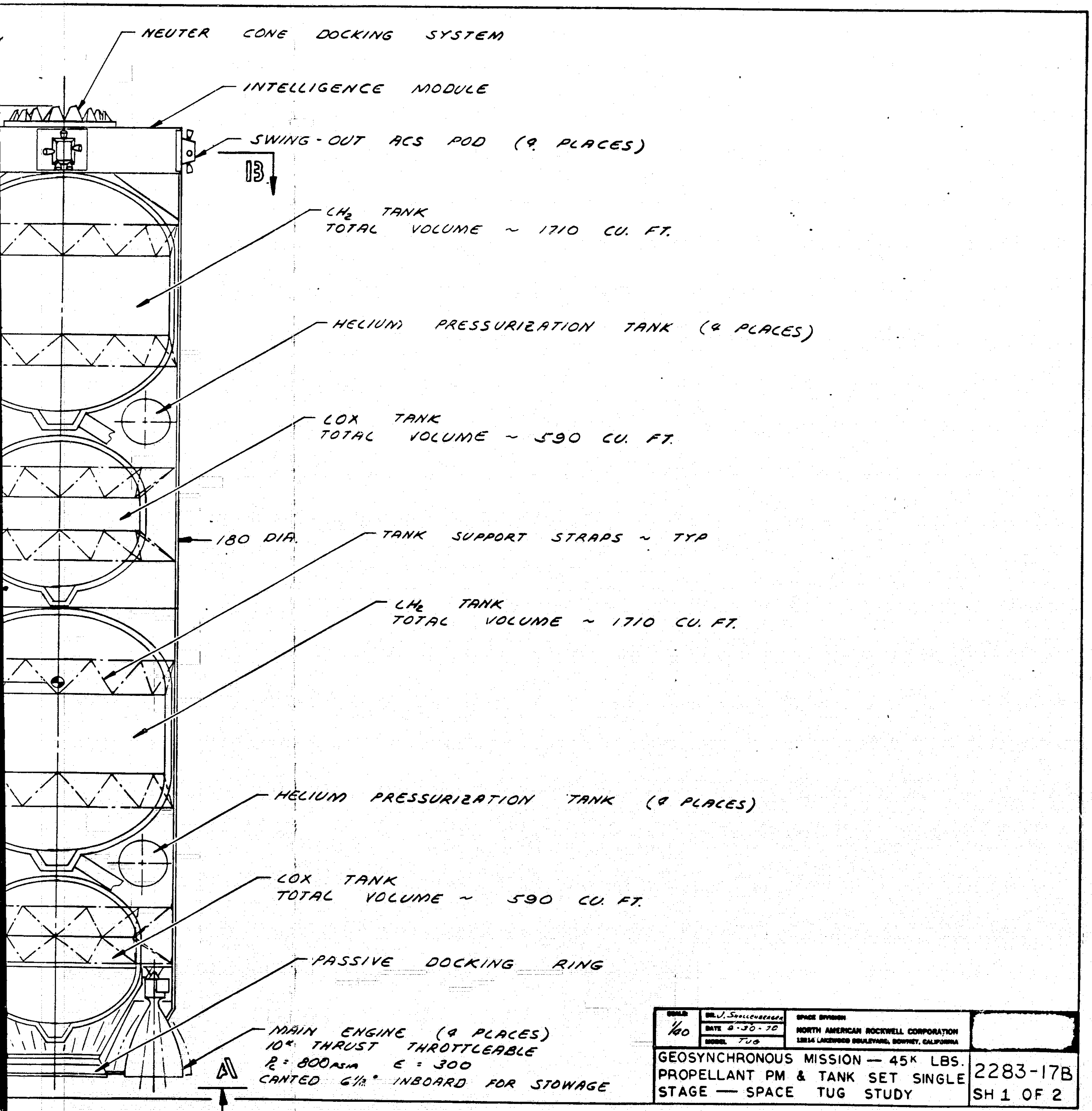
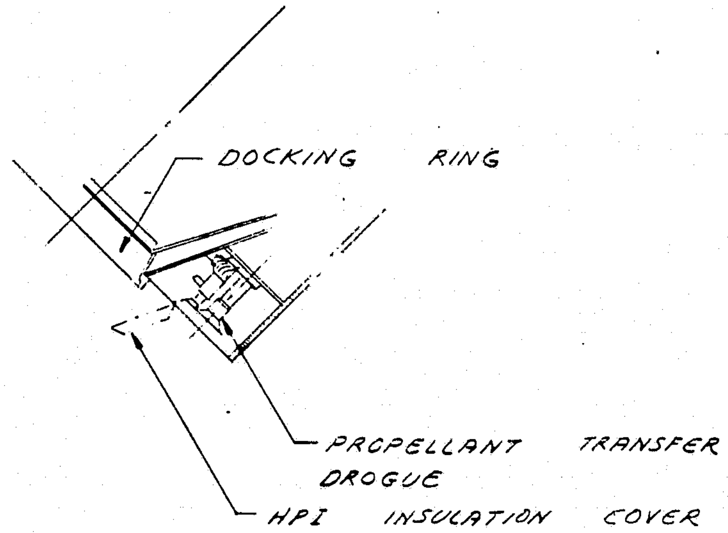
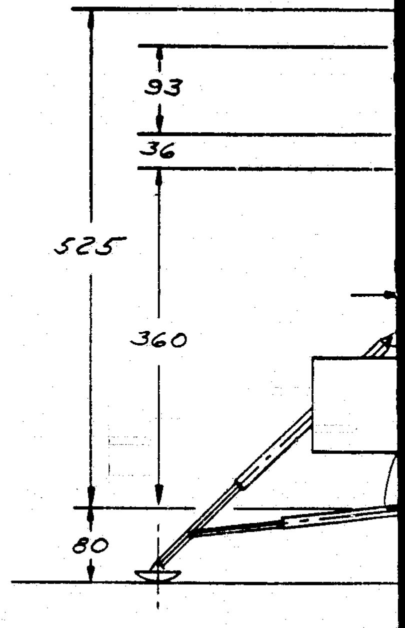


Figure 2-16. 45K Pound Propellant Propulsion Module and Tank Set Single Stage Geosynchronous Mission (Sheet 1 of 2)



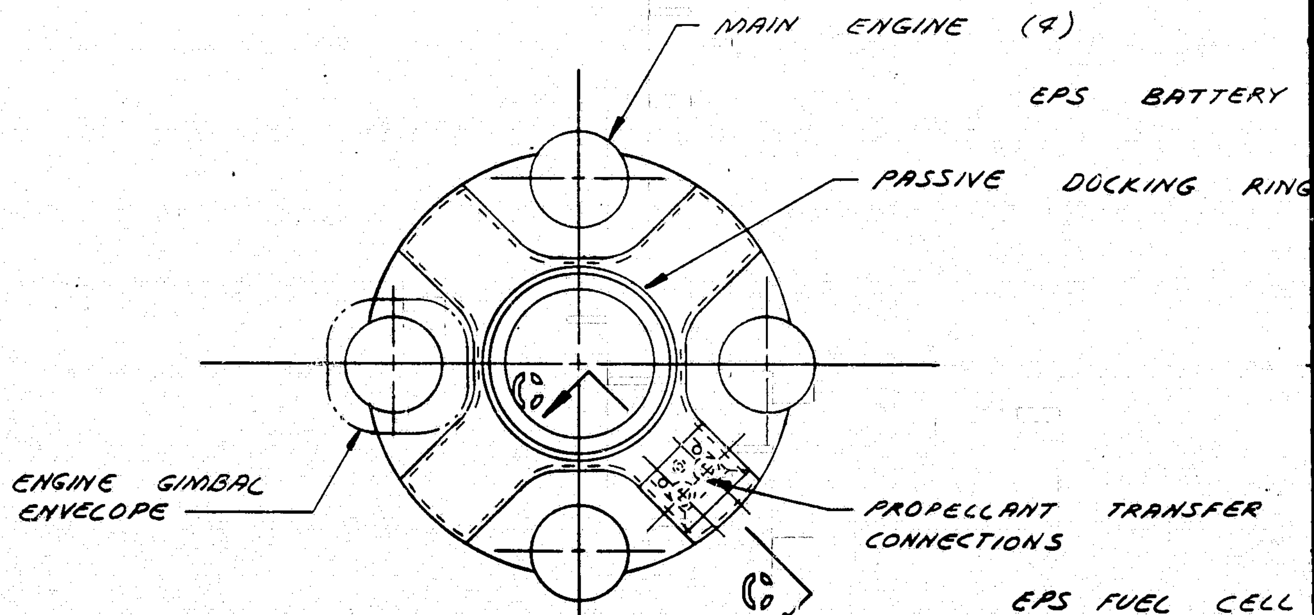
SECTION C – C



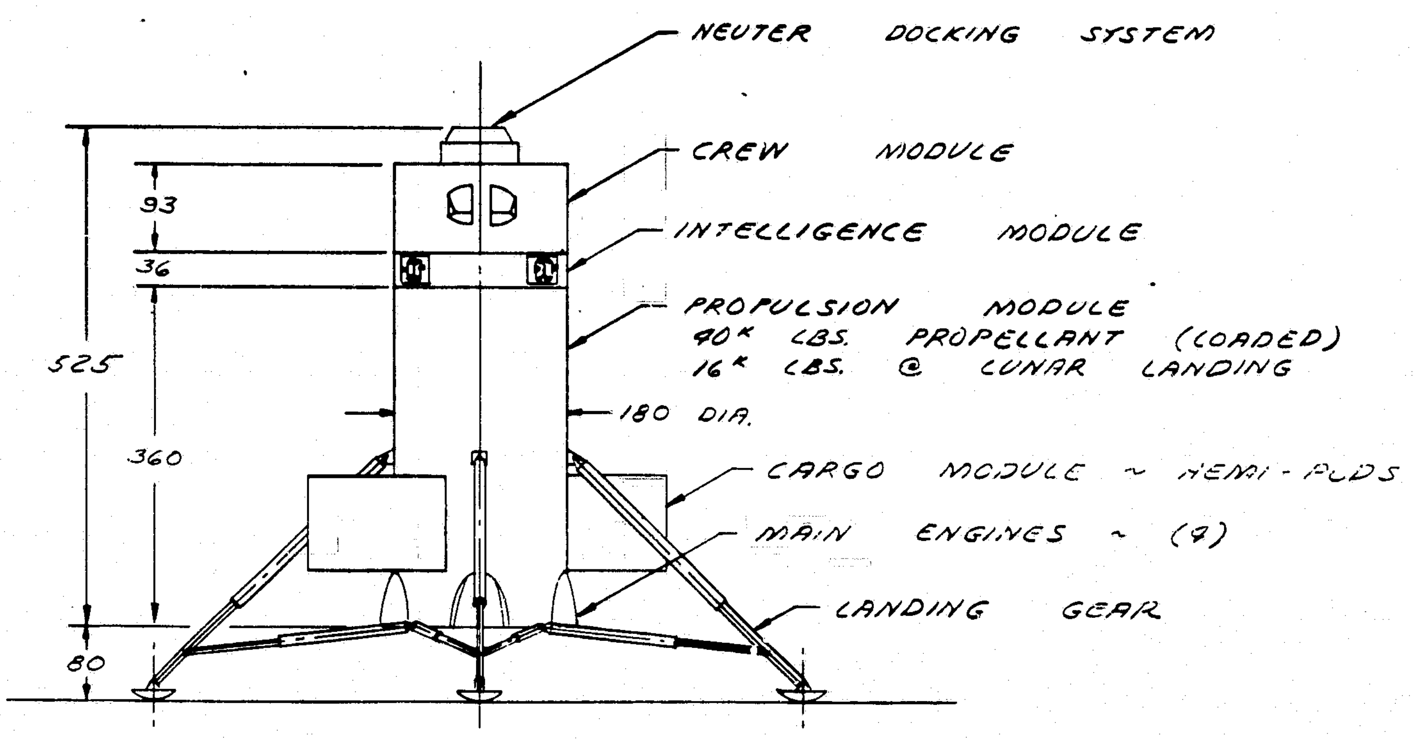
SINGLE STAGE

20K LB. PAYLO

SC



VIEW A – A



SINGLE STAGE LUNAR LANDER MODE B

20K LB. PAYLOAD DOWN — 10K LB. UP

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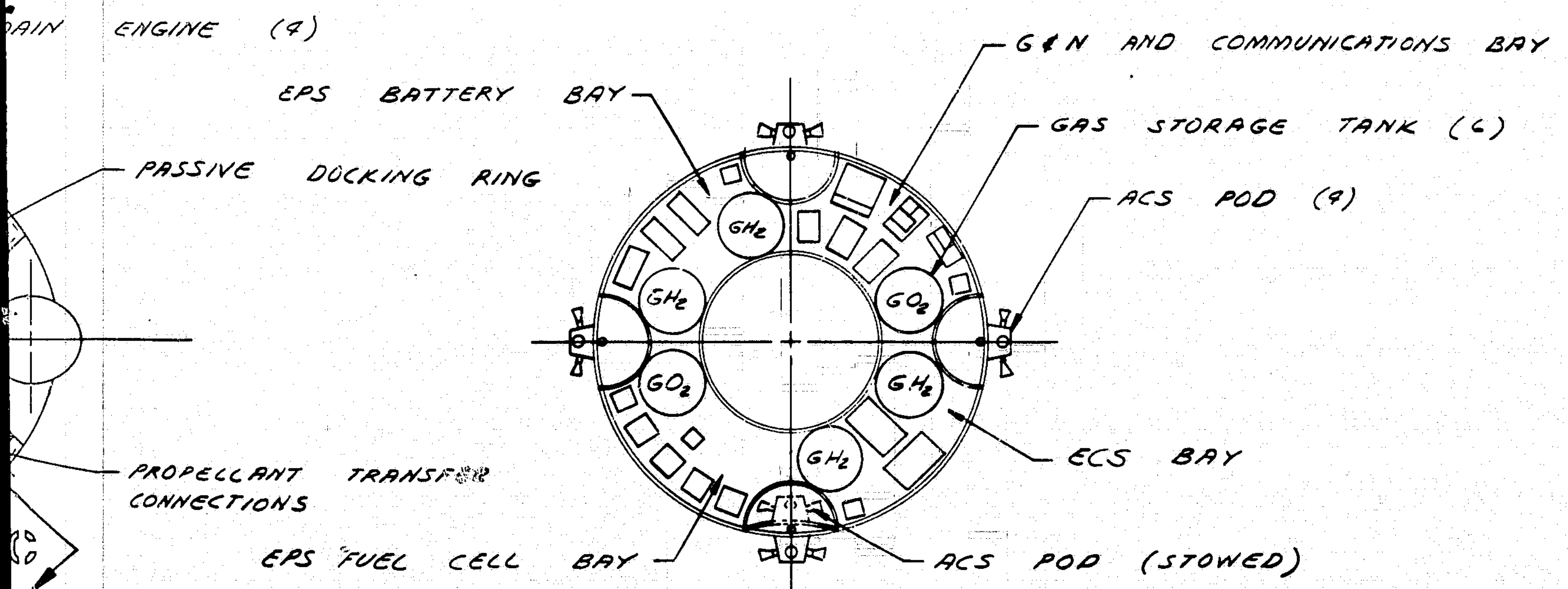
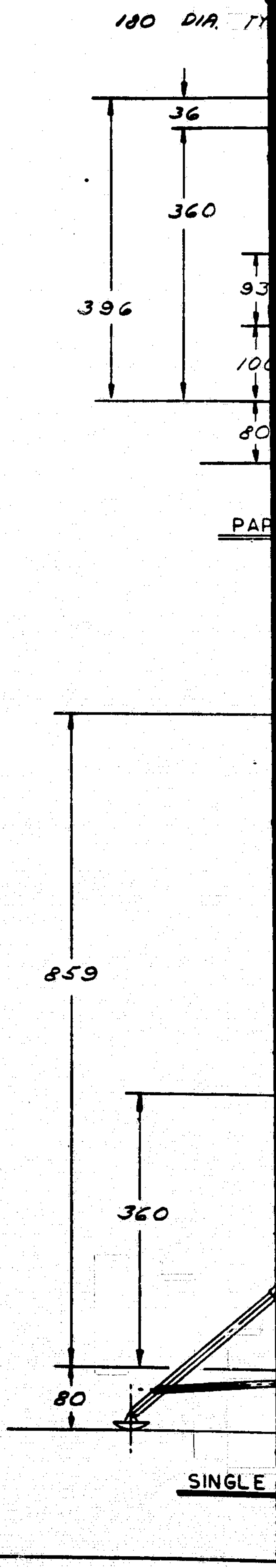
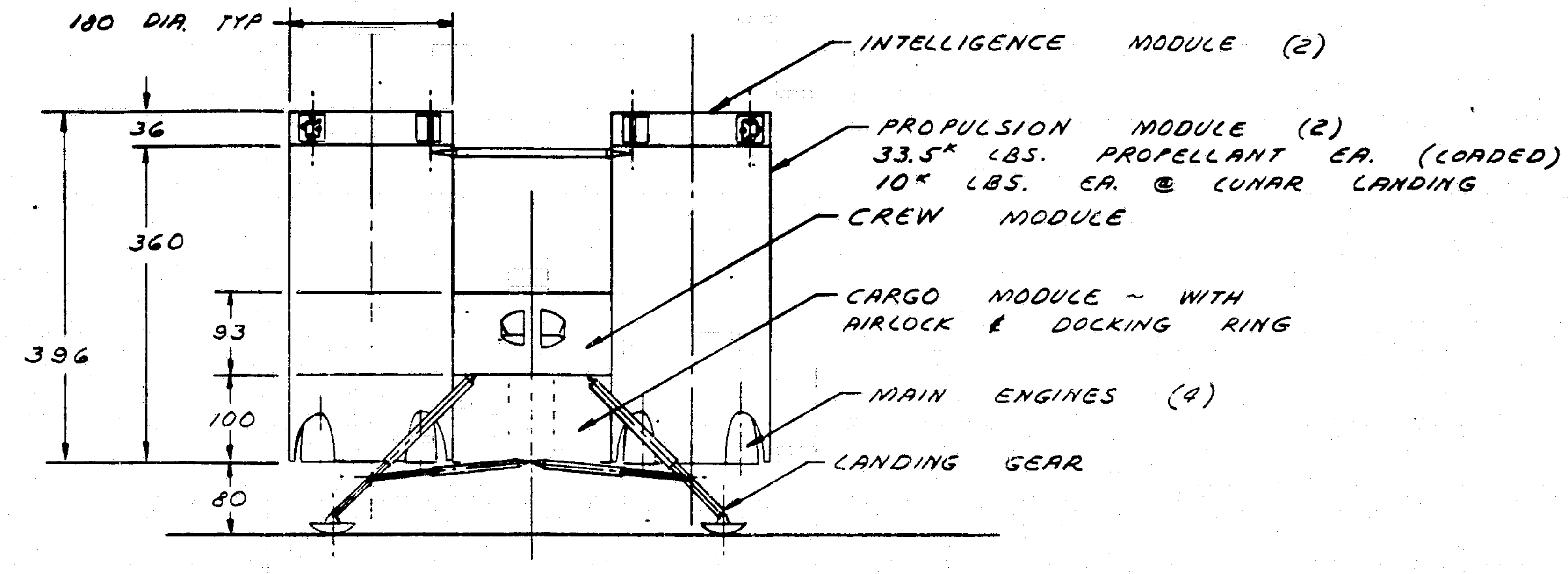


SECTION B - B

INTELLIGENCE MODULE





PARALLEL STAGES LUNAR LANDER MODE A

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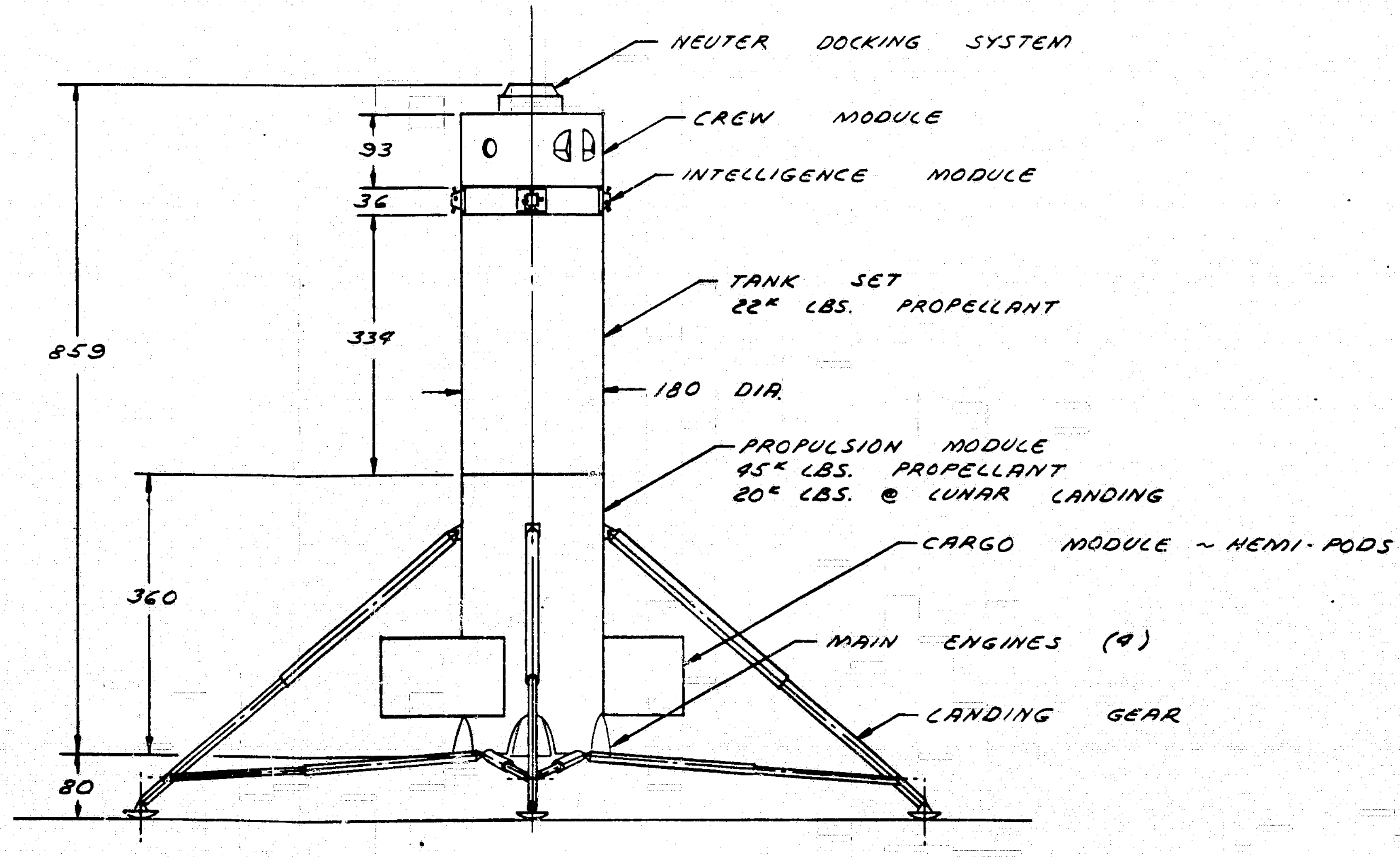

SINGLE STAGE LUNAR LANDER MODE A

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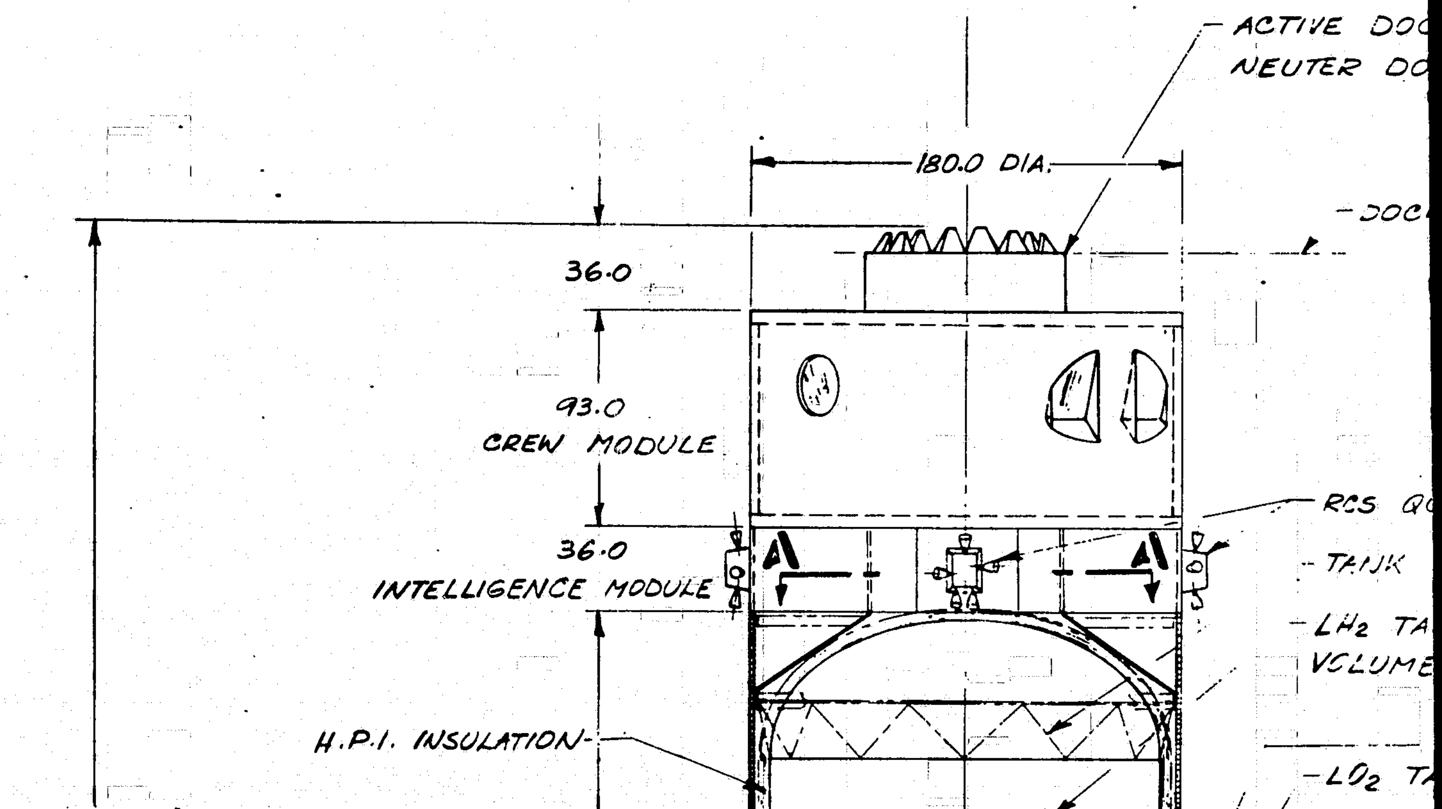
SHEET 2	OF 2

Figure 2-16. 45K Pound Propellant Propulsion Module and Tank Set Single Stage Geosynchronous Mission (Sheet 2 of 2)



LUNAR LANDING CONFIGURATION

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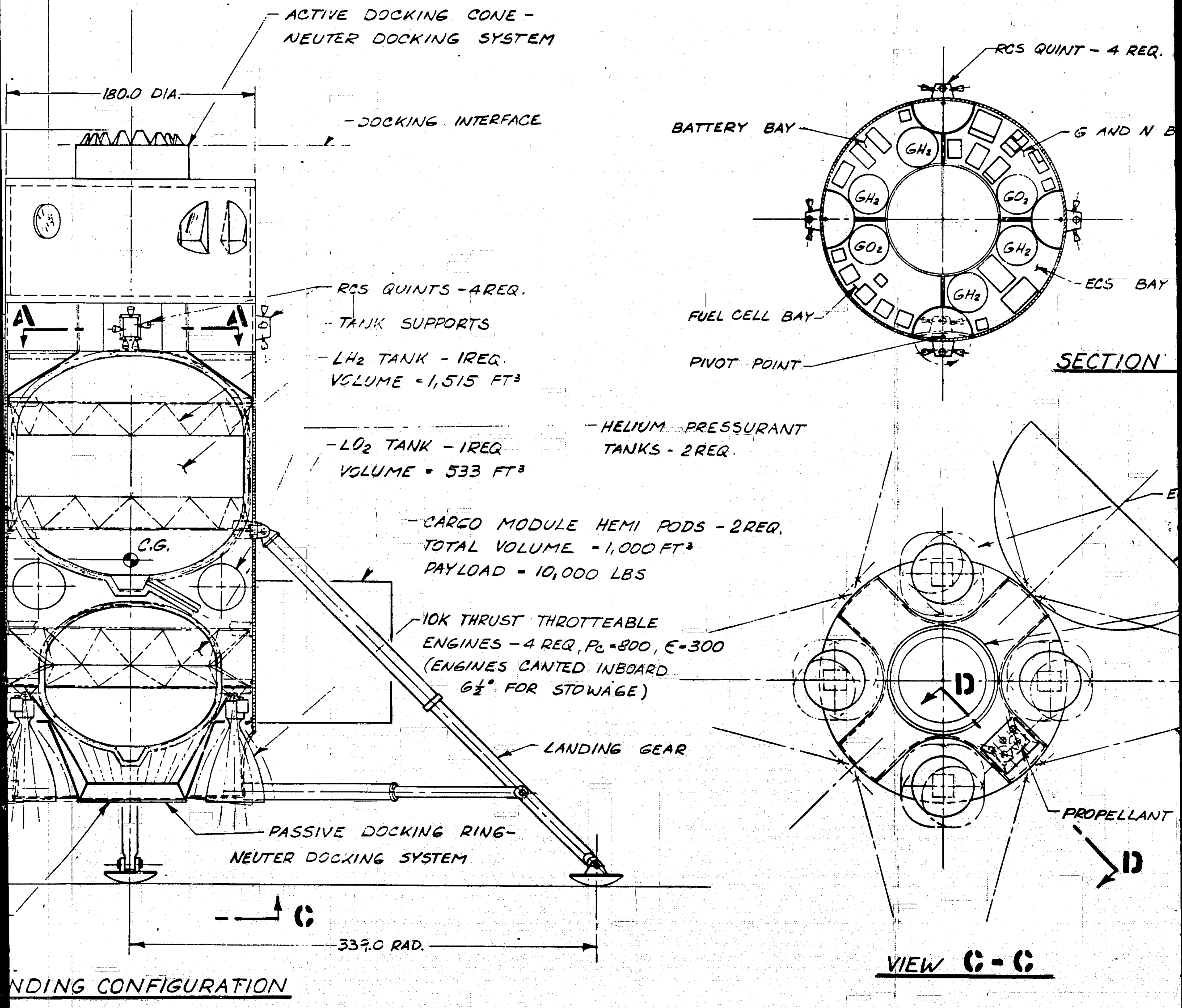

Figure 2-17. 15-Foot-Diameter,
(S

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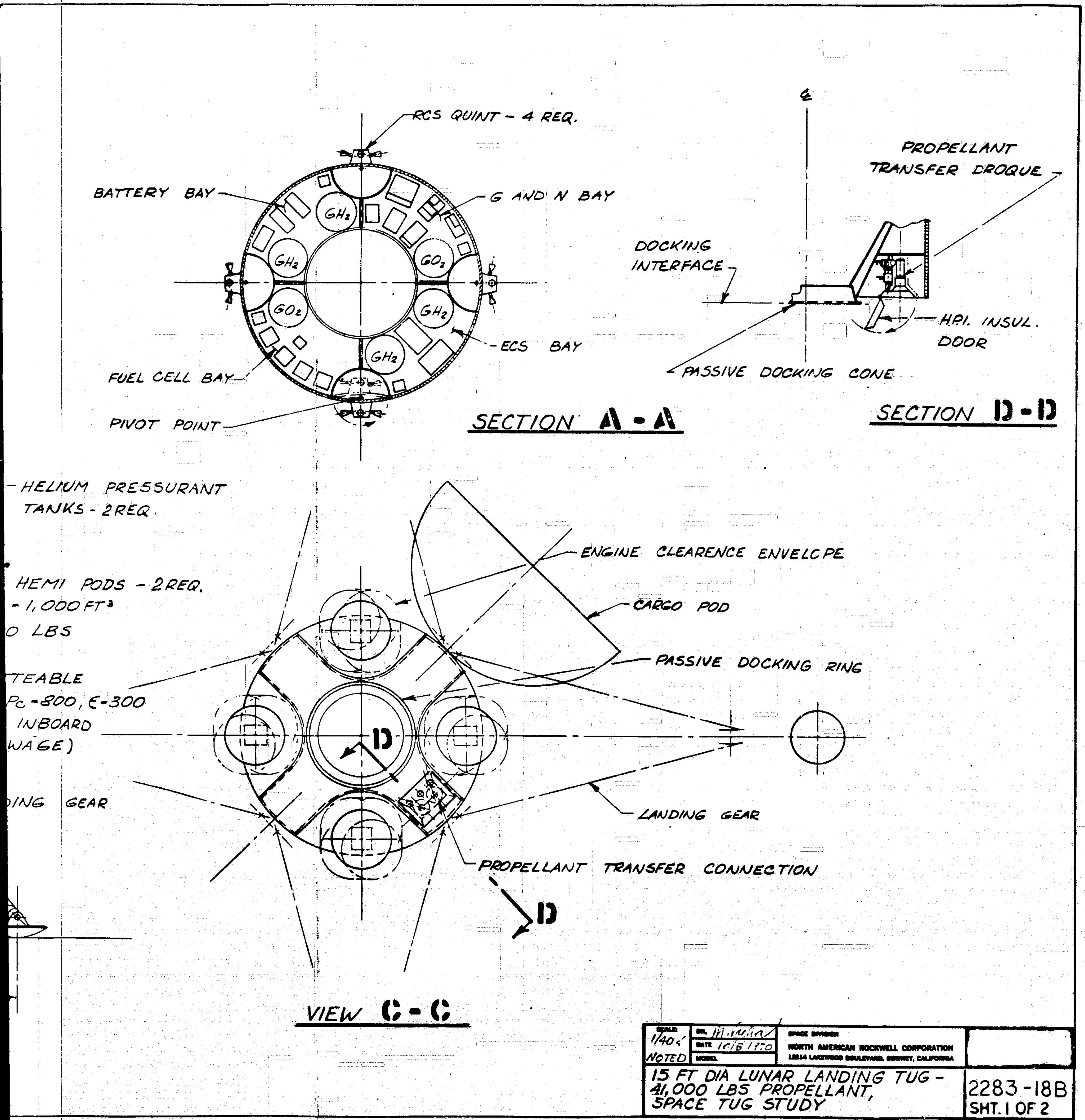

Figure 2-17. 15-Foot-Diameter, 41K Pound Propellant Lunar Landing Tug (Sheet 1 of 2)

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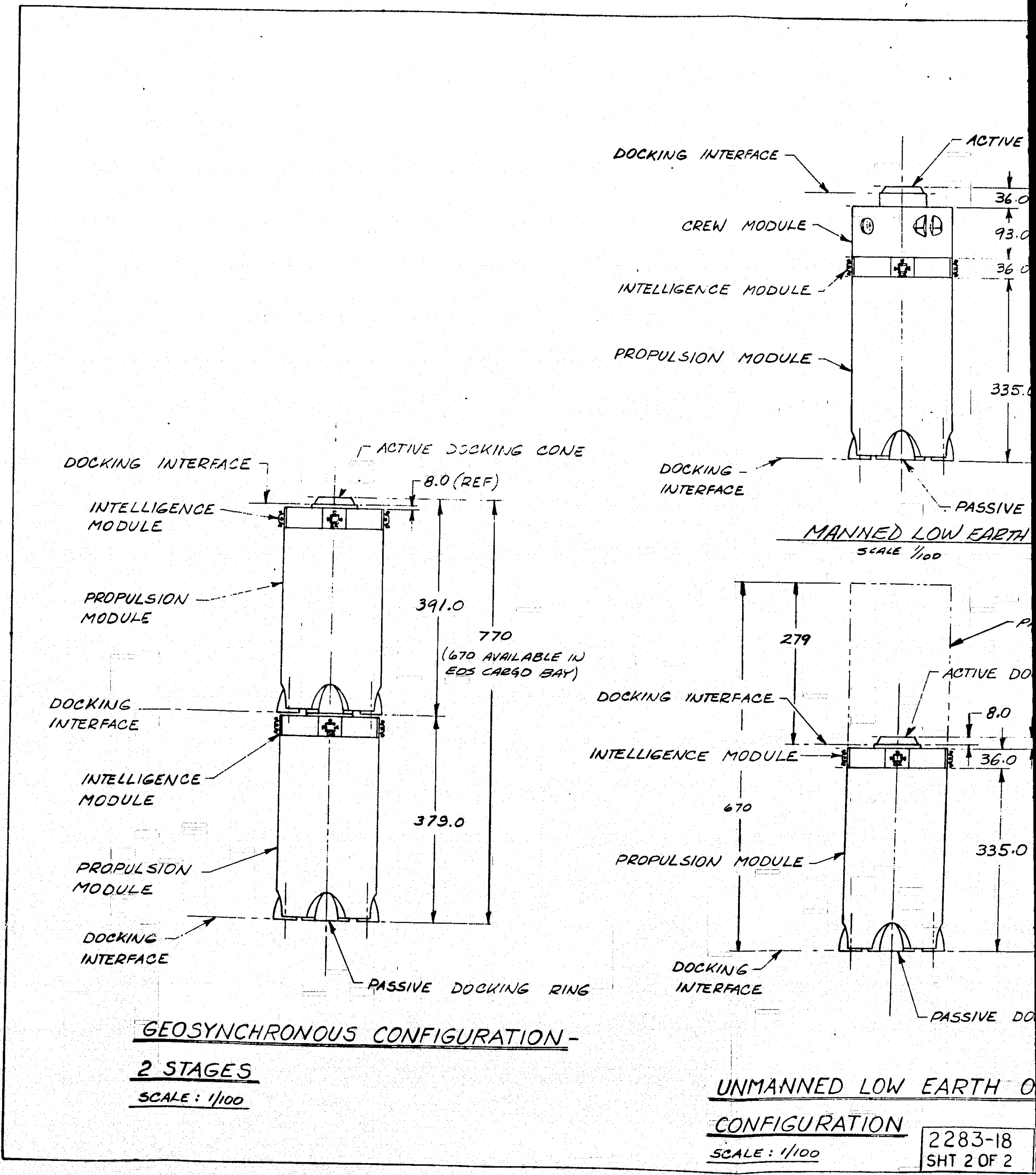

Figure 2-17. 15-Foot-Diameter, 41K Pound Propellant Lu
(Sheet 2 of 2)

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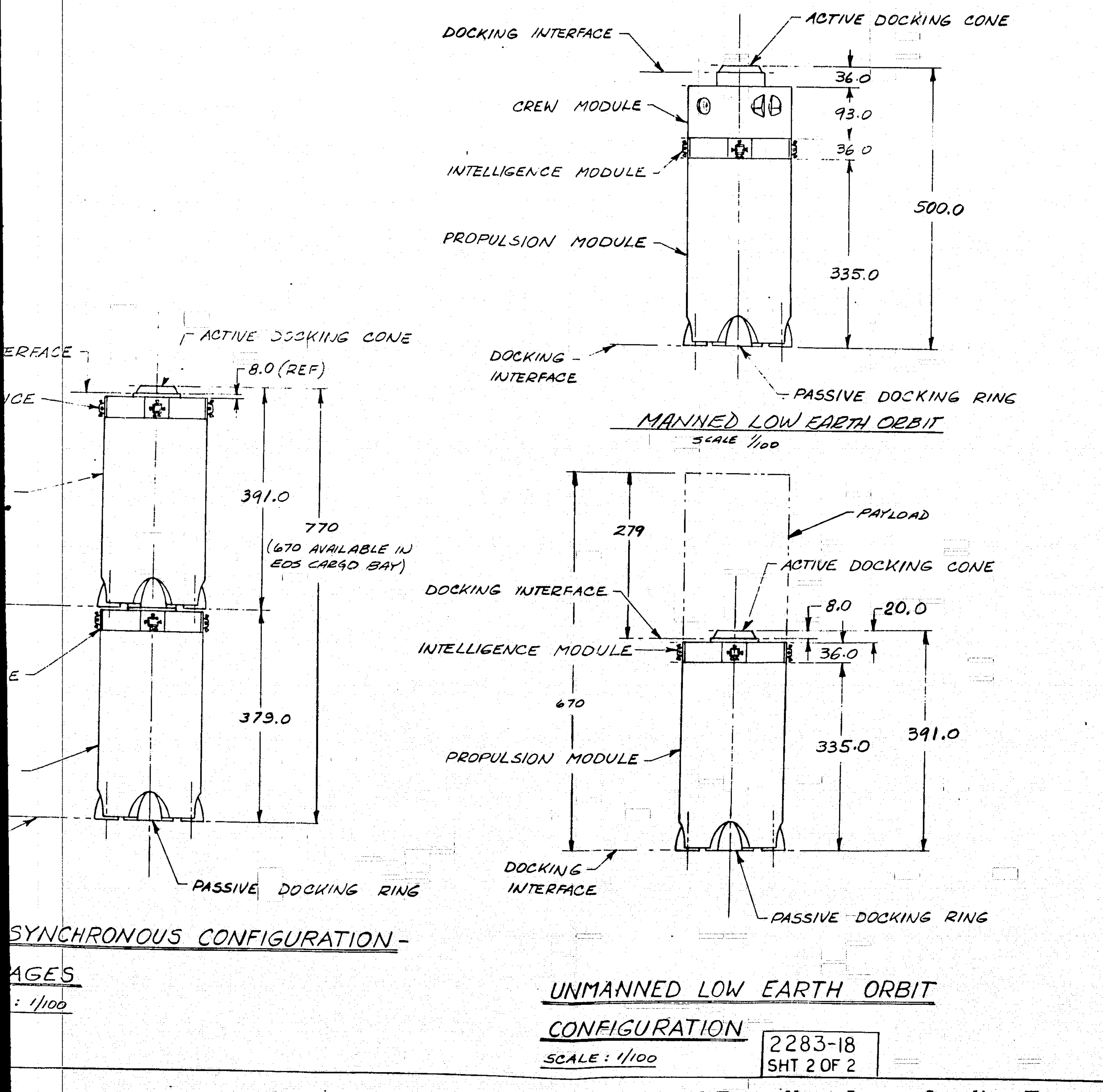
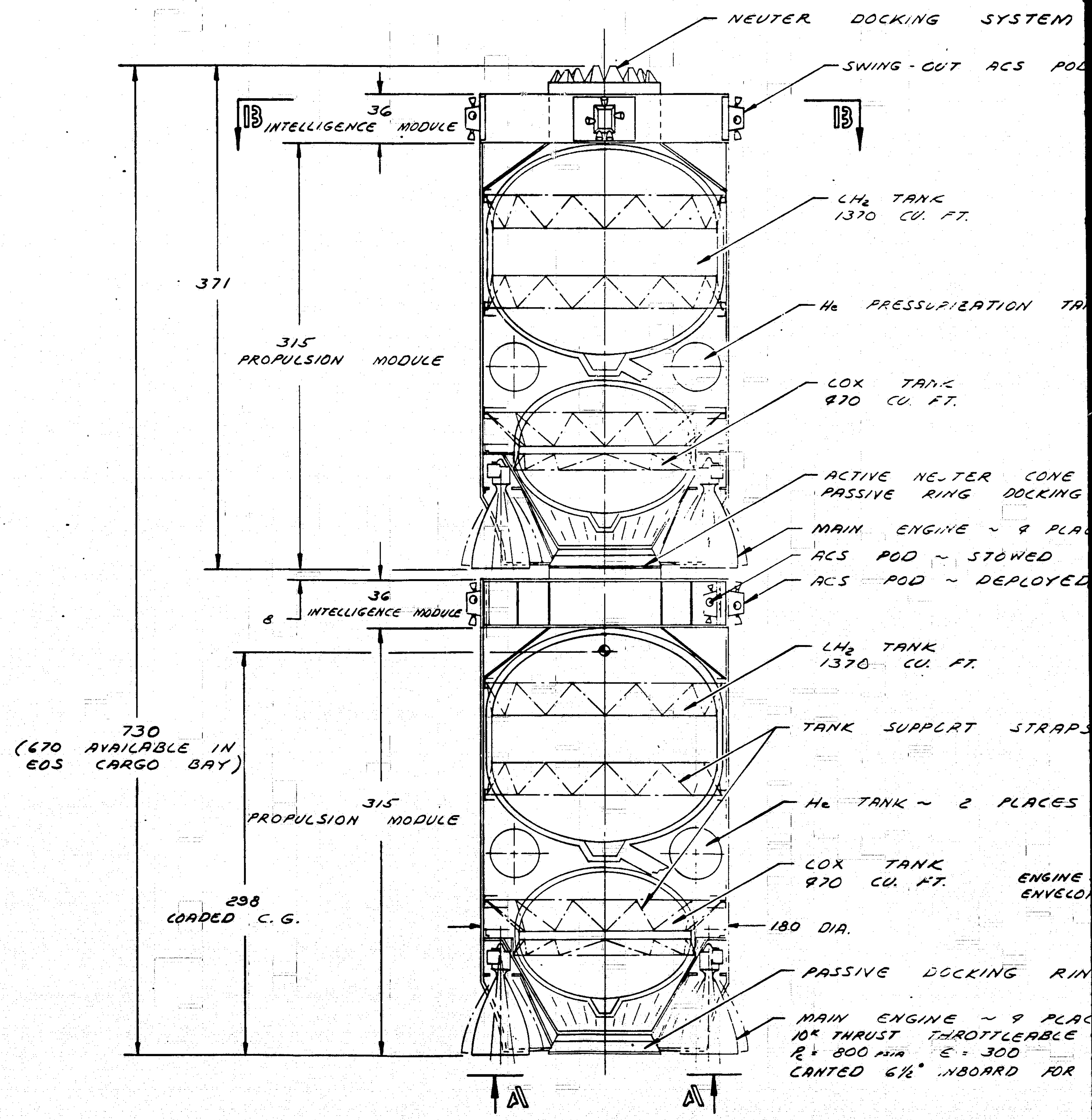


Figure 2-17. 15-Foot-Diameter, 41K Pound Propellant Lunar Landing Tug (Sheet 2 of 2)



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Space Division
North American Rockwell

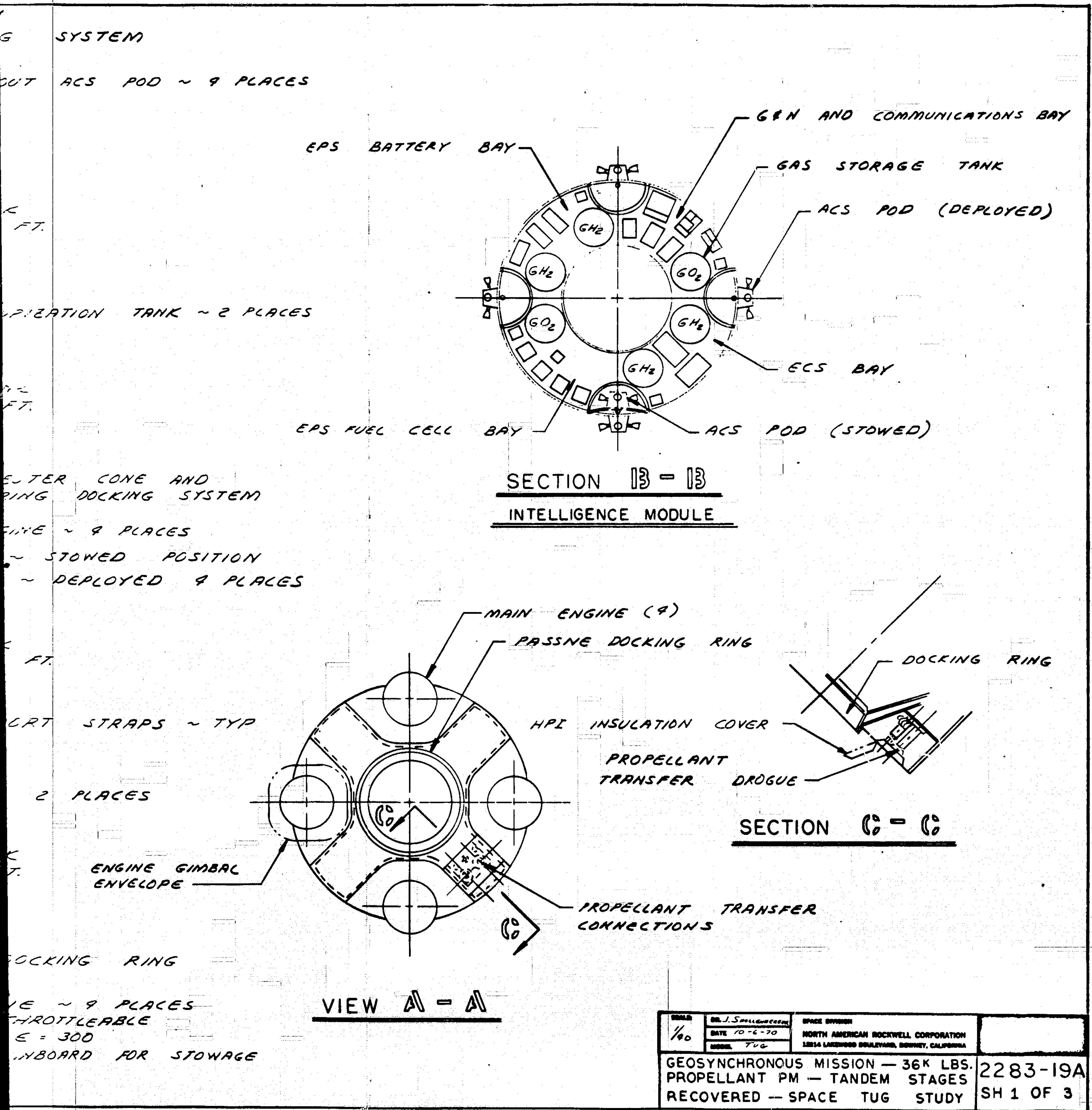

Figure 2-18. 36K Pound Propellant Propulsion Module - Tandem Stages Recovered Geosynchronous Mission (Sheet 1 of 2)

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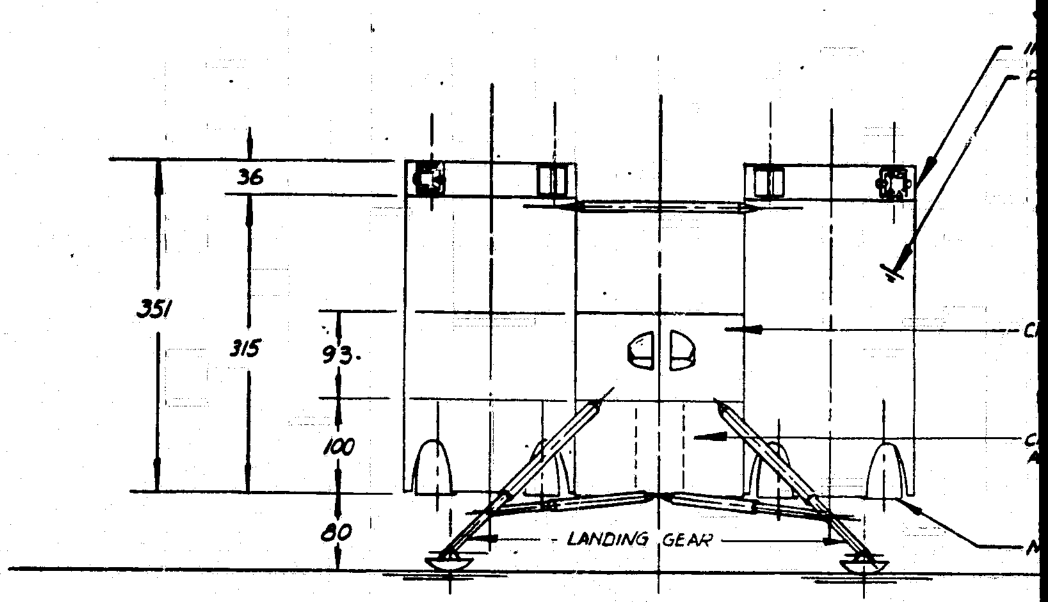

PARALLEL STAGES LUNAR LANDER MODE A

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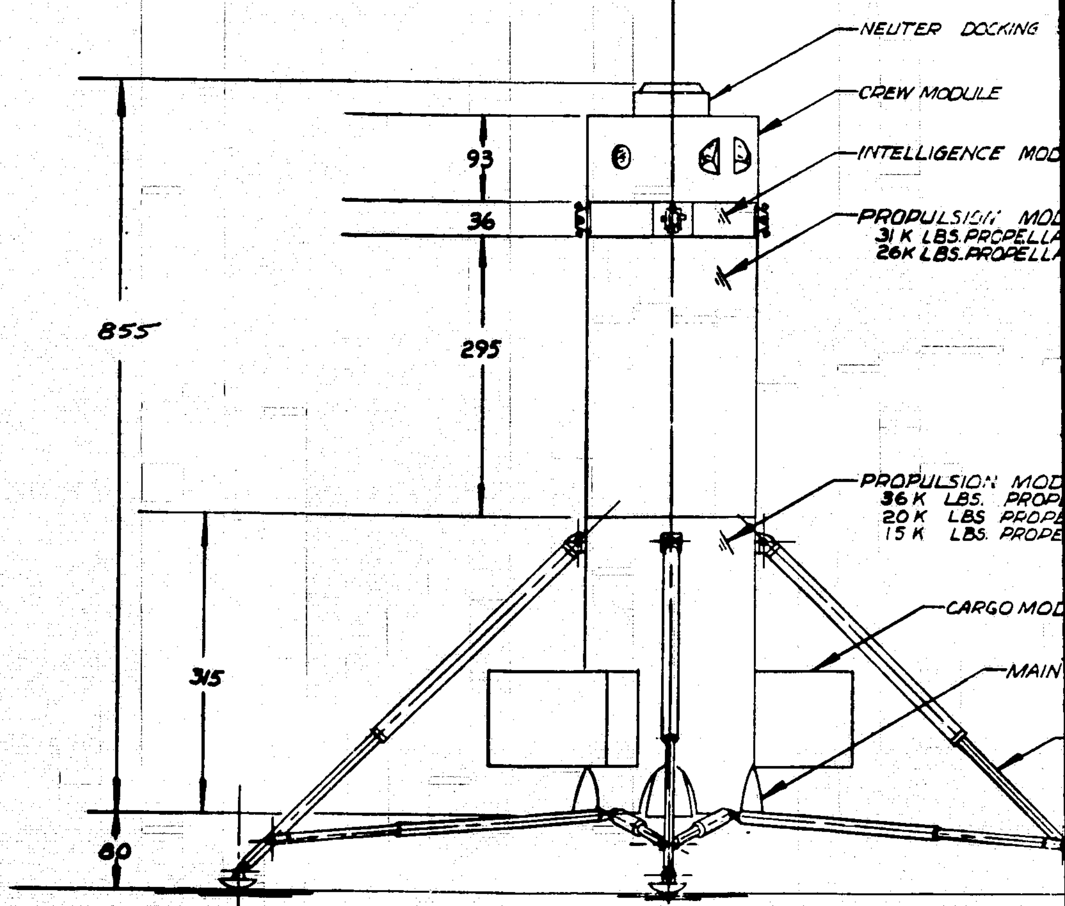

SINGLE STAGE LUNAR LANDER MODE A & B

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2283 - 17
SHEET 3 OF 3

FOLDOUT FRAME 1

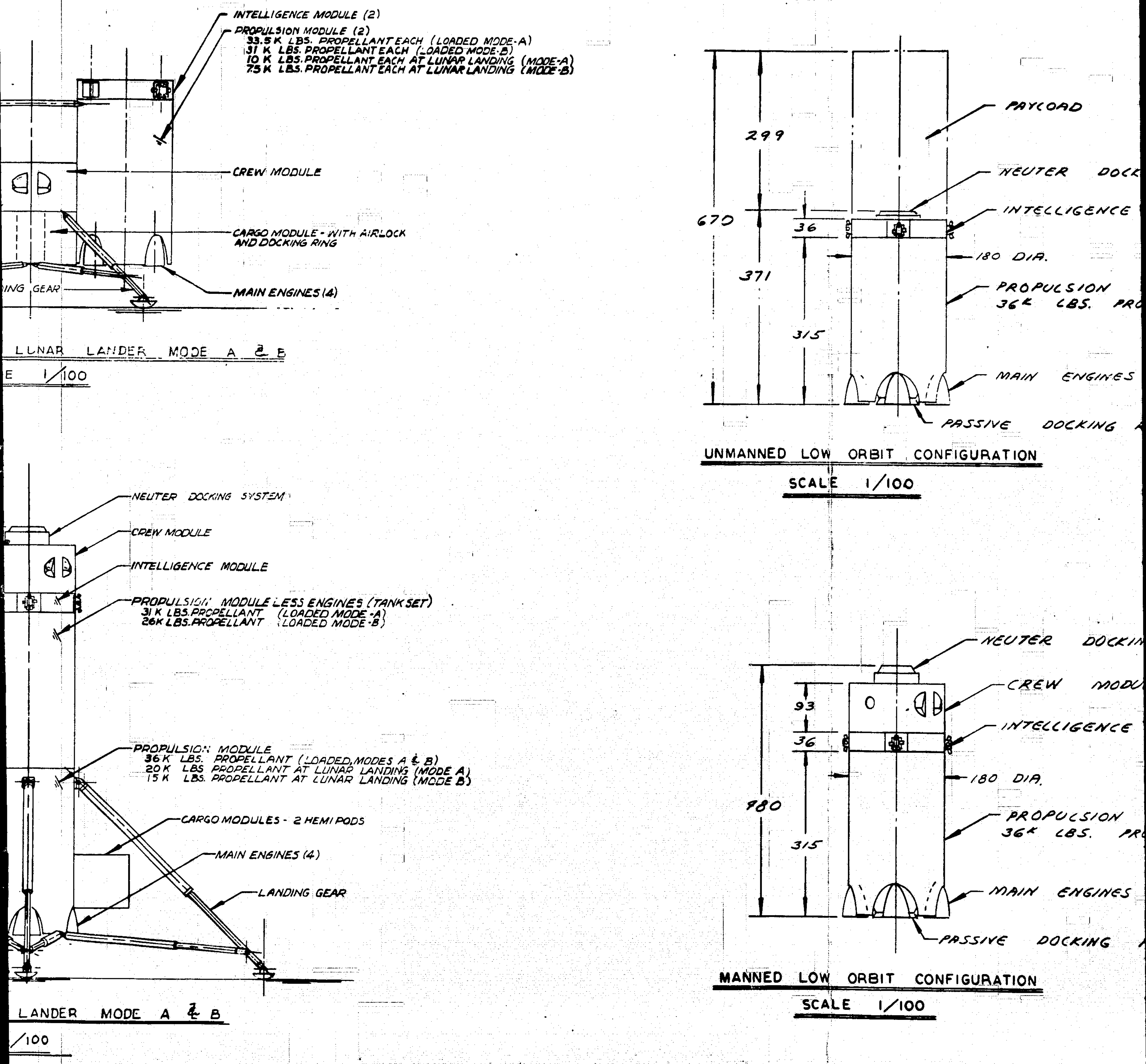
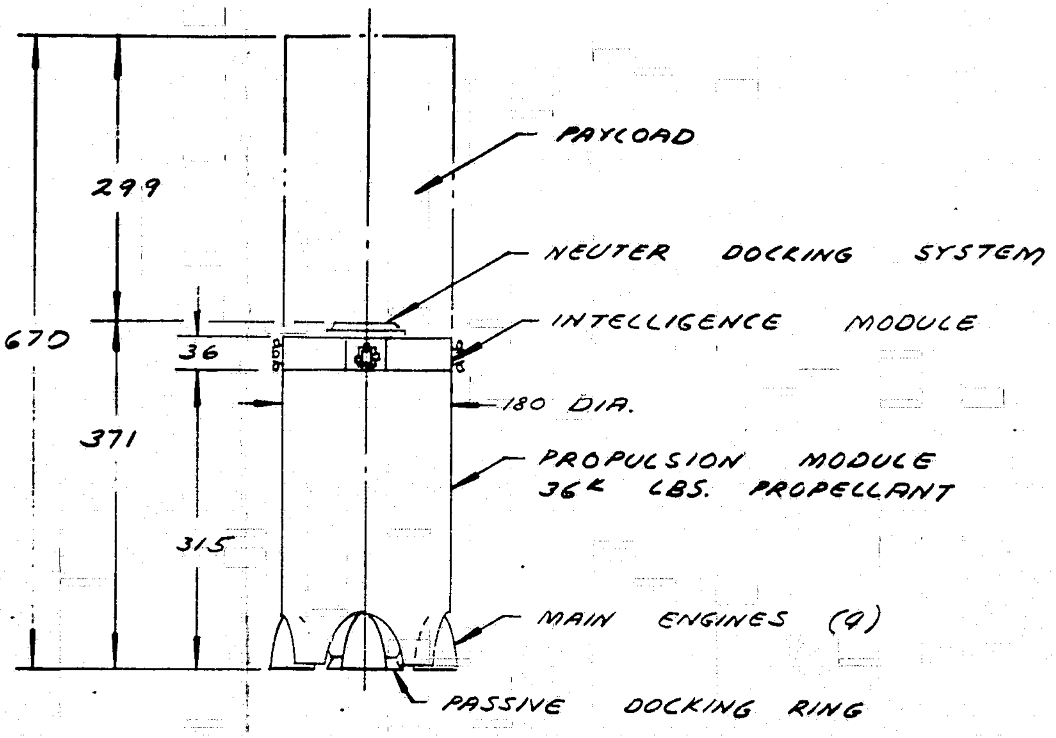


Figure 2-18. 36K Pound Propellant
Recovered Geosy



UNMANNED LOW ORBIT CONFIGURATION

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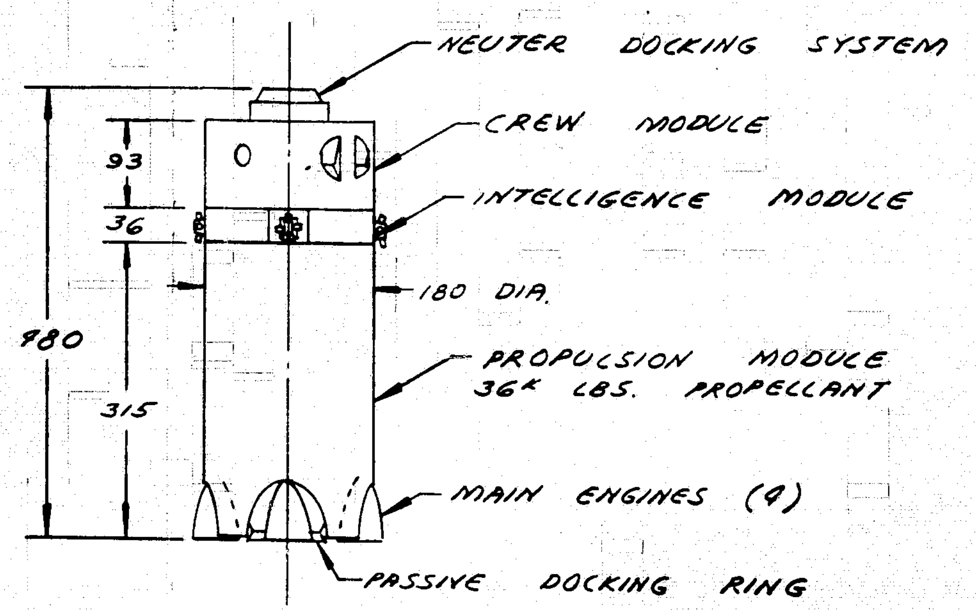


MANNED LOW ORBIT CONFIGURATION

SCALE 1/100

2283 - 19
SHEET 2 OF 3

Figure 2-18. 36K Pound Propellant Propulsion Module - Tandem Stages Recovered Geosynchronous Mission (Sheet 2 of 2)

tug where both stages are recovered. Each stage consists of a PM and an IM. The structural arrangement of the PM is identical to that on drawing 2283-17. Each PM delivers 36,000 pounds (16,329 kilograms) of propellant. The basic stage is 371 inches (9.4 meters) long and when the two are docked together the vehicle is 730 inches (18.5 meters) long.

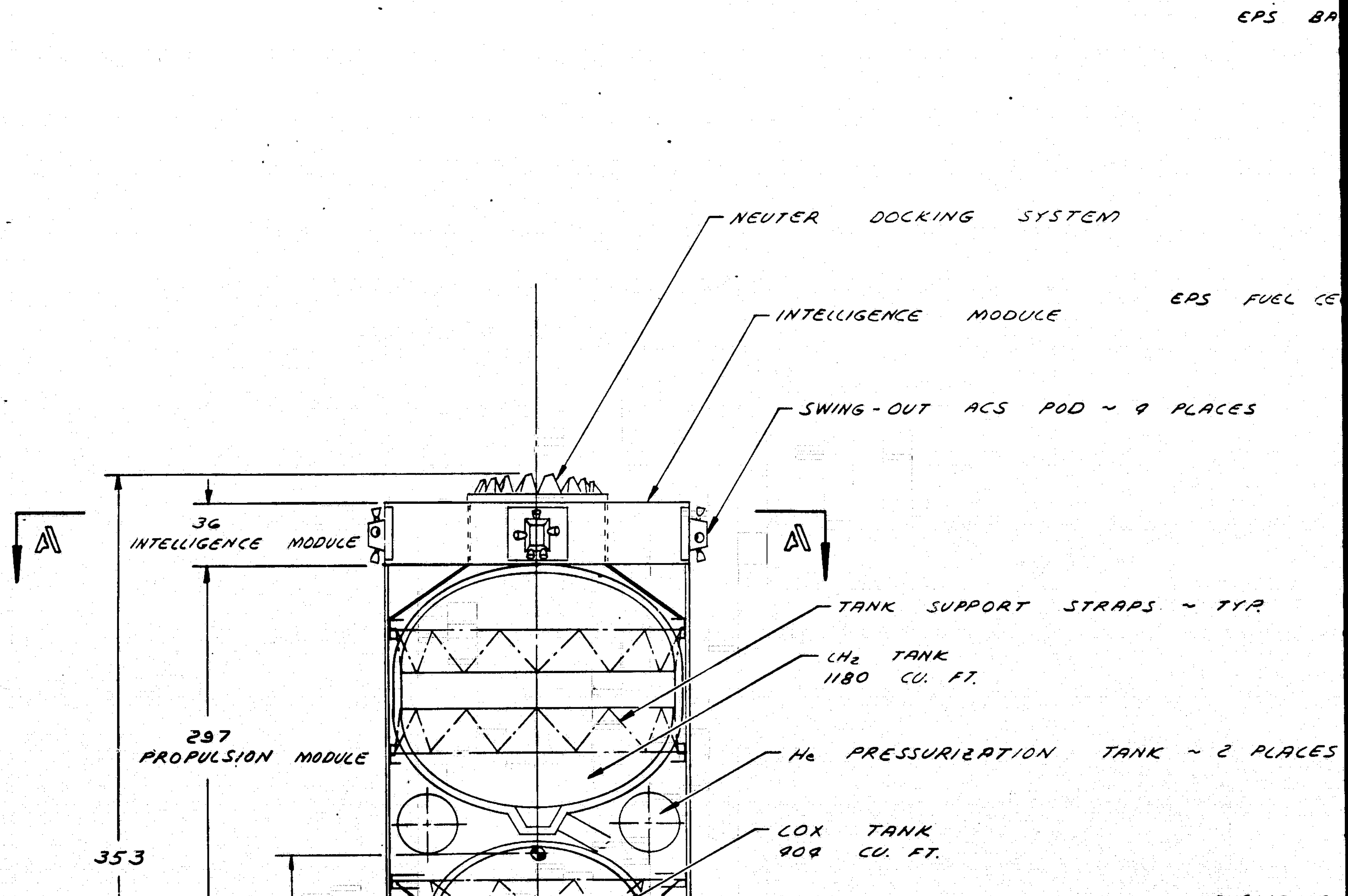
Each stage is made up of a PM with an IM on the forward end. The separable IM is as described and incorporates an active neuter docking system at its forward end. The docking system is partially buried within the central cavity in the IM.

Other mission configurations are also schematically shown on the figure. The basic PM with IM can be used as an unmanned low earth orbit configuration, which is 371 inches (9.4 meters) long. With the addition of a CM at the forward end, the vehicle increases in length to 480 inches (12.2 meters) to perform manned low earth orbit missions.

The stage can perform the lunar landing Mode A mission as either a single stage or two stages operating parallel. As a single stage, the engines and docking provisions are removed from a PM to make a tank set. The tank set is attached atop a PM, and the IM is placed on top of the tank set. A CM is added at the forward end of the vehicle and yields a single-stage lander, which is 855 inches (21.7 meters) long. The PM is fully loaded to perform the Mode A landing mission, and the tank set delivers 31,000 pounds (14,061 meters) or propellant, which is slightly offloading the tank set. The Mode B mission requires the tank set to deliver 26,000 pounds (11,793 kilograms) of propellant, while the propulsion module is again fully loaded.

As in the previous concept (Concept 3) another lander configuration is two stages operating in a parallel mode. A CM and CAM are provided between the two stages. This arrangement is identical to that used on Concept 3, and the approach will not be repeated here. The PM's deliver 33,500 pounds (15,195 kilograms) of propellant each to perform the lander Mode A mission. To perform the Mode B mission requires each stage to deliver 31,000 pounds (14,061 kilograms). This configuration is 351 inches (8.9 meters) long.

Concept 6 was the last configuration to be defined within this group. The concept is based on using two stages to perform a geosynchronous mission where only one stage is recovered. The propulsion module size is such that the lunar landing Mode A mission can be performed and both stages recovered. This concept is shown in Figure 2-19. Since this concept uses various combinations of propulsion modules, intelligence modules and tank sets to perform the baseline missions, only the basic propulsion module with intelligence module has been configured.



31,000 LBS. PROPELLANT PROPULSION MODULE — LOW ORBIT CONFIGURATION

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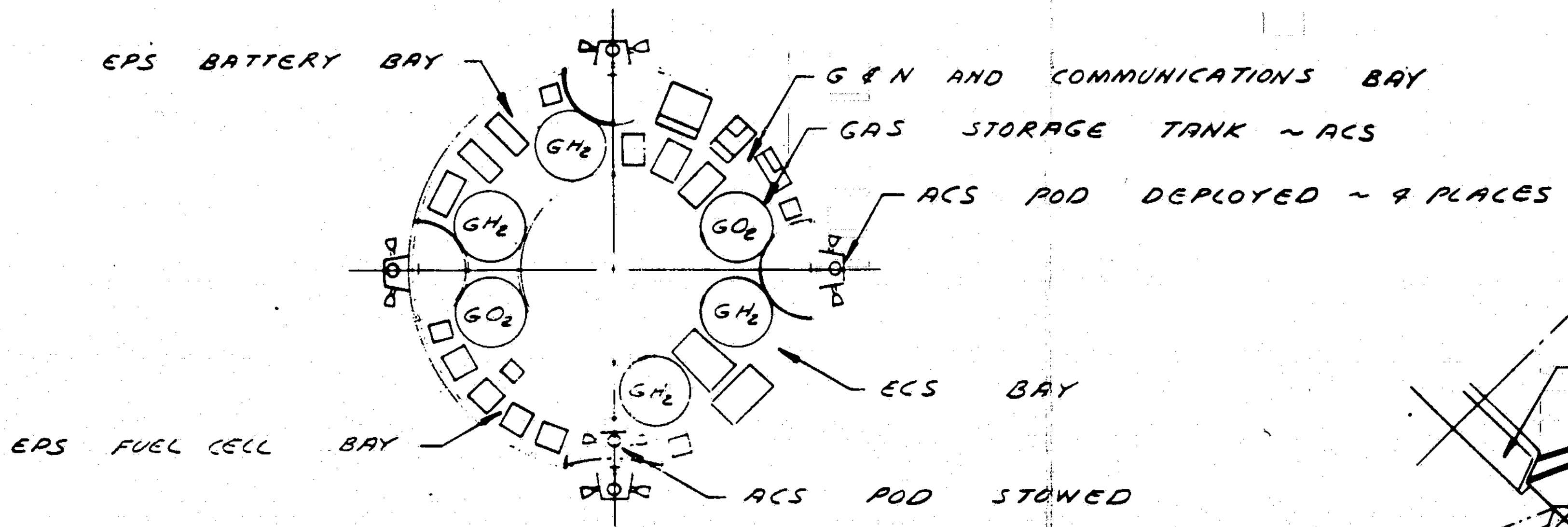
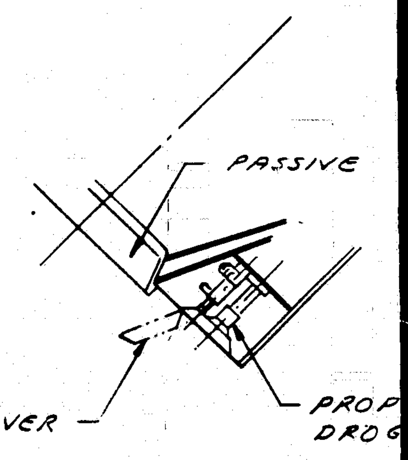


SECTION A - A

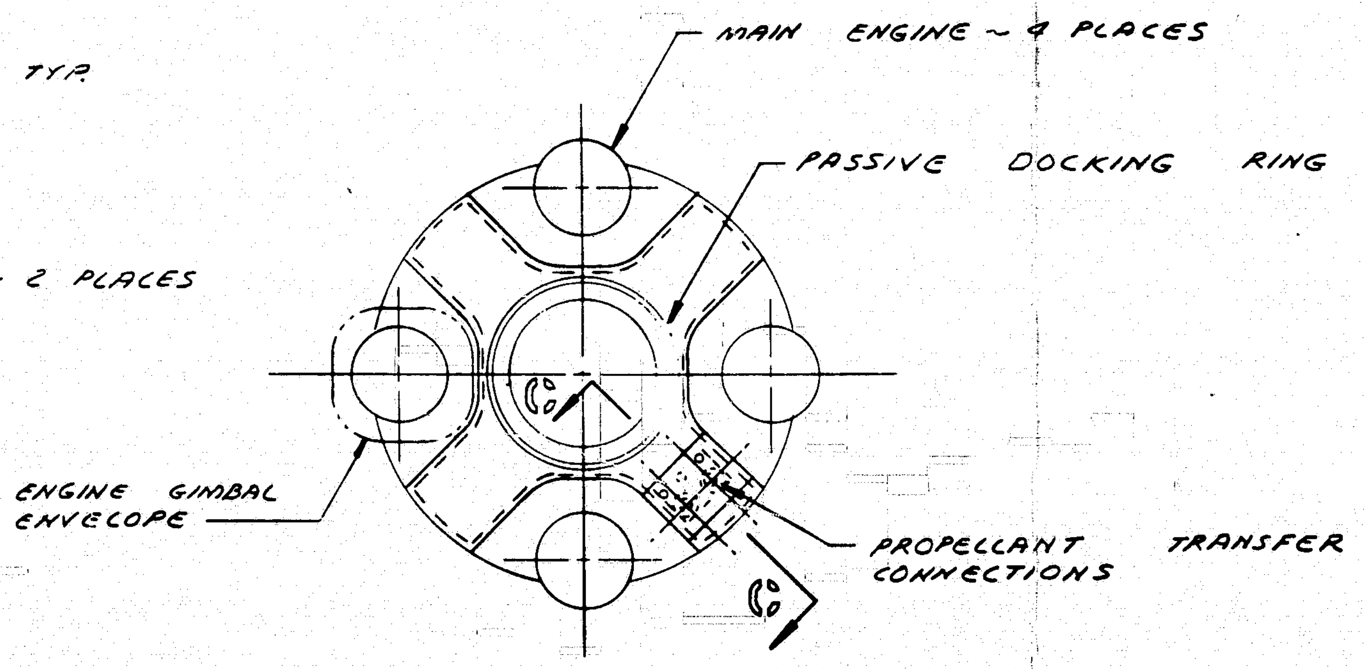
INTELLIGENCE MODULE



SECTION C - C

STRAPS ~ TYP

ION TANK ~ 2 PLACES



~ 4 PLACES
OTTLEABLE
300
6½° INBOARD FOR STOWAGE

VIEW B - B

ATION

1/40	J. SHOLLENBERGER	
	DATE 10-8-70	
	MODEL TUG	

2 STAGE GEOS
LANDER MISSIO
PM — SPACE

Figure 2-19. 31K Pound Propellant Propulsi
Geosynchronous and Lunar Lanc

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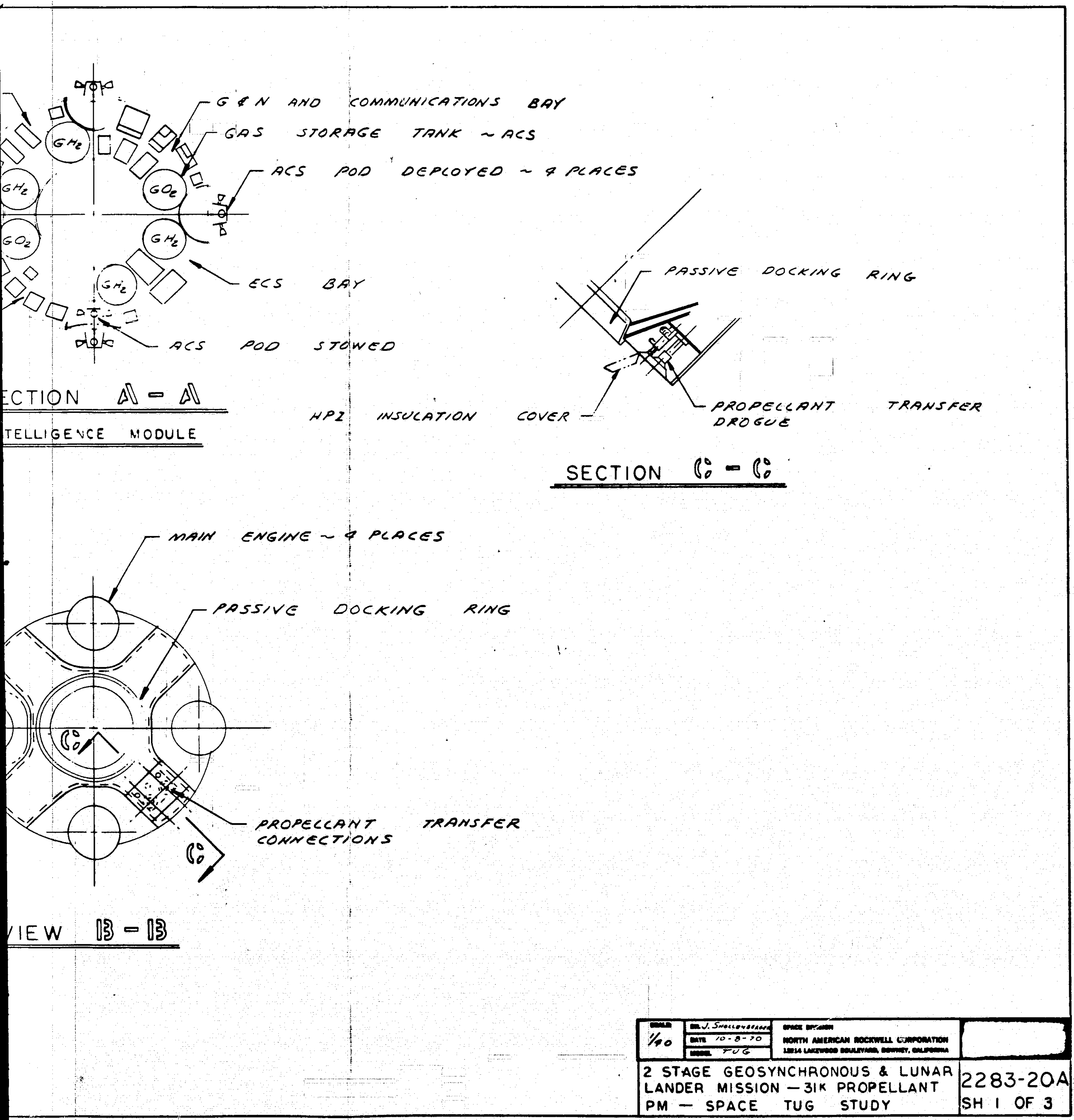

Figure 2-19. 31K Pound Propellant Propulsion Module - Two Stage Geosynchronous and Lunar Lander Mission (Sheet 1 of 2)

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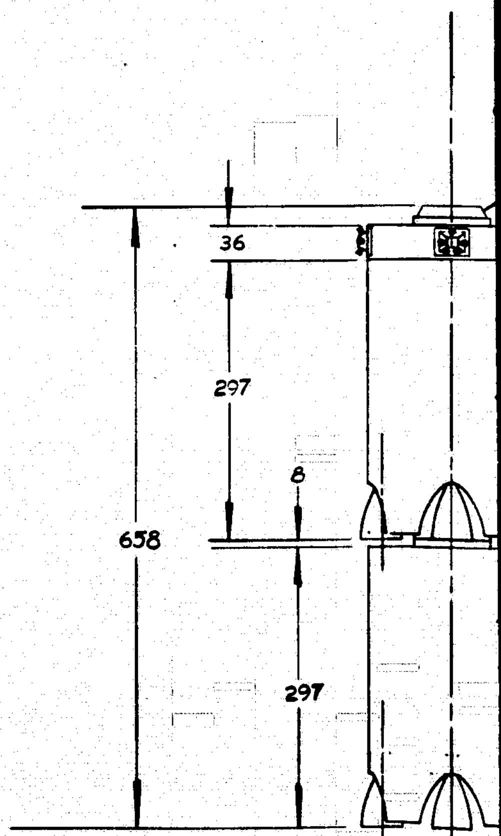

GEOSYNCHRONOUS CONFIGURA

1/100 SCALE

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SHEET 3 OF 3

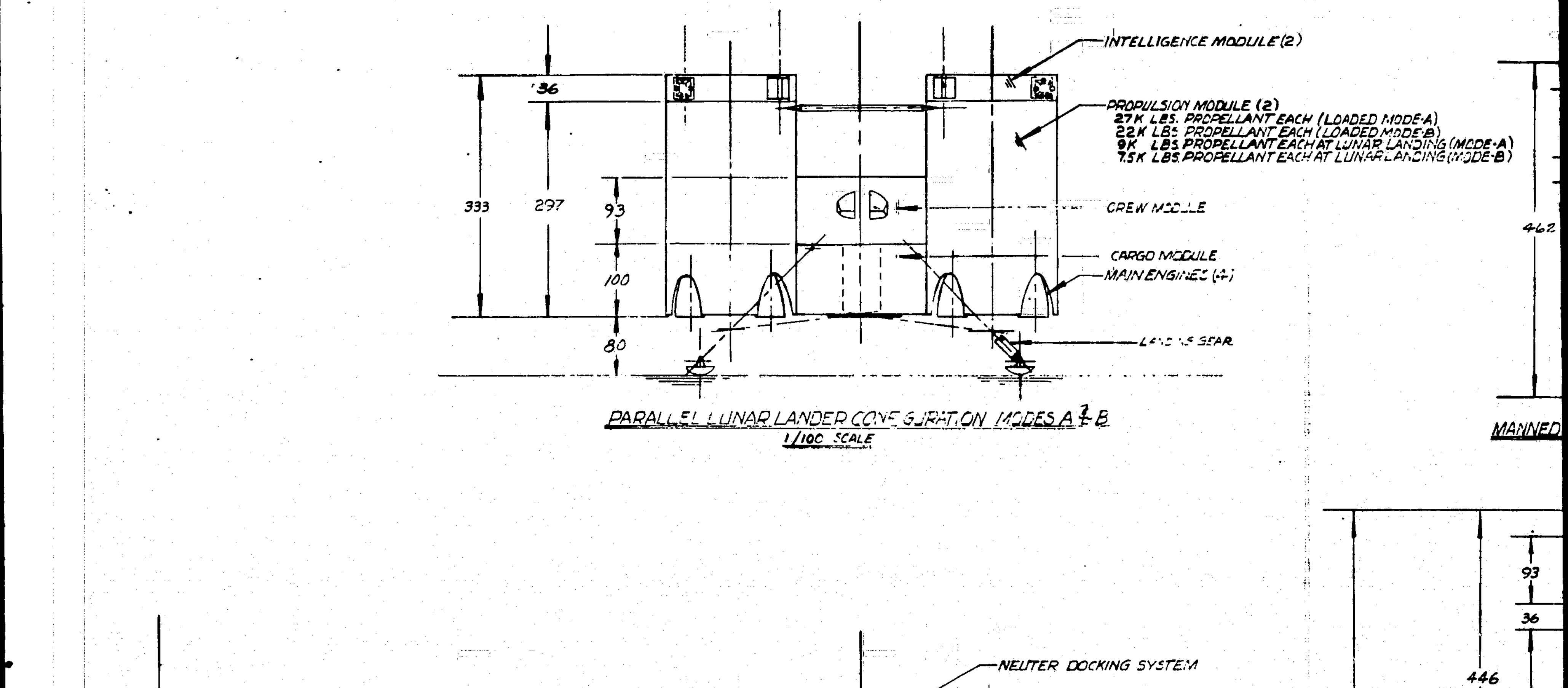

Figure 2-19. 31K Pound P
Geosynchronou

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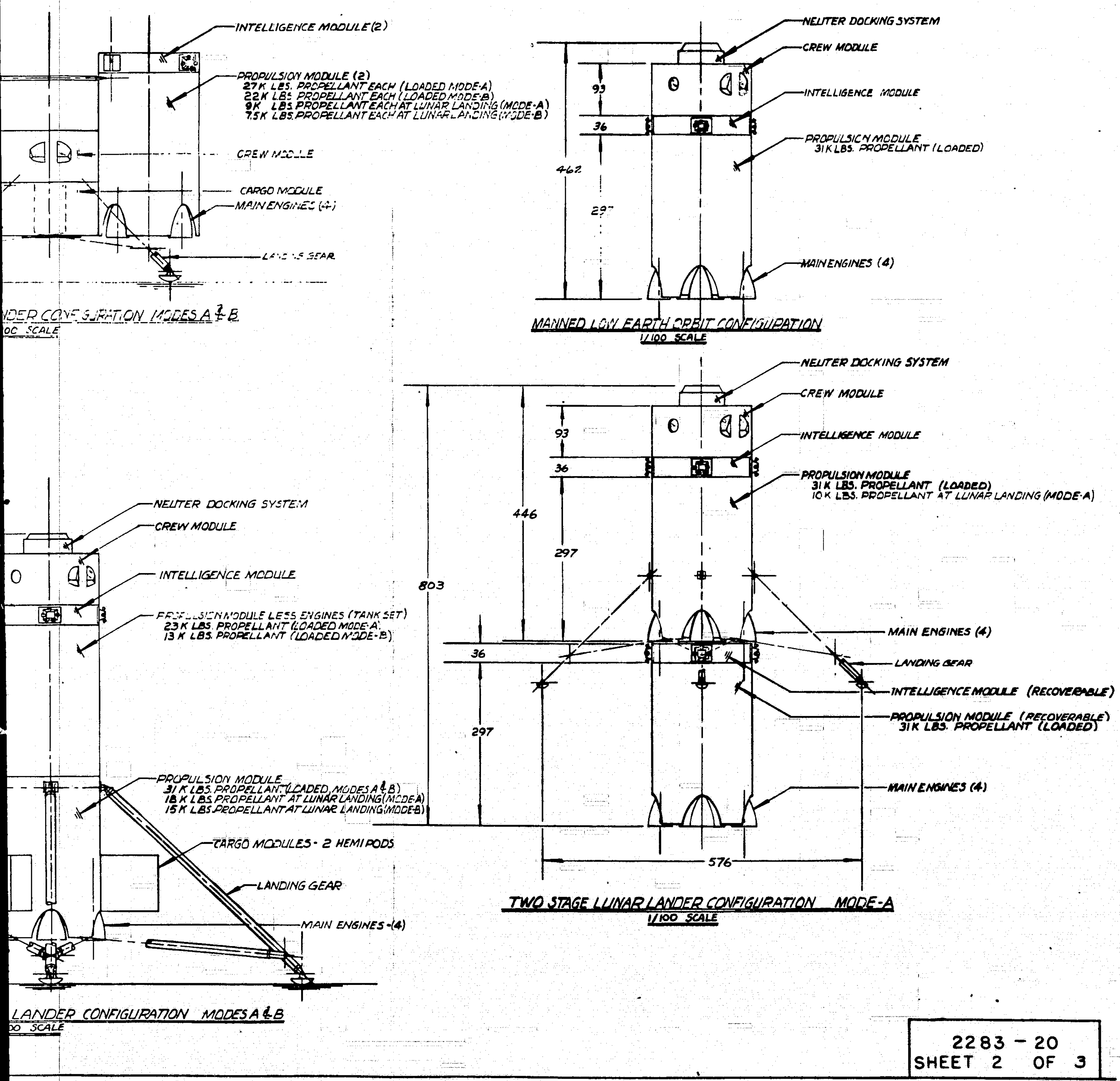


Figure 2-19. 31K Pound Propellant Propulsion Module - Two Stage Geosynchronous and Lunar Lander Mission (Sheet 2 of 2)

FRAME 2

The propulsion module is sized to deliver 31,000 pounds (14,061 kilograms) of propellant. The structural arrangement and design philosophy are as described for this group of concepts. The intelligence module is also identical to that used on the previous concepts. The single LH_2 tank has an internal volume of 1190 cubic feet (33.7 cubic meters), while the LOX tank internal volume is 404 cubic feet (11.4 cubic meters). The propellant refueling system is located on the aft bulkhead and is as described. The unmanned low earth orbit configuration is 353 inches (8.9 meters) long.

A spectrum of configurations is included schematically to encompass the various missions identified for this concept. The manned low earth orbit configuration accommodates a crew module atop the intelligence module of the stage. An active docking system is mounted on the forward end of the crew module. The overall length of this vehicle is 462 inches (11.7 meters). Two stages are arranged in tandem to perform the geosynchronous mission. The aft stage consists of only a propulsion module and is expended in performing the mission. The forward stage is made up of a propulsion module and an IM and is recovered. The two-stage vehicle combination is 658 inches (16.7 meters) long.

Three lunar lander configurations have been shown. The first uses two stages in tandem with the CM and landing gear on the forward stage. Both stages are recovered and consist of a PM with an IM. The aft stage is separated during lunar descent and returns to orbit. The forward stage with the crew module mounted on the forward end and a landing gear kit then descends to the lunar surface and is recovered. This configuration performs the Mode A mission.

The vehicle is 803 inches (20.4 meters) long overall and the landing portion is 446 inches (11.3 meters) long. Another tandem arrangement uses a tank set rather than an additional stage. The tank set is made from a propulsion module without the engines and aft docking provisions. The tank set is 277 inches (7.0 meters) long and is mounted on the forward end of a PM. An IM is located on the forward end of the tank set, and a LM is placed on the forward end of the IM. The landing gear is an in-space-attached kit. The tank set delivers 23,000 pounds (10,433 kilograms) of propellant for the Mode A mission and 13,000 pounds (5897 kilograms) for the Mode B mission. The PM is fully loaded in both mission modes. The vehicle is 739 inches (18.8 meters) long.

The last lunar lander configuration uses two stages operating in a parallel mode. A crew module and cargo module is placed between the stages. Each stage consists of a PM and an IM and delivers 27,000 pounds (12,247 kilograms) of propellant for mission Mode A. Each stage requires 22,000 pounds (9979 kilograms) of propellant to perform the Mode B mission.

The landing gear is attached to the CAM to reduce the scar weight on the stages and CM. The vehicle is 333 inches (8.5 meters) long, which makes it a relatively short lander.

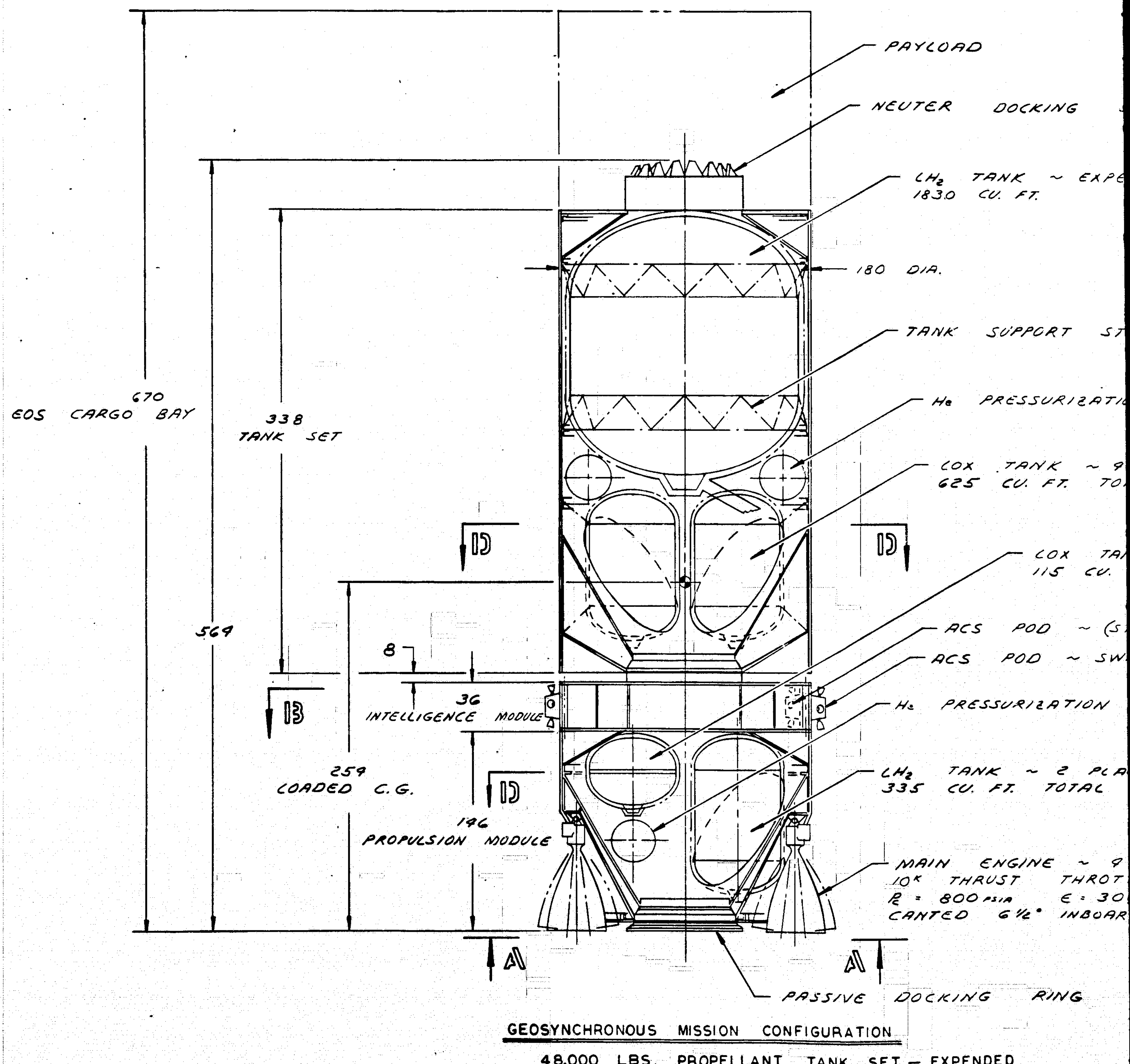
The tandem two-stage lander, where the first stage is separated during descent, makes an attractive lander stage. However, the operational complexity of the separation and recovery of the first stage and landing of the second stage may not be warranted and would perhaps eliminate this concept from consideration.

One-and-One-Half-Stage Expended Tank Set Concept

The last concept selected from Table 1-2 for design definition was Concept 11 which is identified as the 1-1/2-stage concept. The configuration is shown in Figure 2-20. This concept is based on performing a geosynchronous mission with a large expendable tank set and a small recoverable PM. The tank set delivers 48,000 pounds (21,772 kilograms) of propellant, and the PM accommodates 8,800 pounds (3,992 kilograms). The structural arrangement and design philosophy are similar to that of the previous concepts. The tank set uses a single LH_2 tank and four LO_2 tanks. The PM uses two LH_2 and two LO_2 tanks, which have the same inside diameter as the tank set LO_2 tanks. All bulkheads on the tanks are 1.4 to 1 ratio ellipsoids. The LH_2 tank is suspended from a girth ring by support straps. Radial beams are provided in the area between the smaller diameter tanks. The smaller tanks are suspended from the radial beams and girth rings by straps.

The PM is 202 inches (5.1 meters) long and is configured as a shell with four radial beams. The beams support the four main engines as well as the propellant tanks and aft docking system. The LH_2 tanks have an internal volume of 167.5 cubic feet (4.7 cubic meters) each, and the LO_2 tanks have an internal volume of 57.5 cubic feet (1.6 cubic meters) each. An IM is located at the forward end of the PM. The internal arrangement of the IM is identical to that used on the previous concepts.

The tank set is also a shell structure with radial beams between the four LO_2 tanks. The LO_2 tanks are the same diameter as the PM tanks and have the same bulkheads. The tank set LH_2 tank internal volume is 1830 cubic feet (518 cubic meters). The LO_2 tanks have an internal volume of 156.25 cubic feet (44.2 cubic meters) each. The overall length of the tank set is 374 inches (9.5 meters). When the tank set is attached to the PM, the overall vehicle length is 564 inches (14.3 meters). The tank set is expended and jettisoned in geosynchronous orbit, while the PM and IM are returned to low earth orbit. Propellant refueling of the PM is as described for the other concepts.



GEOSYNCHRONOUS MISSION CONFIGURATION

48,000 LBS. PROPELLANT TANK SET — EXPENDED

8,800 LBS. PROPELLANT PROPULSION MODULE — RECOVERED

FOLDOUT FRAME 1

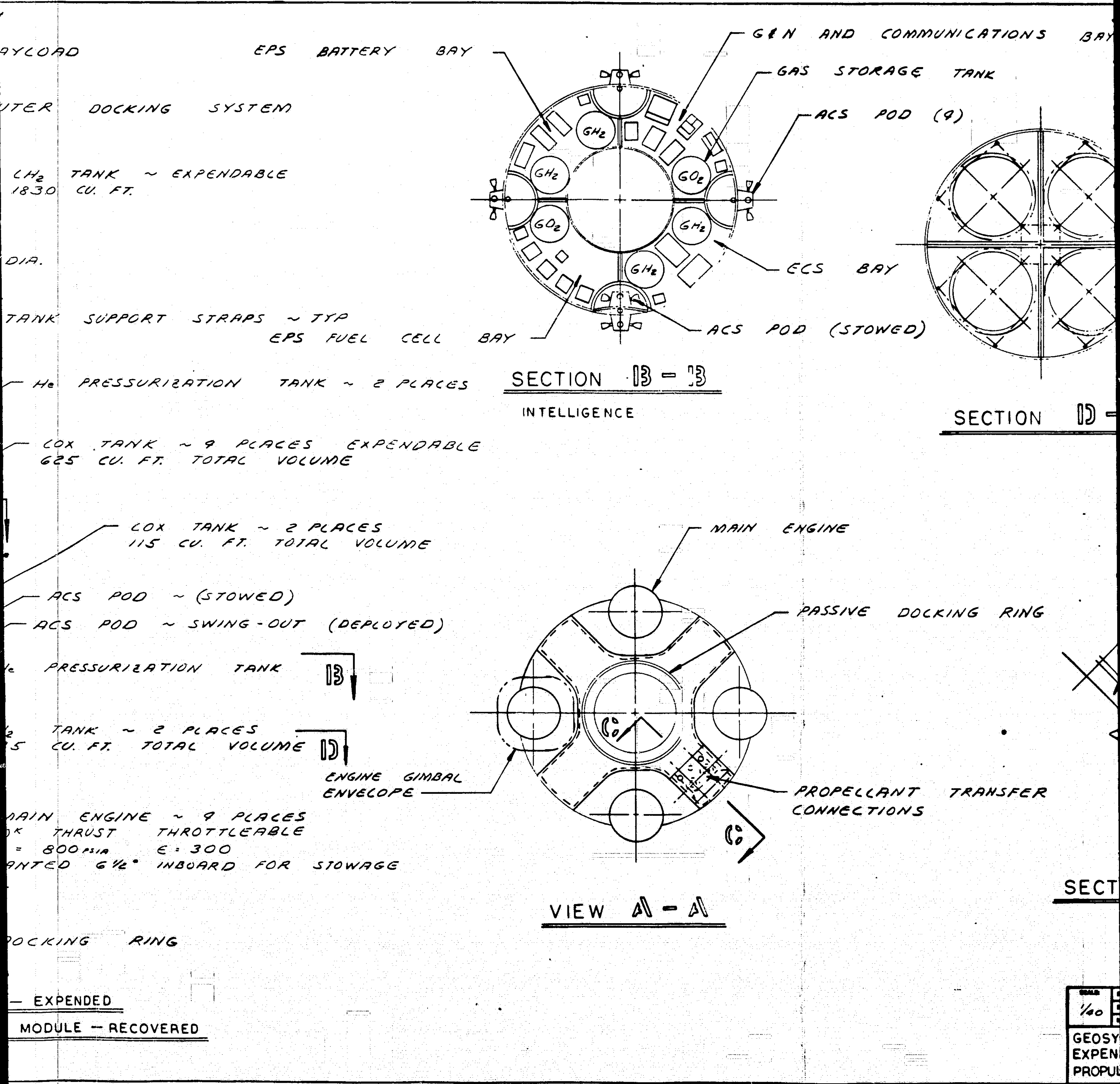


Figure 2-20. Optimum Expendable Tank
Module Geosynchrono

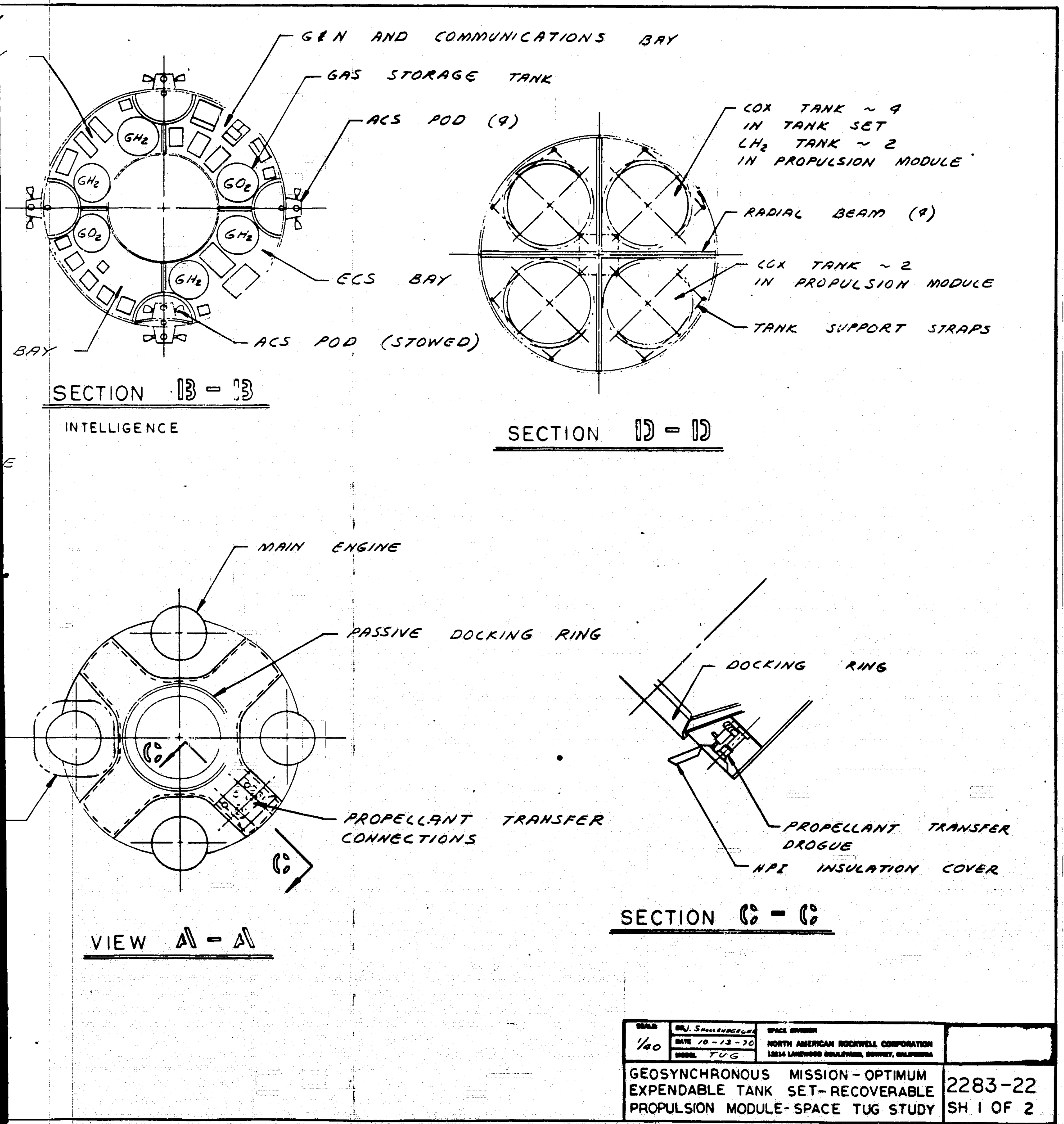
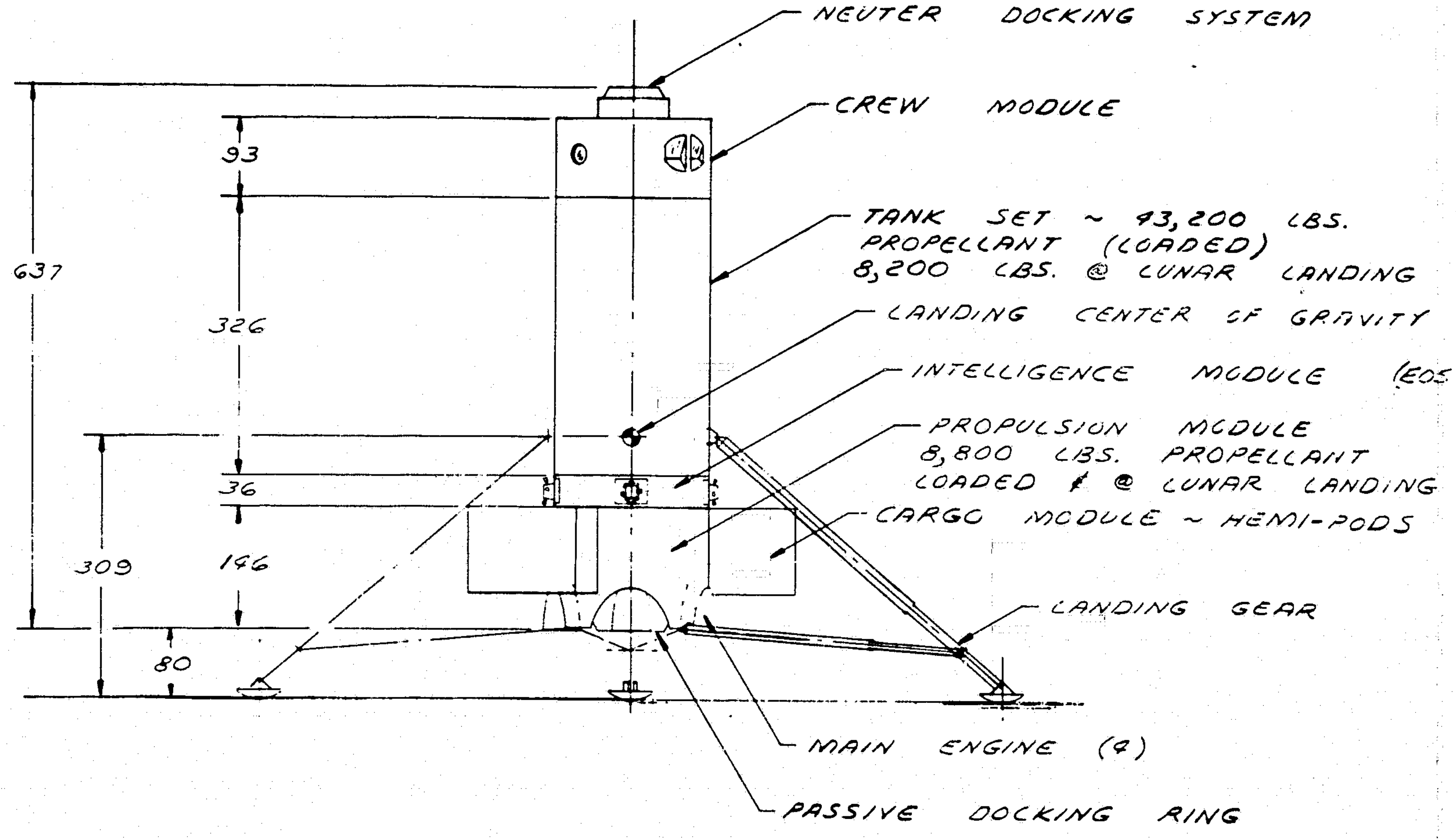


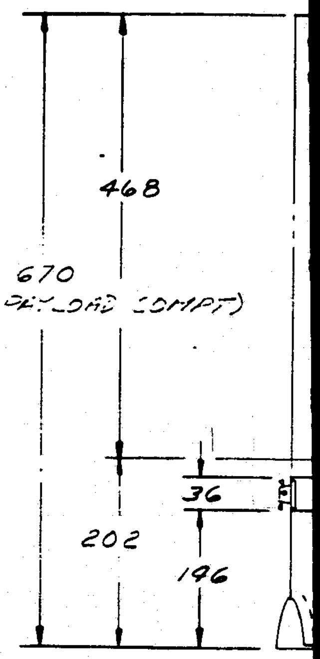
Figure 2-20. Optimum Expendable Tank Set - Recoverable Propulsion Module Geosynchronous Mission (Sheet 1 of 2)

2



SINGLE STAGE LUNAR LANDER CONFIGURATION

SCALE 1/100



UNMANNED LOW

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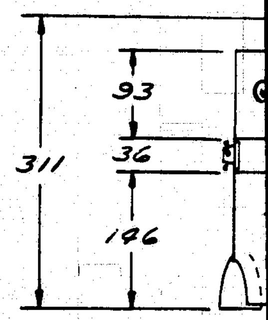

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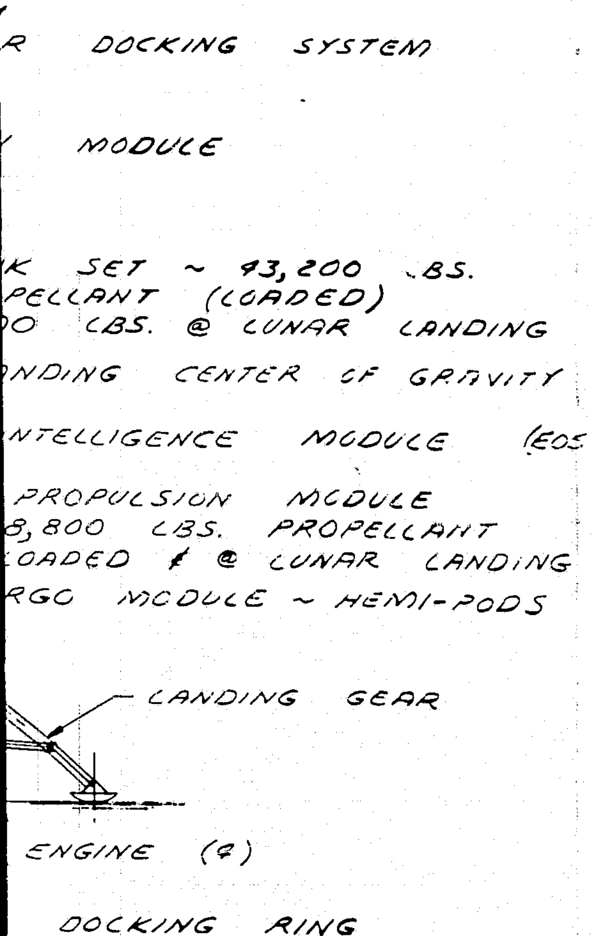
Figure 2-20. Optimum Expe
Module (

FOLDOUT FRAME 1



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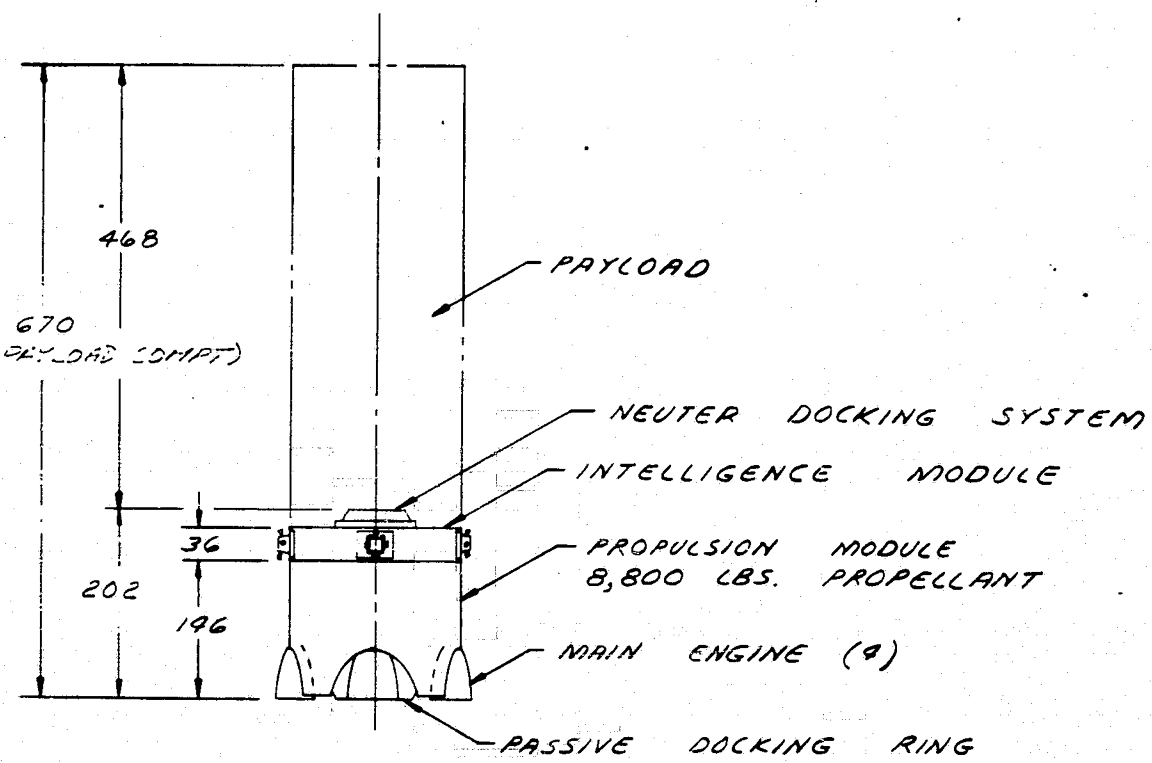
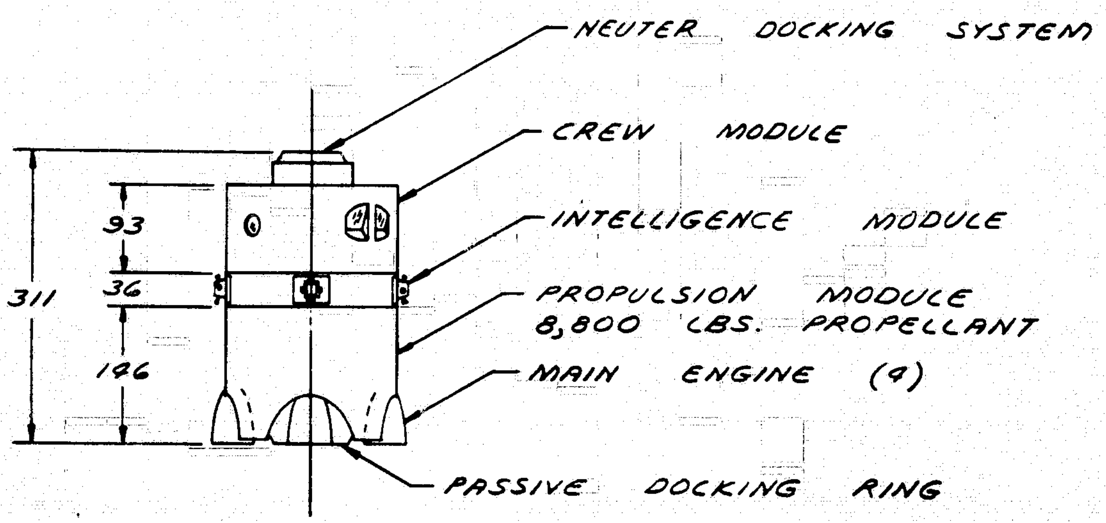


UNMANNED LOW EARTH ORBIT CONFIGURATION

SCALE 1/100



MANNED LOW EARTH ORBIT CONFIGURATION

SCALE 1/100

2283 – 22
SHEET 2 OF 2

Figure 2-20. Optimum Expendable Tank Set - Recoverable Propulsion Module Geosynchronous Mission (Sheet 2 of 2)

FOLDOUT FRAME 2

The PM with IM makes a small unmanned low earth orbit stage that is 202 inches (5.1 meter) long and uses 8800 pounds (3992 kilograms) of propellant. A CM may be added to the forward end of the IM to configure a manned low earth orbit vehicle with an overall length of 311 inches (7.9 meter). This vehicle also has a capacity of 8,800 pounds (3993 kilograms). A lunar lander configuration is shown that is similar to the geosynchronous configuration. For the lunar lander, the tank set is not staged or jettisoned for the lander mission. A CM is added at the forward end of the tank set with an active venter docking system at its forward end. The landing gear is attached as a kit in space. The lander vehicle is 637 inches (16.2 meter) long. The tank set is required to deliver 43,200 pounds (19,595 kilograms) of propellant when the PM is fully loaded.

This concept provides an attractive low earth orbit stage and geosynchronous mission vehicle. The lunar lander configuration is lengthy but not entirely unwieldy since placement of the CM lower on the vehicle could make the lander version also an attractive vehicle.

2.2.6 INTEGRATED INTELLIGENCE MODULE SUBSYSTEMS STUDIES

In the overall Tug vehicle configurations, the most costly module in every instance is the IM. Contained within the IM is all of the equipment necessary to operate the tug vehicle—the attitude control; communications and data management; guidance, navigation and control; electrical power; rendezvous and docking; and thermal control, environmental control, and life support equipment. Early vehicle configurations incorporated a separable IM section that was toroidal shaped and about 3 feet (0.91 meter) deep. Preliminary internal arrangements were prepared, which can be seen on the early concept configurations, based on equipment as it was known at that time. It was decided to retract the ACS modules into the basic IM 15-ft-diameter (4.57-m) moldline in order not to violate the 15-ft (4.57-m) moldline of the EOS cargo bay.

The concept of a separable IM appeared attractive and this early size of module with an internal usable volume of about 425 cubic feet (12 cubic meter) appeared capable of accommodating some growth in the subsystems equipment. Later in the program it was revealed that volume existed within the PM's of various configurations and that it might be used for mounting the equipment in the IM. As a result, a drawing was prepared (Figure 2-21) to investigate the feasibility of integrating equipment into selected areas within PM's.

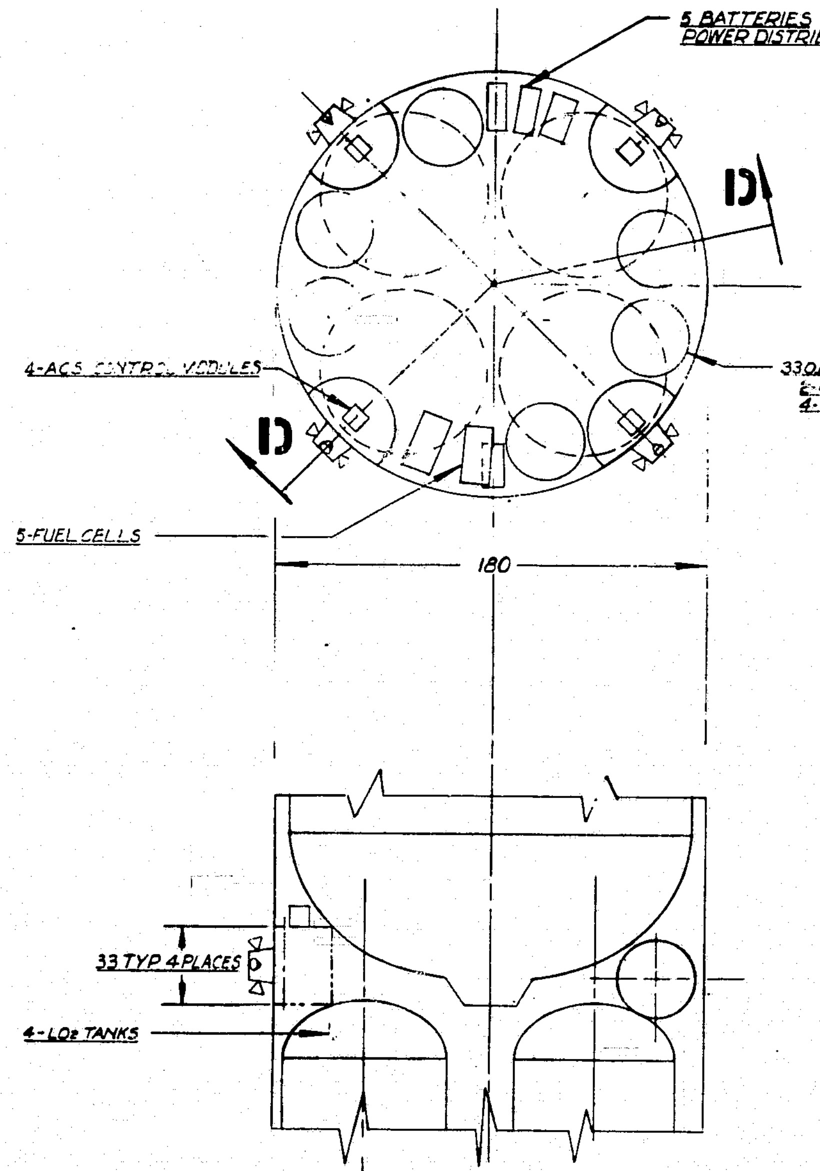
The vehicles selected had propellant capacities of 60,000 and 80,000 pounds (27,215 and 36,287 kilograms) with single LH_2 tanks and

single, as well as multiple, LO_2 tanks. Two configurations were developed that used two areas in the PM to mount the equipment. These configurations accommodated the ACS engine swing-out modules, as well as all equipment. These two concepts (Figures 2-21 and 2-22) used the interstage area between the LH_2 and LO_2 tanks to mount the nonavionics-oriented equipment. The ACS engine modules were incorporated into the basic vehicle structure, and the subsystem equipment then is accommodated in the circumferential area between the engine modules. In each of these concepts the subsystem equipment accommodated is only that which is not avionics oriented, such as the electrical power and ACS control equipment. The ACS gaseous propellant accumulator tanks are also mounted in this circumferential area. Four GH_2 and two GO_2 tanks are used for storage of the ACS and EPS propellants.

The second and third configurations shown on the figures use the annular area near the forward end of the LH_2 tank to mount all of the avionics-oriented equipment, as well as other subsystem equipment not mounted in the interstage area. All of the communications and data management, guidance, navigation and control, rendezvous and docking, thermal control, environmental control, and life support equipment is mounted in this area. Equipment is grouped by subsystem into modules that can be easily inserted into and removed from this annular area. This concept would require configuring this area with subframes and structural supports to accept the equipment modules. None of the ACS equipment, engine modules and tanks, or electrical power equipment is mounted in this area.

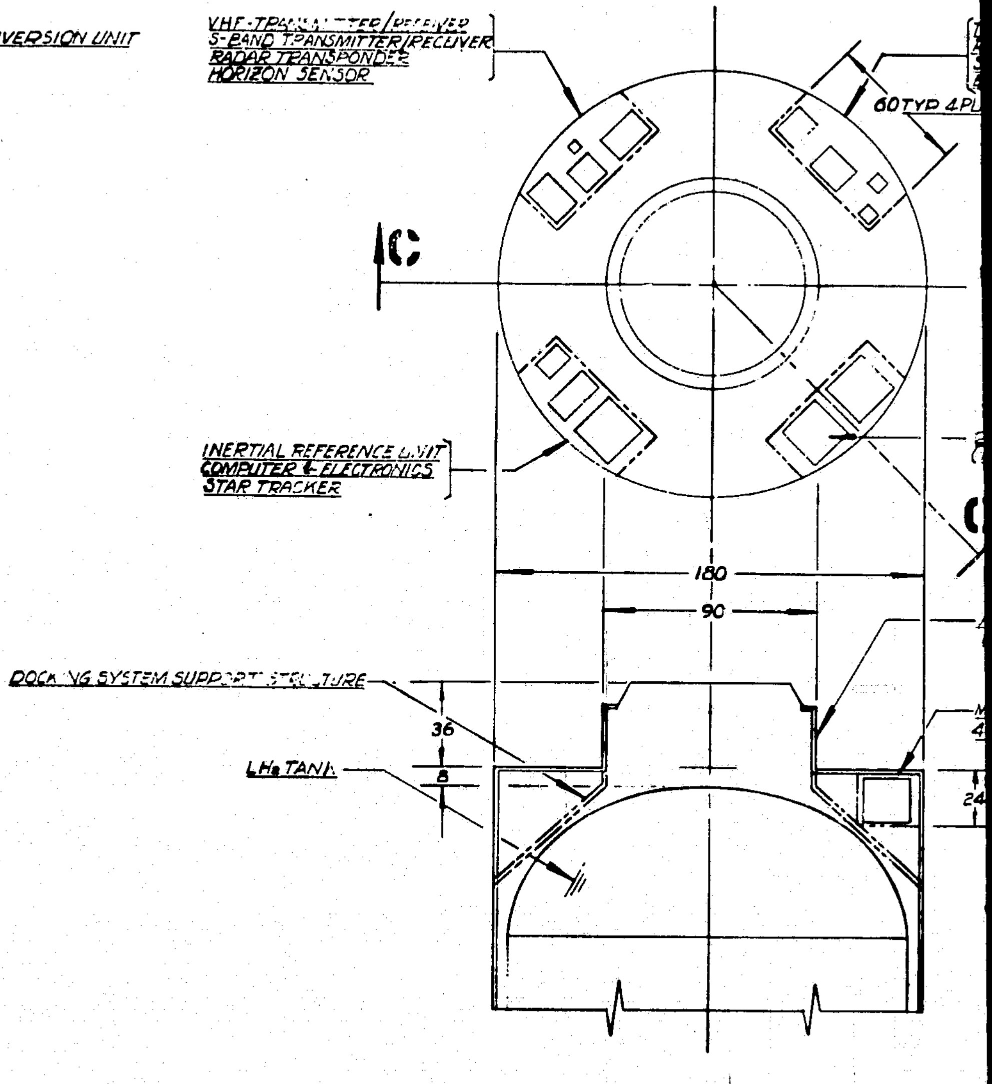
The integrated approach would save some structural weight in that a separate IM structure is not required. However, the heat loads to the areas surrounding the propellants require further investigation to determine if additional thermal control equipment is required. The problems associated with manufacturing a completely integrated PM must also be investigated. A potential manufacturing or refurbishment problem arises in that once the intelligence equipment support structure is installed, it may be impossible to remove the LH_2 tank without first removing the intelligence equipment support structure. Once the intelligence equipment is incorporated into the PM, the concept of a separable IM to be used in conjunction with a small tank set and crew module as a minitug does not exist. The concept of having a separable IM for the minitug, as well as for use with other components of the space program, is attractive and might outweigh the small decrease in Tug vehicle weight attributable to an integrated intelligence equipment approach.

A result of this integrated IM approach study is that the concept appears feasible and the actual weight savings to the vehicle is approximately



SECTION-D D

INTELLIGENCE EQUIPMENT & ACS UNITS
INTER TANK REGION
80K PROPELLANT CAPACITY CONFIGURATION
REF. VEHICLE DRAWING 2283-11



SECTION-C C

INTELLIGENCE EQUIPMENT
REGION AT UPPER END OF LH2 TANK ADJACENT TO DOCKI
60K OR 80K PROPELLANT CAPACITY CONFIGURATIO

SHEET 2 OF 2

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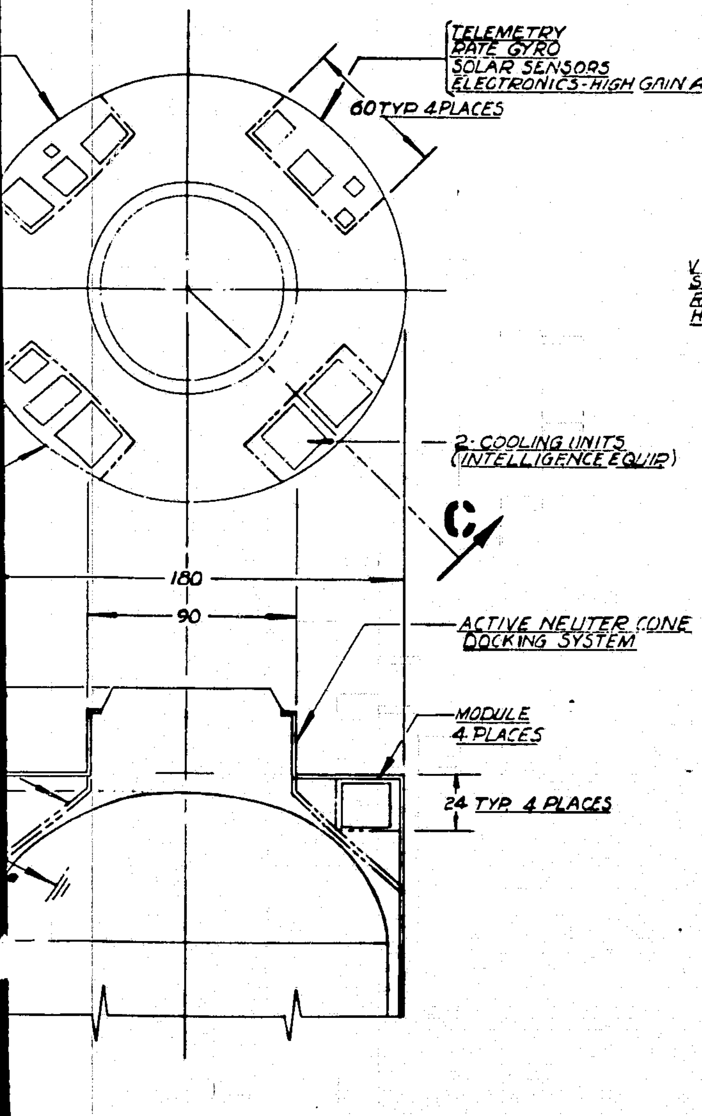

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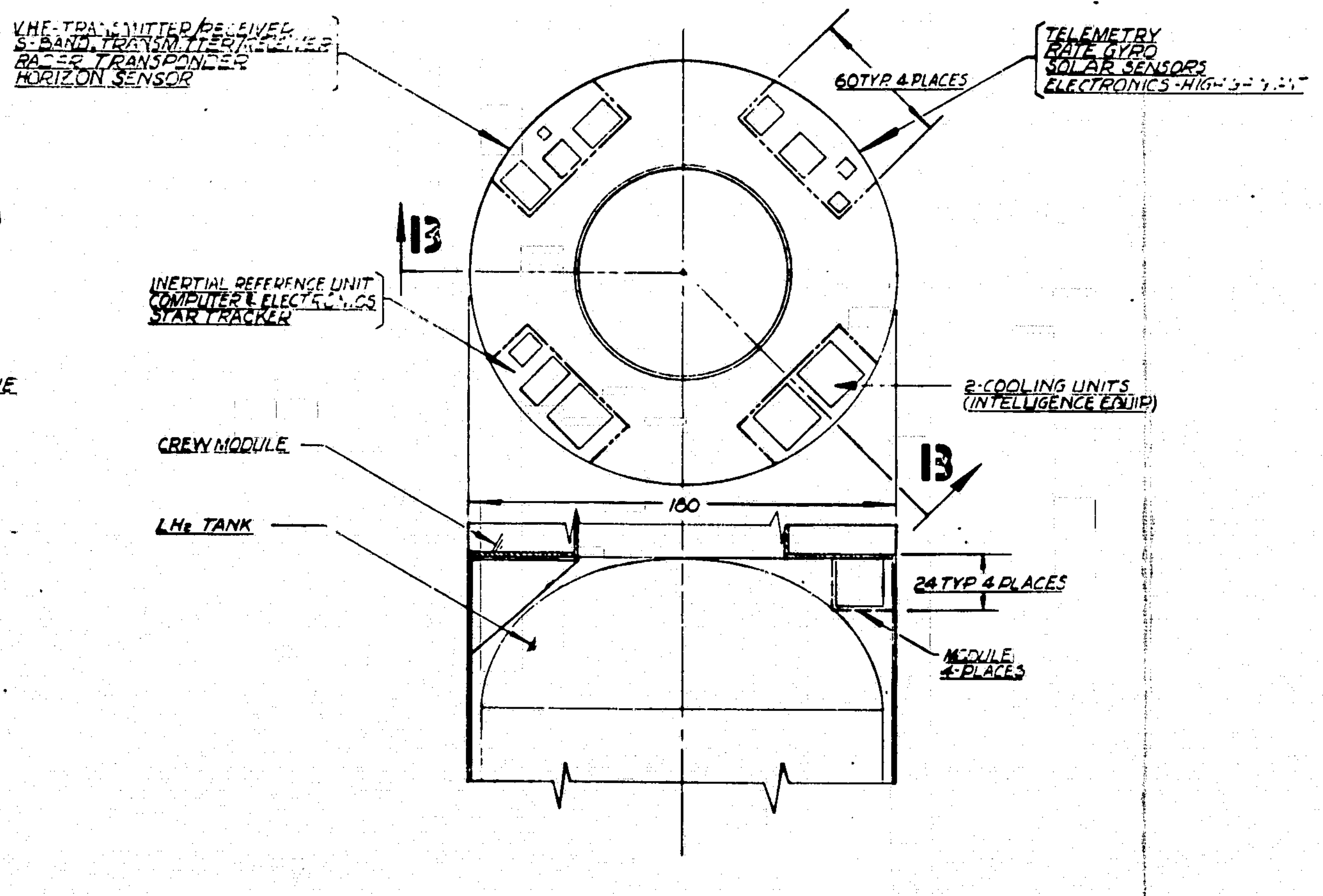

SECTION-B B

INTELLIGENCE EQUIPMENT
ER END OF LH2 TANK ADJACENT TO DOCKING CONE
0K PROPELLANT CAPACITY CONFIGURATIONS

INTELLIGENCE EQUIPMENT
REGION BETWEEN UPPER END OF LH2 TANK & CREW MODULE
FLOOR BULKHEAD
60K or 80K PROPELLANT CAPACITY
CONFIGURATIONS

SHEET 2 OF 2

Figure 2-21.

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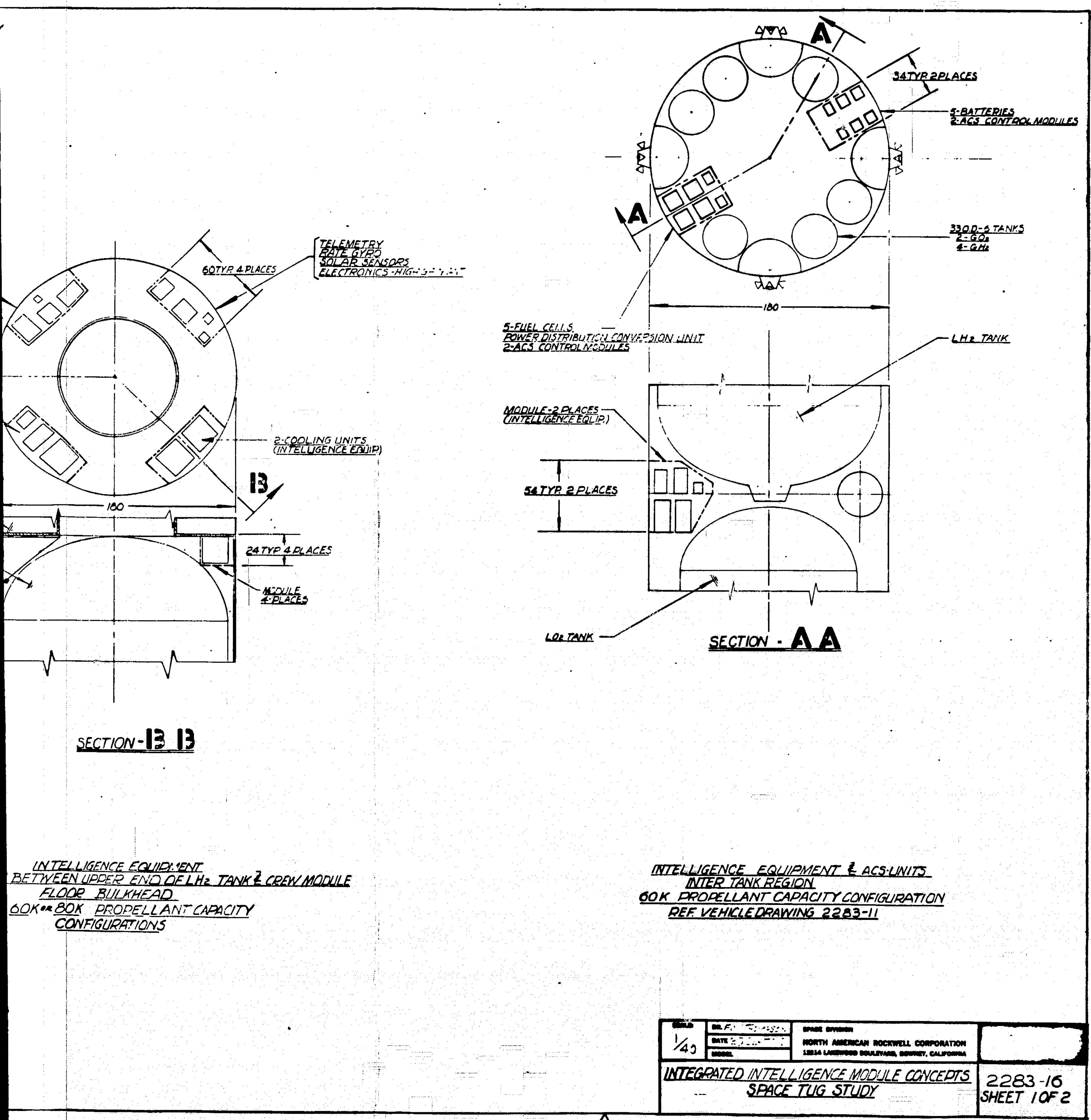
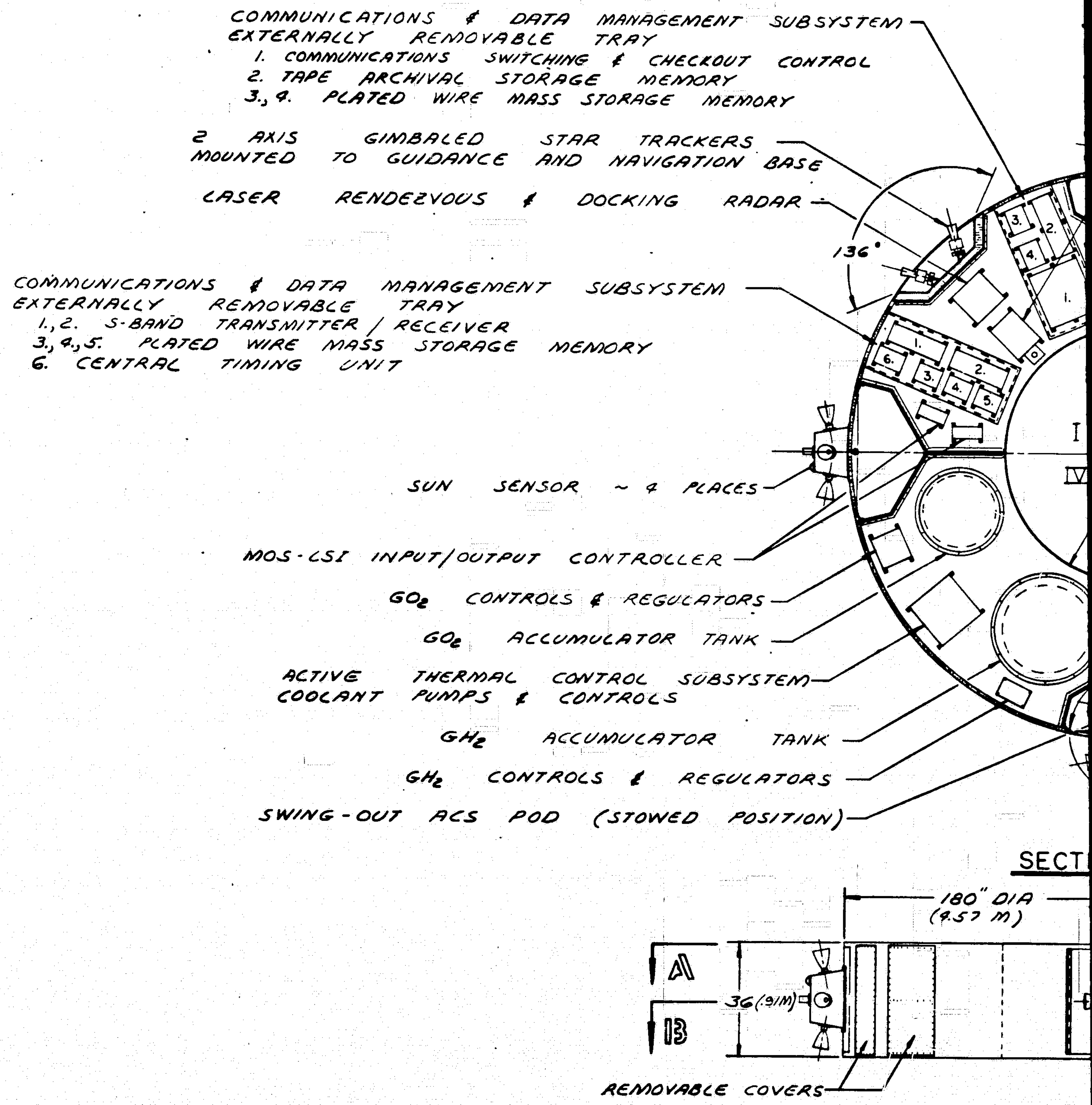


Figure 2-21. Integrated Intelligence Module Concepts


COMMUNICATIONS & DATA MANAGEMENT SUBSYSTEM
EXTERNALLY REMOVABLE TRAY
1. COMMUNICATIONS SWITCHING & CHECKOUT CONTROL
2. TAPE ARCHIVAL STORAGE MEMORY
3., 4. PLATED WIRE MASS STORAGE MEMORY
2 AXIS GIMBALED STAR TRACKERS
MOUNTED TO GUIDANCE AND NAVIGATION BASE
LASER RENDEZVOUS & DOCKING RADAR
136°
COMMUNICATIONS & DATA MANAGEMENT SUBSYSTEM
EXTERNALLY REMOVABLE TRAY
1., 2. S-BAND TRANSMITTER / RECEIVER
3., 4., 5. PLATED WIRE MASS STORAGE MEMORY
6. CENTRAL TIMING UNIT
SUN SENSOR ~ 4 PLACES
MOS-LSI INPUT/OUTPUT CONTROLLER
GO_2 CONTROLS & REGULATORS
GO_2 ACCUMULATOR TANK
ACTIVE THERMAL CONTROL SUBSYSTEM
COOLANT PUMPS & CONTROLS
GH_2 ACCUMULATOR TANK
GH_2 CONTROLS & REGULATORS
SWING-OUT ACS POD (STOWED POSITION)
SECT
180" DIA
(4.57 M)
A
B
36 (.91M)
REMOVABLE COVERS

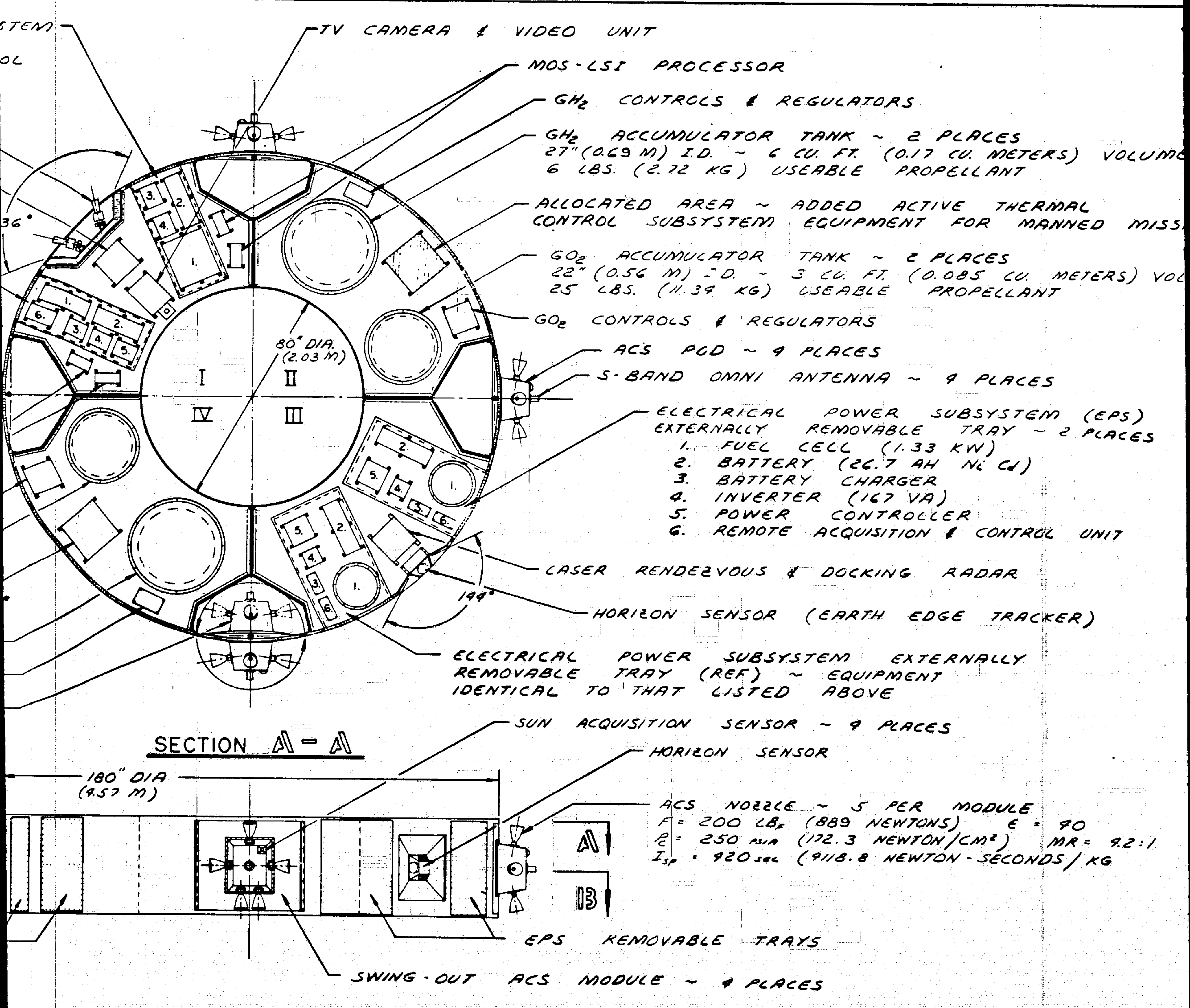


Figure 2-22. Inte

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/IT

PROCESSOR

ONTROLS & REGULATORS

CCUMULATOR TANK ~ 2 PLACES
M) I.D. ~ 6 CU. FT. (0.17 CU. METERS) VOLUME
(2.72 KG) USEABLE PROPELLANT

ED AREA ~ ADDED ACTIVE THERMAL
SUBSYSTEM EQUIPMENT FOR MANNED MISSION

CCUMULATOR TANK ~ 2 PLACES
6 M) I.D. ~ 3 CU. FT. (0.085 CU. METERS) VOLUME
(11.39 KG) USEABLE PROPELLANT

TROLS & REGULATORS

S POD ~ 4 PLACES

BAND OMNI ANTENNA ~ 4 PLACES

— ELECTRICAL POWER SUBSYSTEM (EPS)
EXTERNALLY REMOVABLE TRAY ~ 2 PLACES
1. FUEL CELL (1.33 KW)
2. BATTERY (26.7 AH Ni Cd)
3. BATTERY CHARGER
4. INVERTER (167 VA)
5. POWER CONTROLLER
6. REMOTE ACQUISITION & CONTROL UNIT

RENDEZVOUS & DOCKING RADAR

IZON SENSOR (EARTH EDGE TRACKER)

OWER SUBSYSTEM EXTERNALLY
RAY (REF) ~ EQUIPMENT
THAT LISTED ABOVE

ITION SENSOR ~ 4 PLACES

— HORIZON SENSOR

— ACS NOZZLE ~ 5 PER MODULE
$F = 200$ LB$_F$ (889 NEWTONS) $\epsilon = 40$
$P_c = 250$ PSIA (172.3 NEWTON/CM^2) MR = 4.2:1
$I_{sp} = 420$ SEC (4118.8 NEWTON-SECONDS/KG

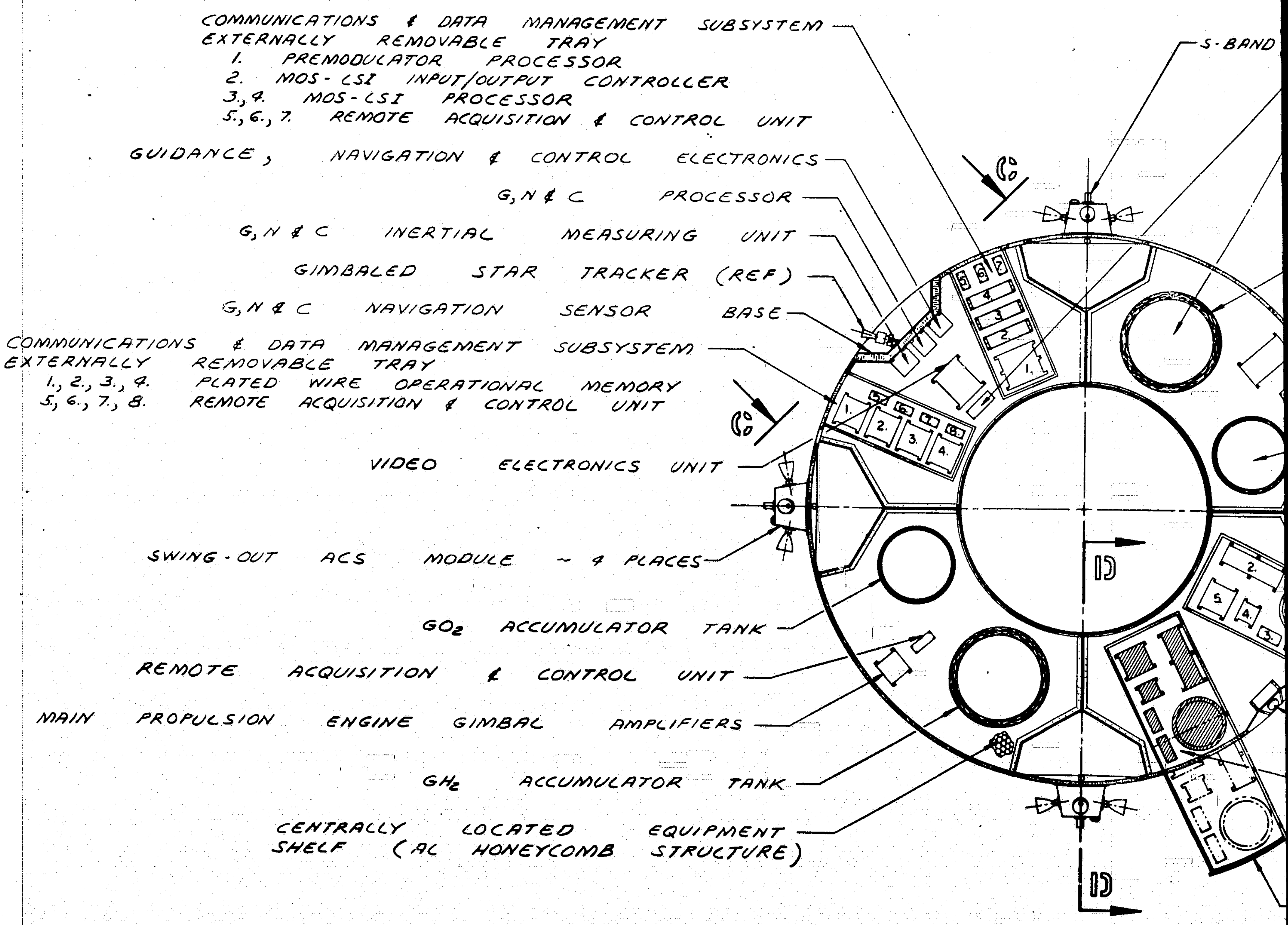
MOVABLE TRAYS

LE ~ 4 PLACES

SCALE 1/20	DR. SHOULDERS / DATE 12-30-70 / MODEL TUG	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
INTELLIGENCE MODULE GENERAL ARRANGEMENT SPACE TUG STUDY		2283-35A

Figure 2-22. Intelligence Module General Arrangement (Sheet 1 of 3)

T FRAME 2



SECTION B - B

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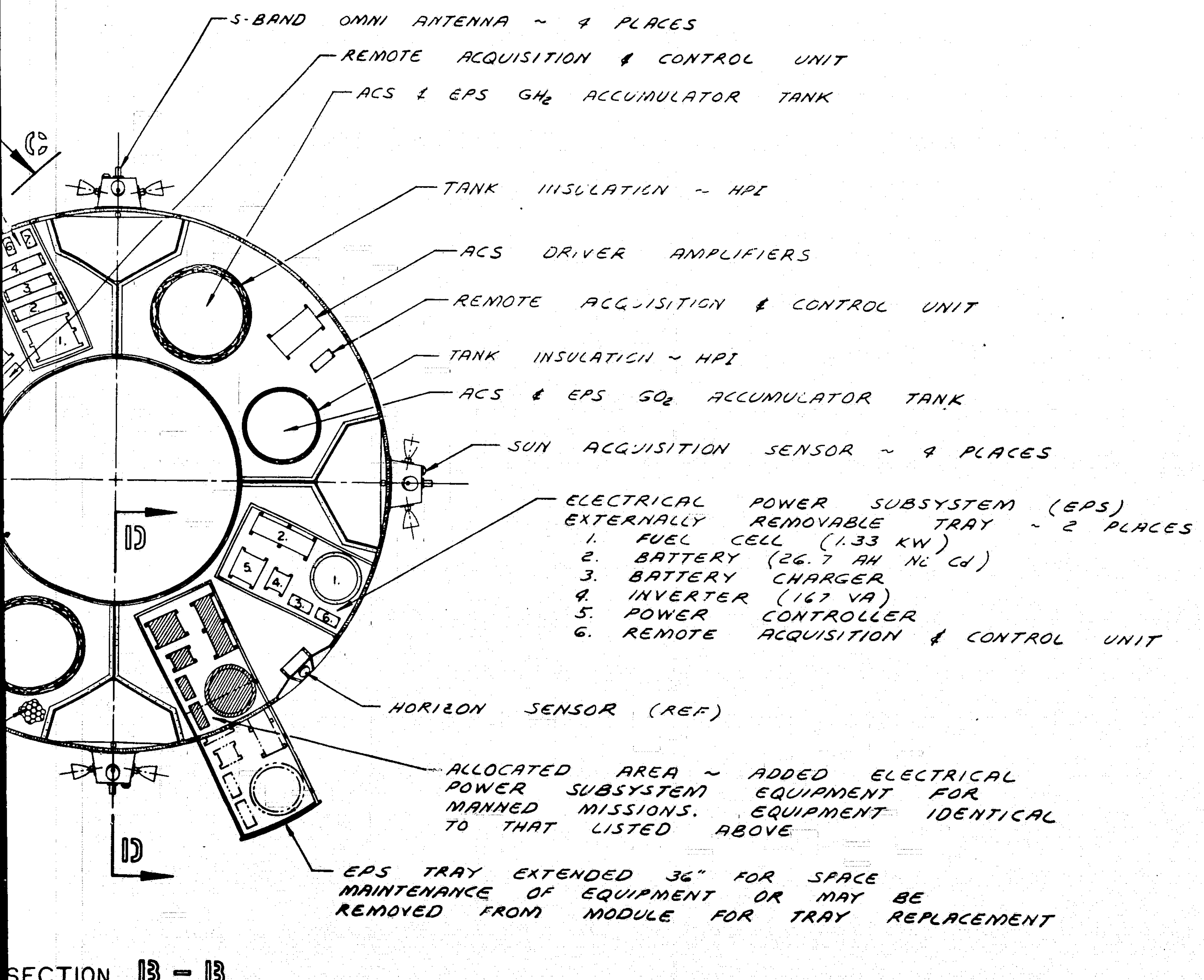

Figure 2-22. Intelligence Module General Arrangement (Sheet 2 of 3)

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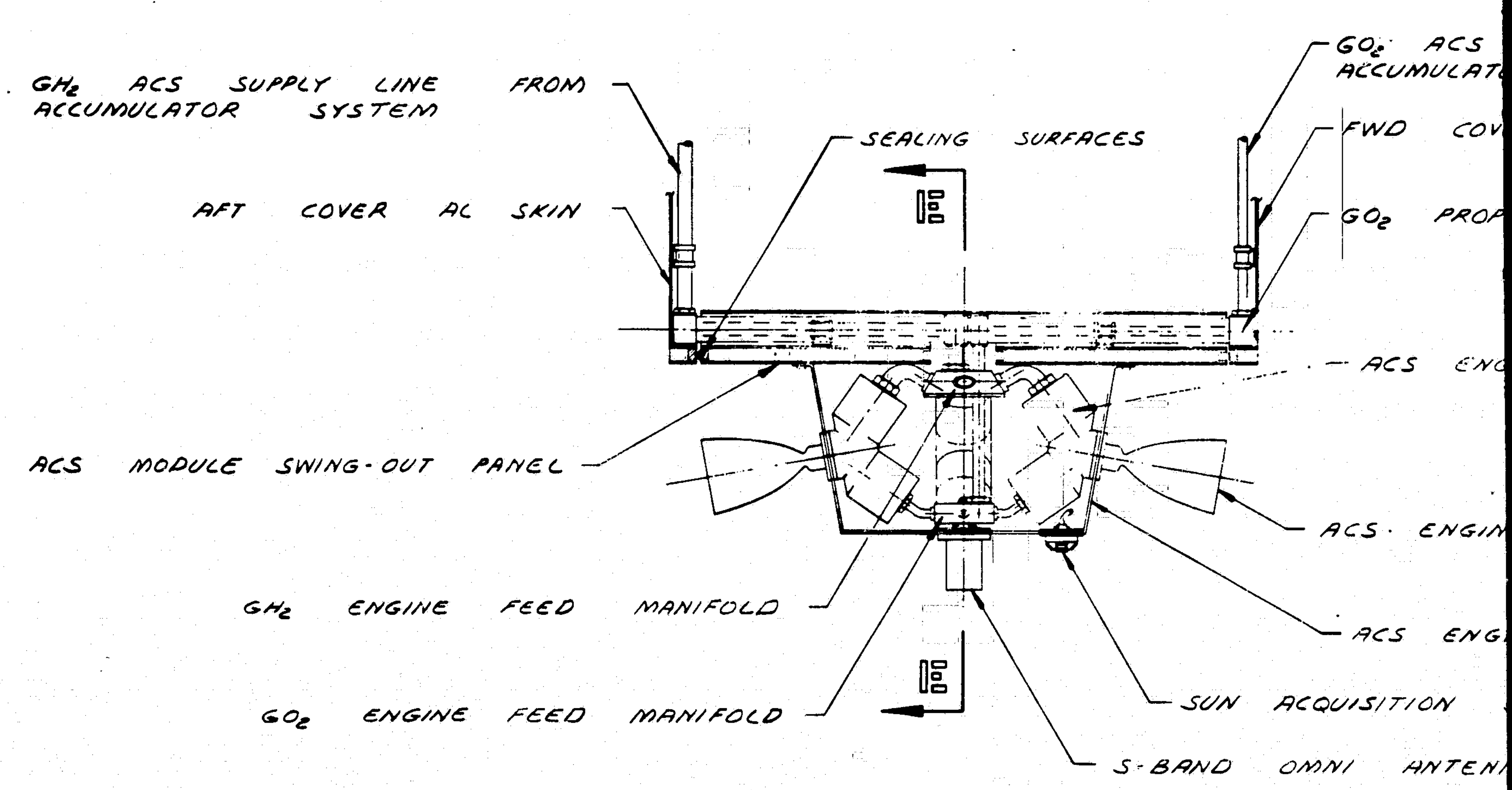
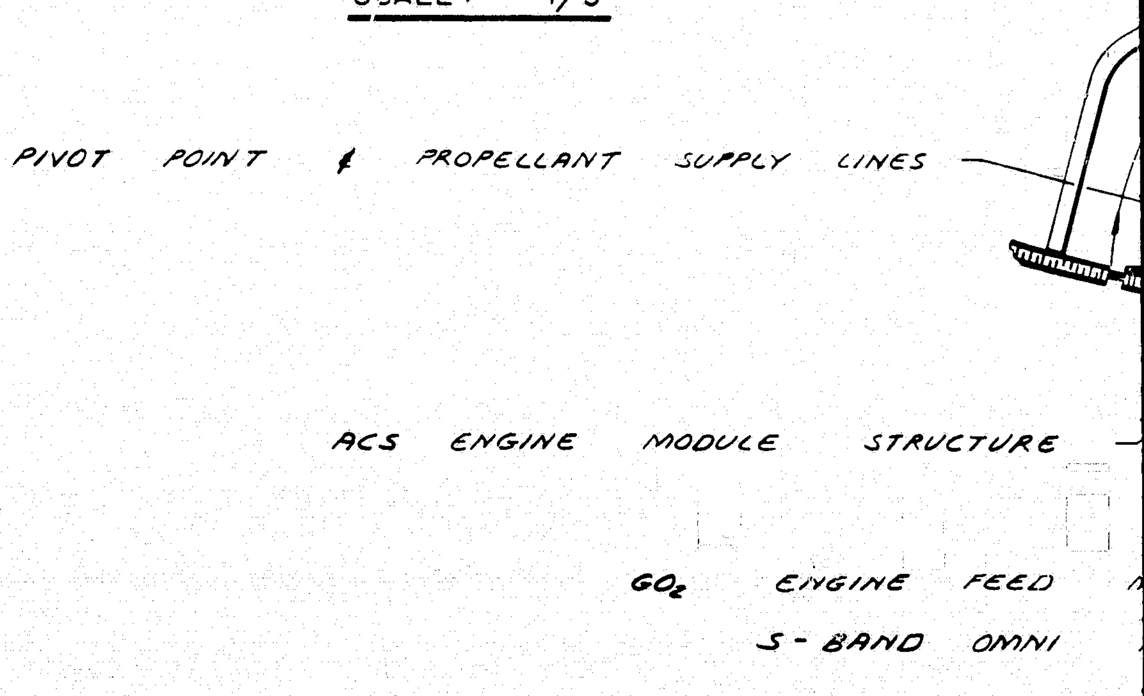


SECTION D – D
SCALE: 1/5



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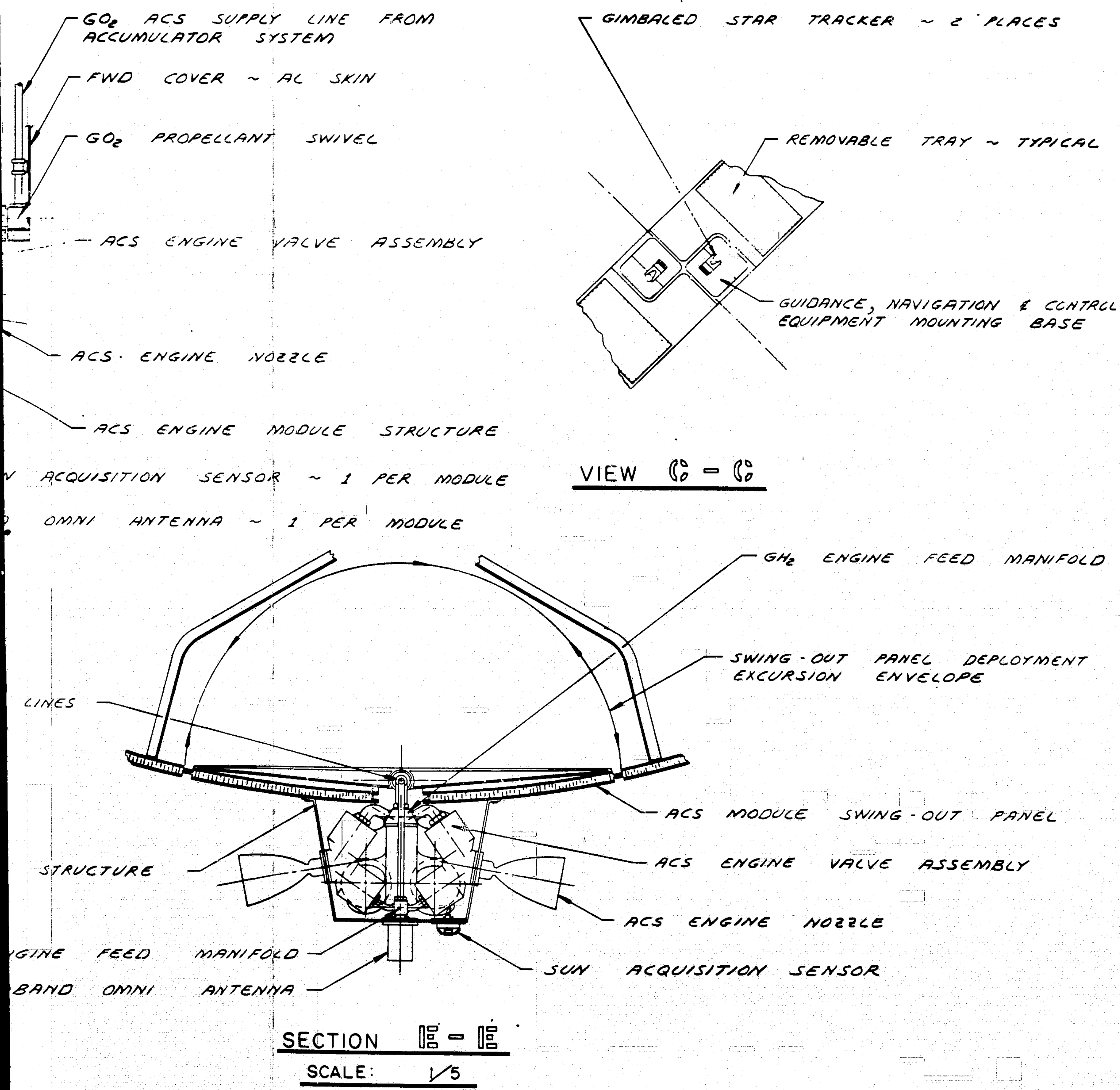


Figure 2-22. Intelligence Module General Arrangement
(Sheet 3 of 3)

FOLDOUT FRAME 2

400 pounds (181 kilograms) of structure. It is not known what the weight increase to the PM would be to handle the additional equipment heat loads and structural loads. In the performance of this study, the decision was made to continue with the separable module for baseline studies. As a result of this decision, an IM configuration was prepared that reflected all the latest updates of the equipment. This configuration was then used on all subsequent tug baseline vehicle configurations and refinements.

2.2.7 RADIATOR LOCATION STUDY

During the initial configuration studies of the orbital tug and the various operational derivatives, it was necessary to examine accommodation of the radiator systems on the vehicles from considerations of commonality and physical compatibility with the configurational characteristics of each vehicle. Preliminary analyses established tentative radiator areas for both the unmanned and the manned operational concepts. For the manned concepts, the radiator requirements for the orbital, as well as lunar lander, cases were defined. The radiator areas used in the configurational studies were 180 square feet (16.7 square meters) for the PM and an additional 150 square feet (14 square meters) to be added for the manned configurations. In this study, the full range of candidate vehicle diameters was considered. These included 22 feet (6.4 meters), 15 feet (4.57 meters), and 12 feet (3.65 meters). Figure 2-6 presents the radiator integration studies. Initial investigations showed that in most configurations the PM-mounted radiators could be located in at least two different areas. The radiators can be positioned aft on the PM cylindrical structure, as shown. For maximum compatibility between the orbital and the lander configurations, the radiators are divided into two panels and positioned in quadrants on opposite sides of the vehicle. This arrangement will permit the externally mounted cargo pods to be located in the other two quadrants. For the earth-orbital configuration, the same radiator arrangement will suffice. Another radiator arrangement is also shown. This concept features the radiators accommodated in a four-panel band around the PM at the forward end below the IM. To reduce impingement of the aft firing reaction control nozzles on the radiators, a nominal separation distance of about 36 inches has been shown. This separation distance in the same as used on the Apollo service module, which has proved satisfactory. The basic advantage of the circumferential radiator arrangement in that it provides a better viewing factor for the major radiator area, regardless of vehicle orientation. In addition, radiators are not influenced by the number or configuration pods that may be carried.

For the manned configurations, an additional 150 square feet (14 square meter) is required. Although additional area could be accommodated in the PM, there is sufficient area on the CM. This approach offers two basic advantages. The PM radiator structural arrangement is basically the same whether the vehicle is manned or unmanned. In addition, the radiators that are required for the CM can be integrated into the cylindrical surface and, being on the CM, results in simpler system interconnections between the radiators and the environmental control system located in the module. This would eliminate radiator system interfaces between the CM and the remainder of the PM.

For the 12-ft-diameter (3.65-m) vehicles, there is adequate surface space on the CM for the radiator area since the module basically features two decks with double the module length compared to the 15-ft-diameter (4.57-m) module. The aft-mounted CM vehicle arrangement requires that all of the radiators be mounted on the PM surface as shown. This is considered necessary because of the proximity of the swing-out engine nozzles to the side of the crew module and the attendant thermal environment.

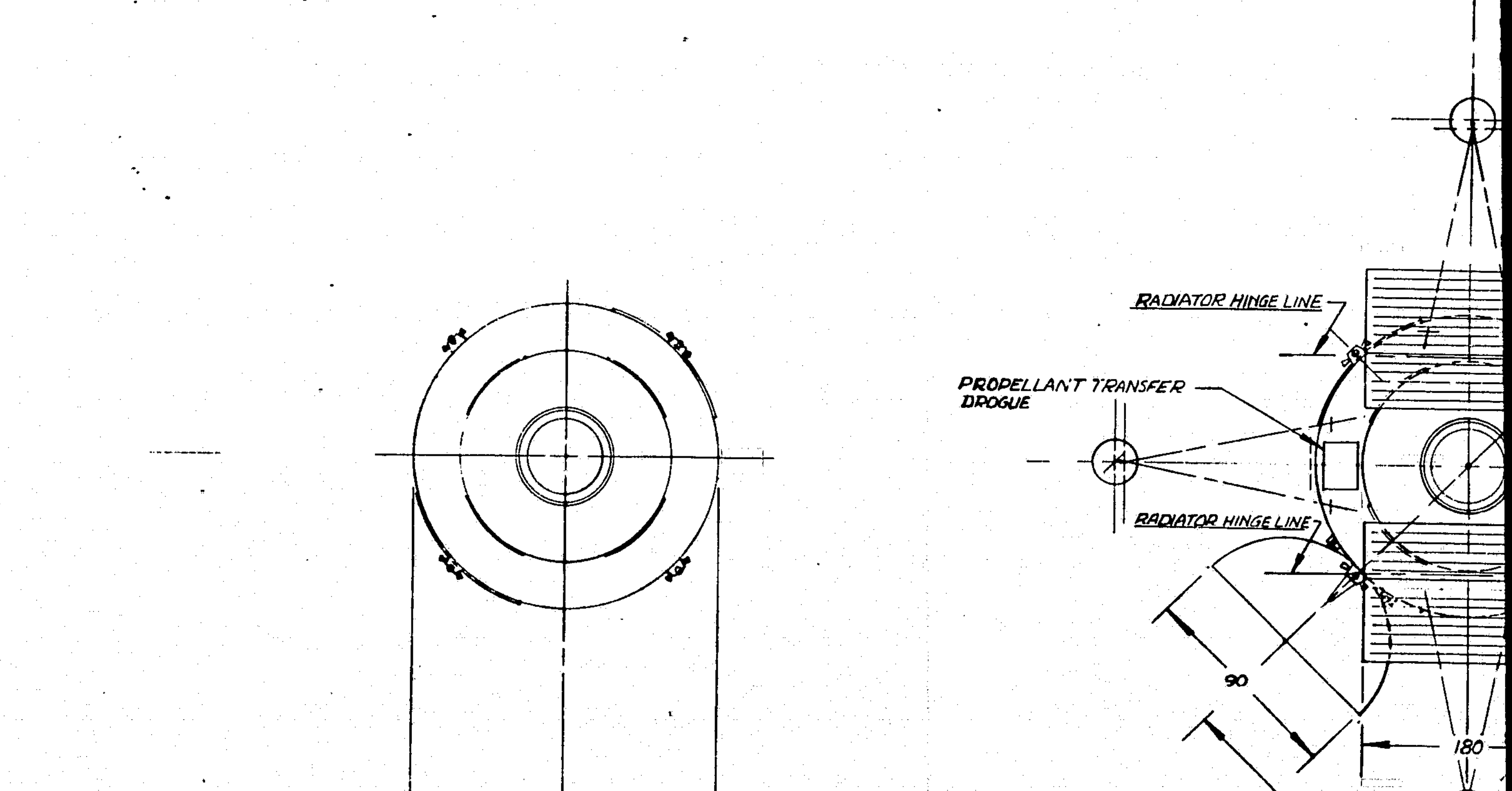
In the lunar lander vehicle configurations, analyses have shown that the radiators mounted on the sides of the vehicle are not effective because of the thermal input from the sunlit lunar surface. Consequently, separate deployable radiator panels would be required. These panels would be attached to the forward end of the module and deploy to be parallel to the lunar surface. The total required area of approximately 300 square feet (28.0 square meter) can be provided in either two or four panels, which can be folded to lie within the basic body diameter, as shown.

For the lander vehicle configurations that use two basic PM's attached in parallel to the sides of a crew/cargo module, the radiator system could be attached to the forward end of the two PMs, (Figure 2-23). The radiator panel would be added in earth orbit during the clustering and assembly of the modules into the lander configuration.

From this investigation of radiator integration, it was concluded that there are feasible alternatives for accommodating the radiator panels on the orbital tug and its lander configurations. The integration concepts are equally compatible with the three different diameters of candidate vehicles.

2.2.8 AEROSPIKE ENGINE AND DROGUE INTEGRATION

During the early conceptual studies of the tug, assessment of orbital operations indicated the desirability of equipping the vehicle with docking



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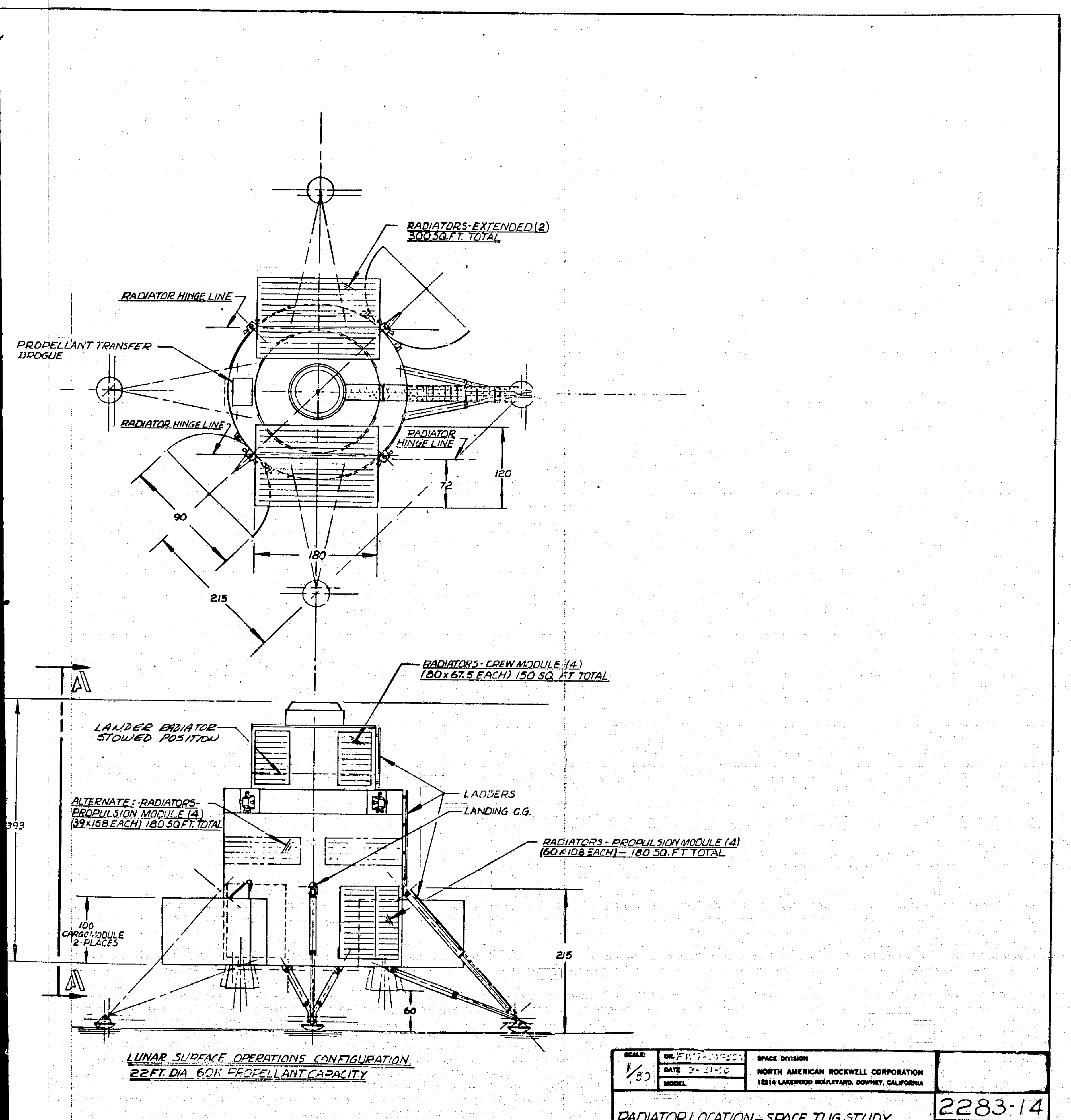

Figure 2-23. Radiator Location
(Sheet 1 of 3)

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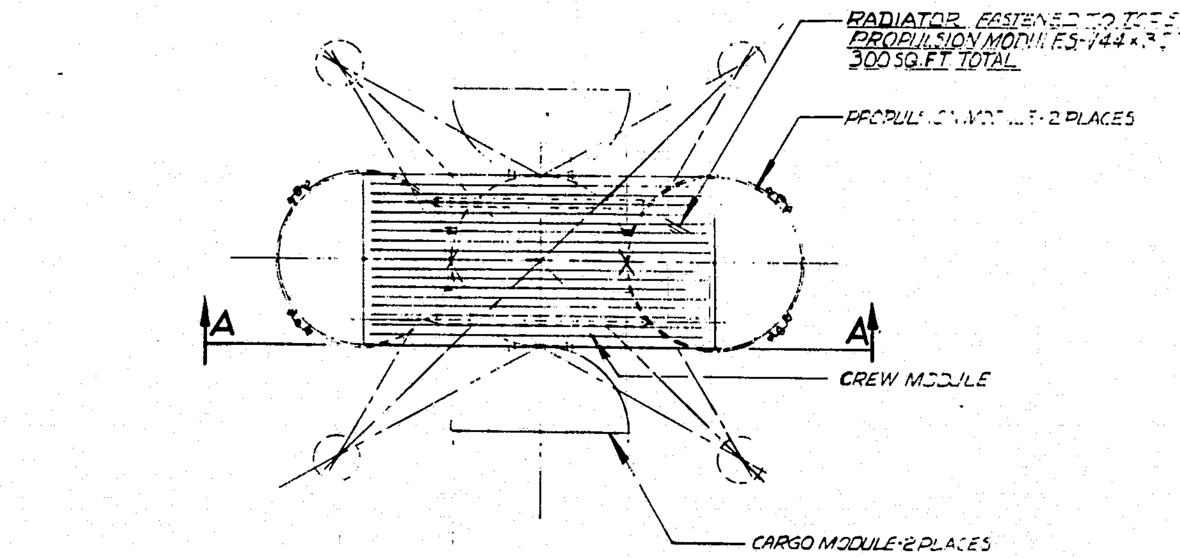
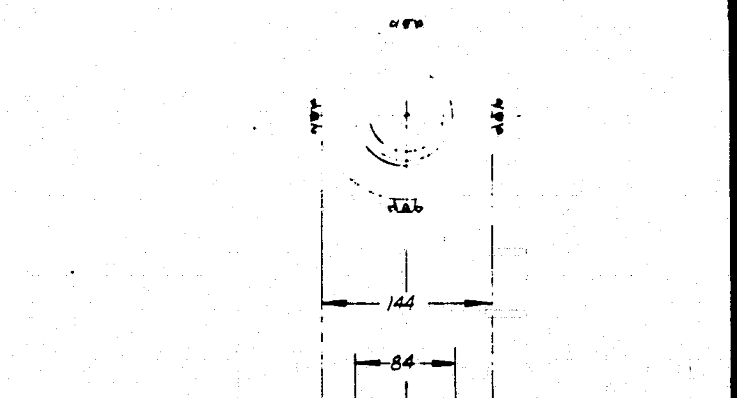


LUNAR SURFACE OPERATIONS CONFIGURATION – 12 FT. DIA. MODULES
60 K PROPELLANT CAPACITY (CLUSTERED 30K UNITS)



ORBITAL OPERATIONS CONFIGURATION
12 FT. DIA. MODULES
30K PROPELLANT CAPACITY (BASIC STAG

FOLDOUT FRAME 1

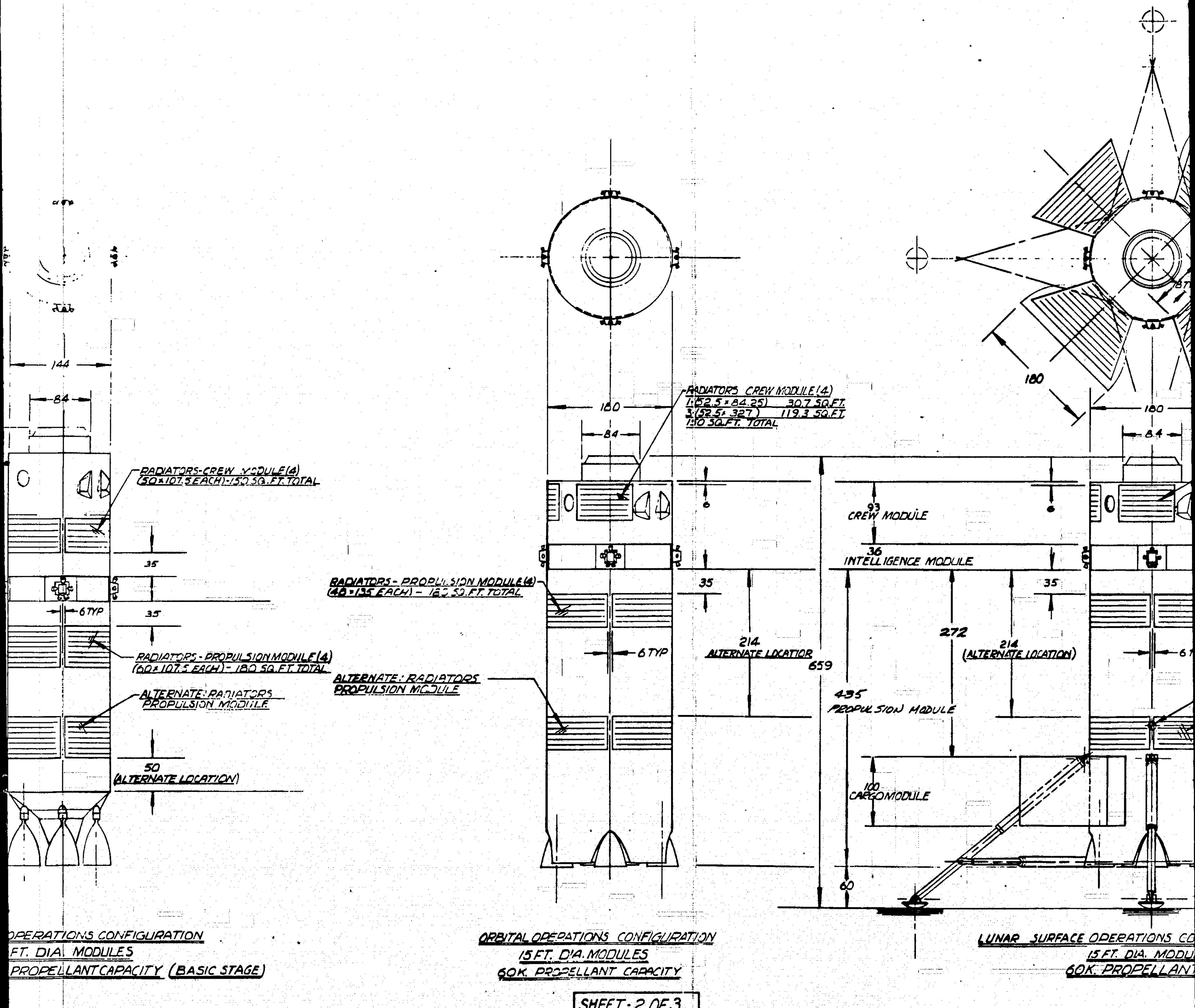


Figure 2-23. Radiator Loc
(Sheet 2 of 3)

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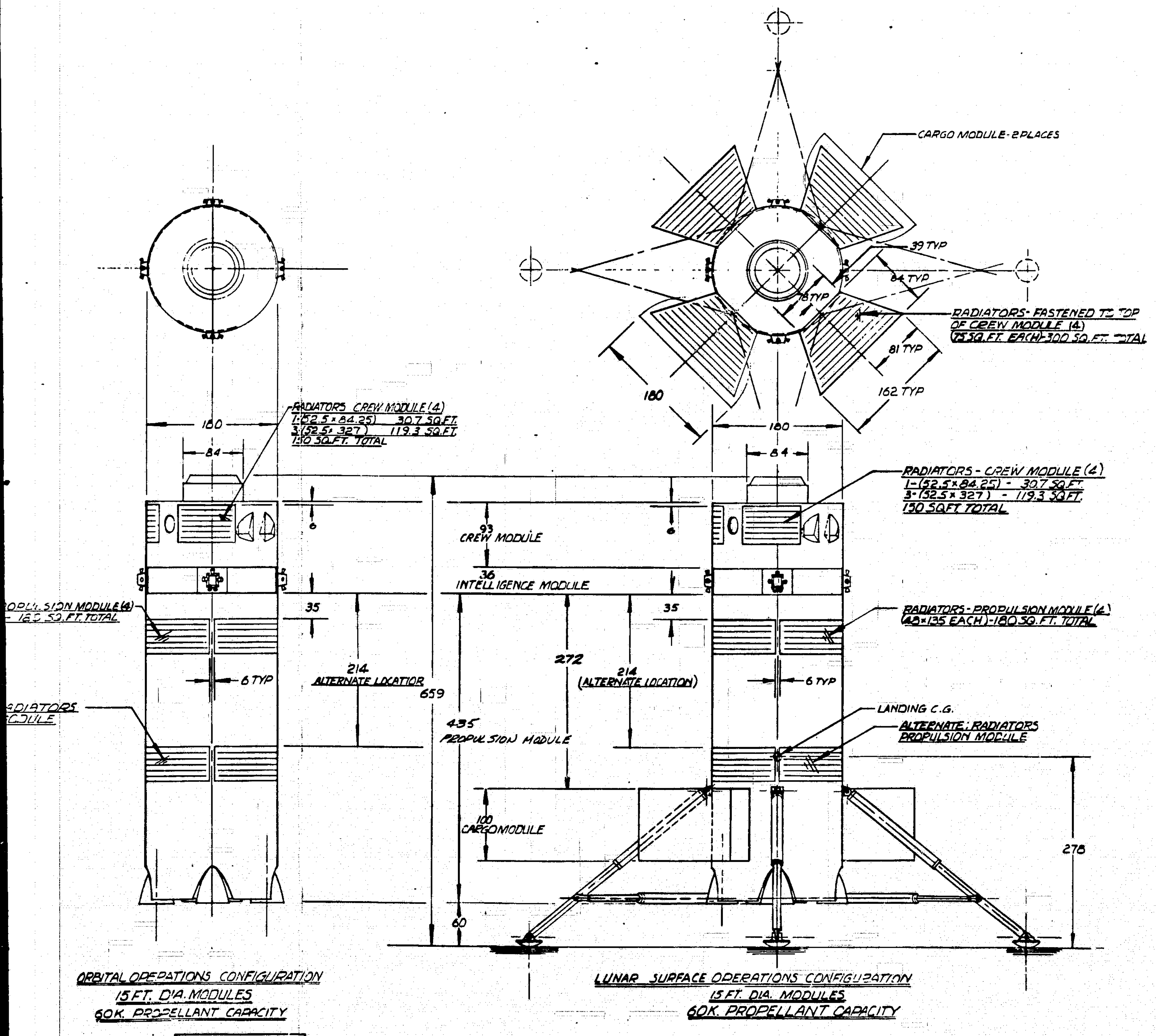
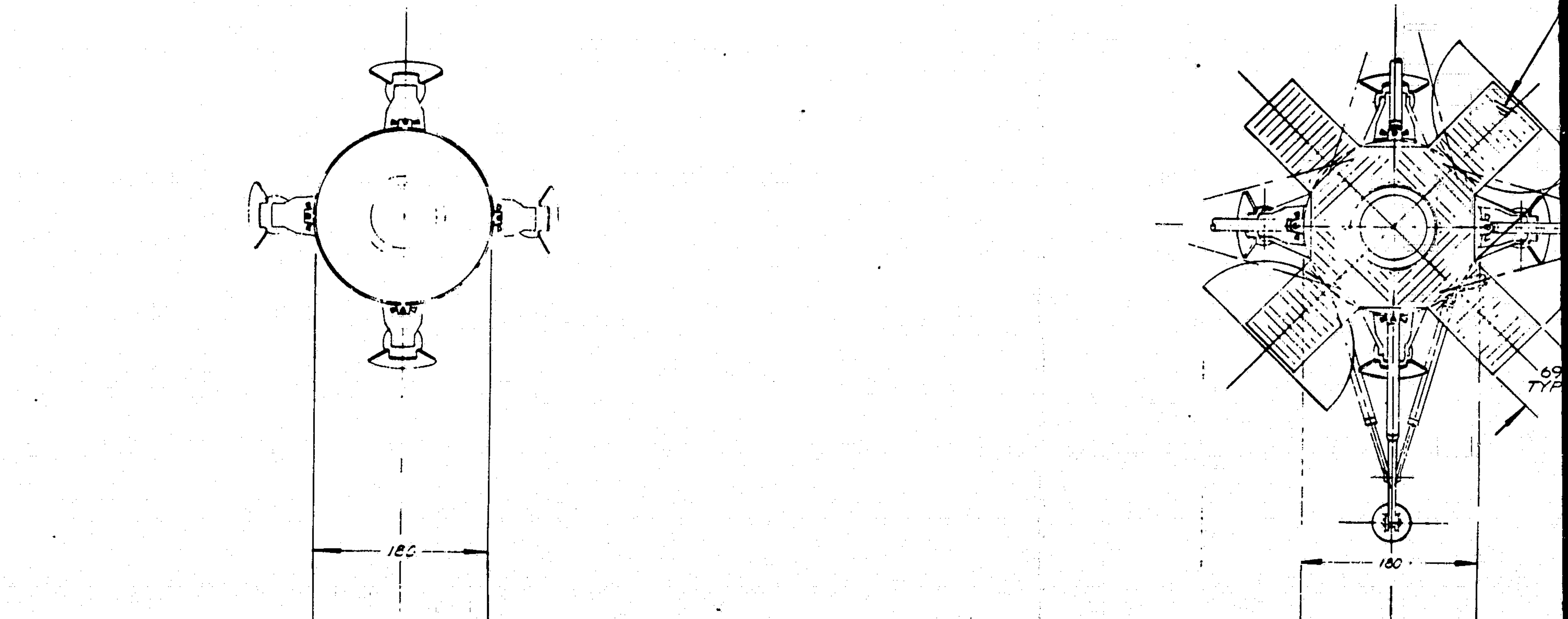


Figure 2-23. Radiator Location
(Sheet 2 of 3)



ORBITAL OPERATIONS CONFIGURATION - DEPLOYABLE ENGINES
15 FT. DIA. MODULES 60K PROPELLANT CAPACITY

LUNAR SURFACE OPERATIONS CONFIGURATION - DEPLOYABLE ENGINES
60K PROPELLANT CAPACITY

SHEET 3 OF 3

Figure 2-23. R
(Sheet

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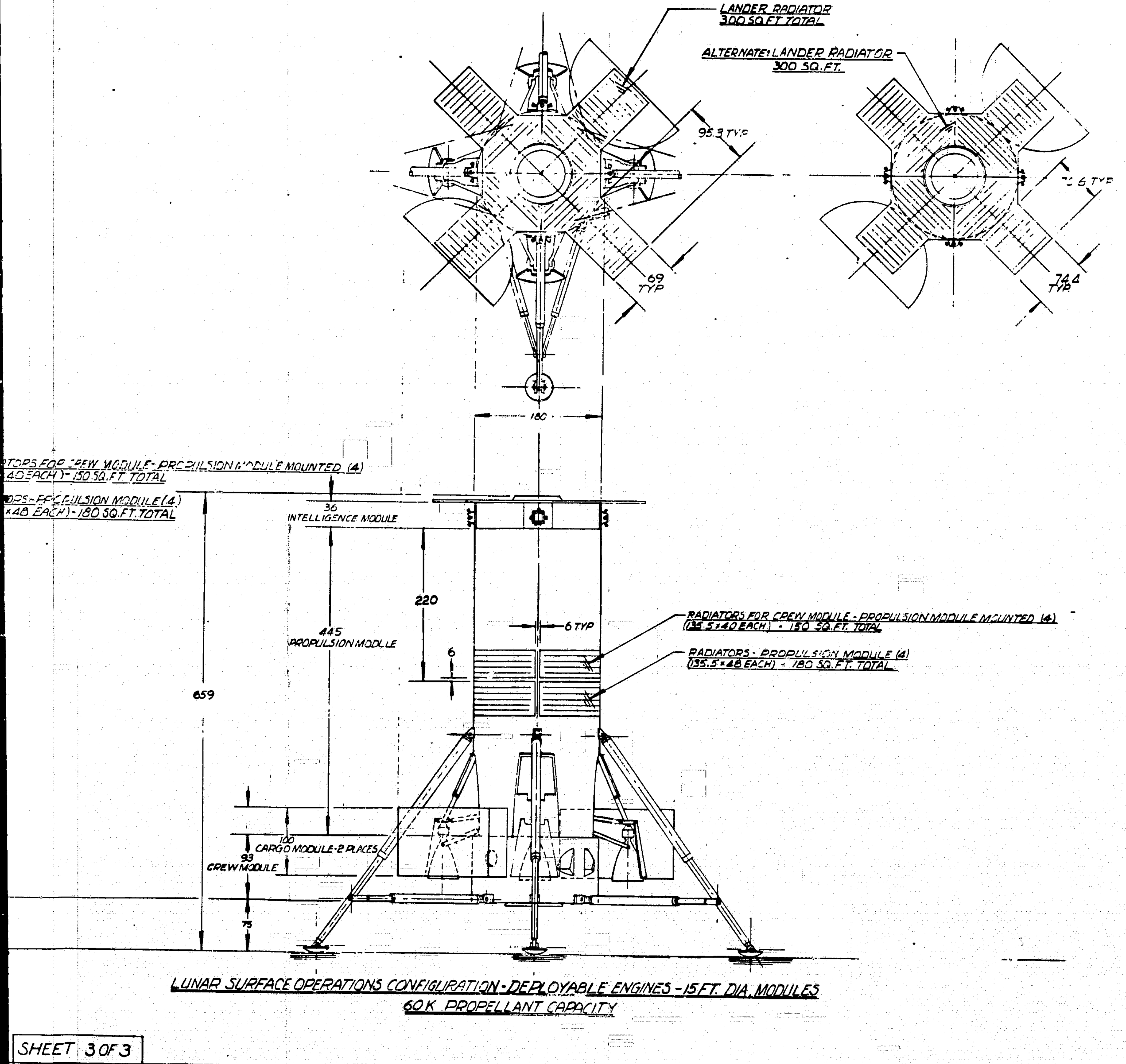
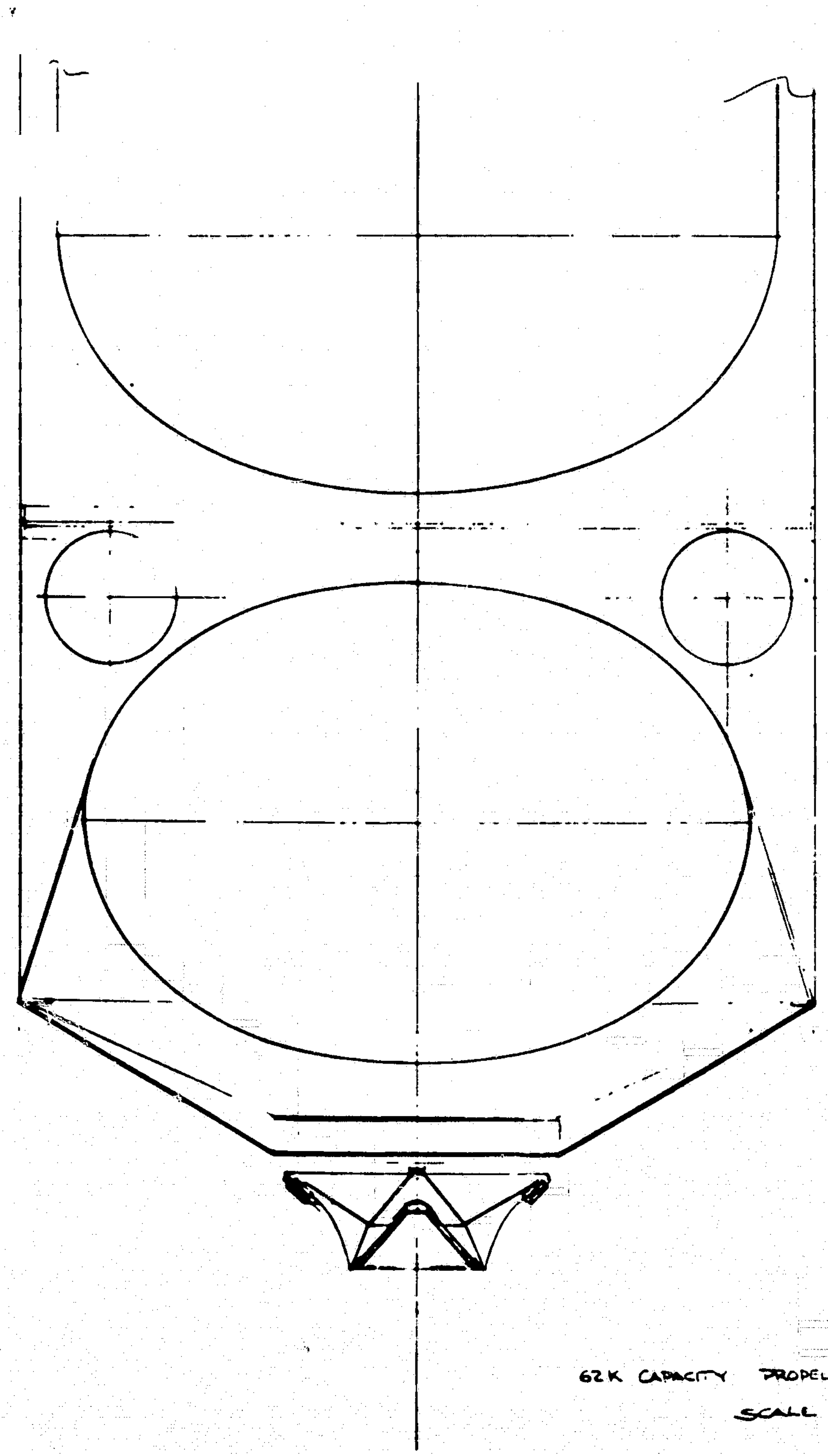


Figure 2-23. Radiator Location
(Sheet 3 of 3)

FOLDOUT FRAME 2

systems at both ends. Consequently, this capability has been incorporated in all of the baseline configurations. The systems used include the space-station-compatible passive ring with a 5-ft (1.52-m) clear center opening and a version of the small Apollo probe and drogue. The passive ring only has been used on the crew module when the tug is operating in the manned mode. In all other cases, the Apollo probe and drogue is used, with the probe positioned at the forward end of the vehicle and the drogue at the aft end.

Single-engined tug concepts present an obvious problem in accommodation of a docking mechanism on the aft end of the module. During the initial phase of the study, consideration was given to a docking mechanism with a diameter large enough to surround the engine nozzle and positioned near the nozzle exit plane. However, this docking system would be unique and not compatible with those currently considered in the major space program elements, including the orbital propellant depot, earth-orbital space station, RNS, etc. As a consequence, and concurrently with propulsion system studies that validated the same decision, multiple-engined configurations were adopted for the tug propulsion system. One single-engine configuration that would lend itself to a docking mechanism is the aerospike engine. Figure 2-24 illustrates a conceptual variation of an aerospike engine concurrently under study by Rocketdyne. It would, potentially, permit the integration of an Apollo-type drogue within the conical portion of the engine. On the nominal engine, the spike is closed out by a concave section with openings through which the pump turbine exhaust gases exit into the engine exhaust wake. A design investigation suggested that, by rearranging the turbines and pumps, it would be possible to integrate the drogue structure and have it suffice as the end closure for the spike. It would be fabricated from high-temperature material, and the peripheral ring would be designed to serve as the exit manifold for the turbine exhaust. Since the docking interface is incorporated in the engine, the docking loads and bending moments would have to be accommodated by structural additions. This could be accomplished by the use of retractable struts that would lock the engine in the neutral position or by increasing the load capacity of the engine gimbal actuators. As a result of this preliminary evaluation, it is considered to be a feasible concept and one that could possibly be implemented with the aerospike engine configuration. However, a design study delving into the additional requirements imposed upon the engine design and their consequences would be required as a first step in further consideration of this concept.


62K CAPACITY PROPELLANT
SCALE 1/20

ROPELLANT
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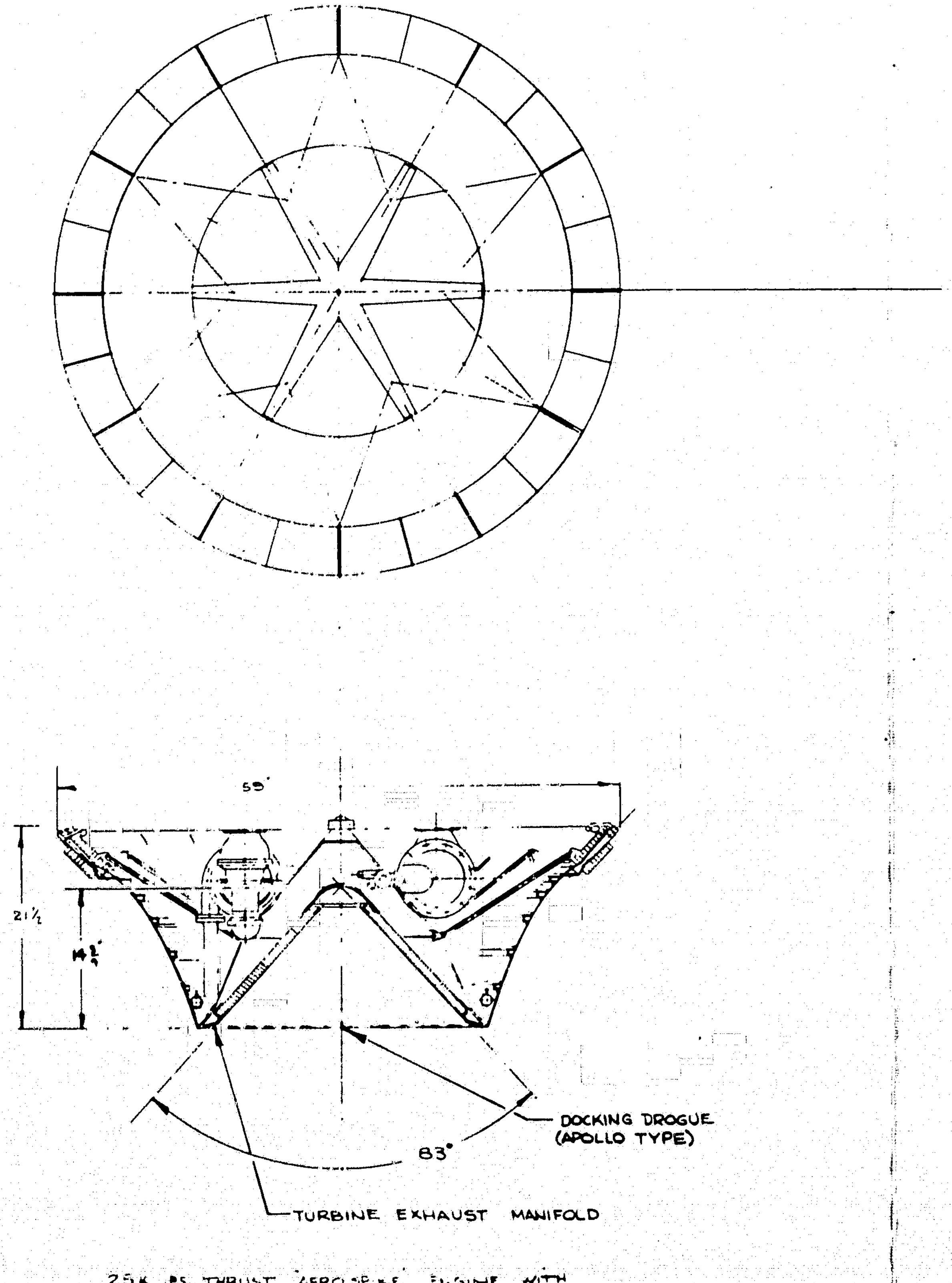


25K LBS THRUST AEROSPIKE ENGINE WITH INTEGRAL PASSIVE DOCKING DROGUE.

SCALE 1/8

Figure 2-24. Aero

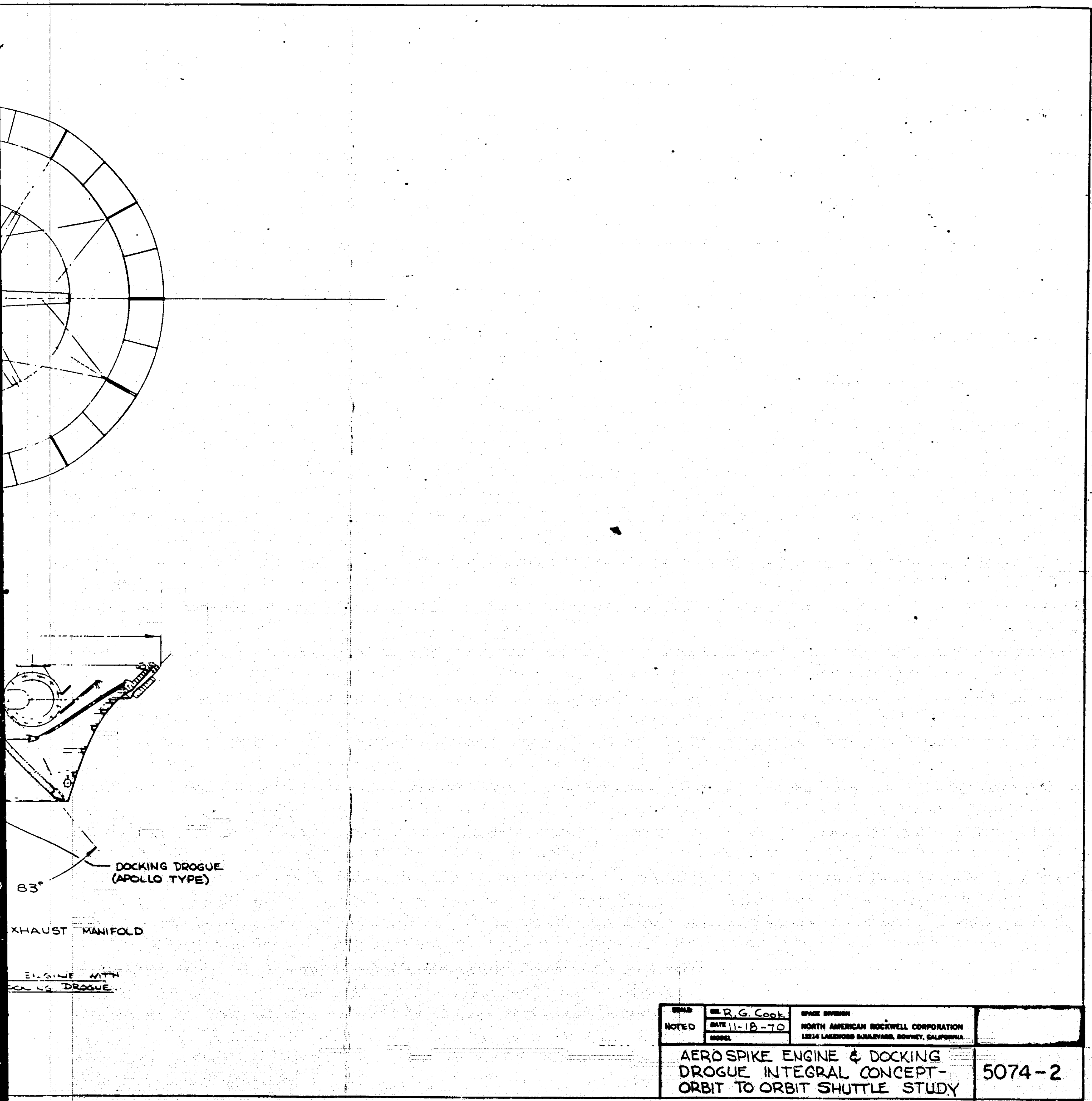


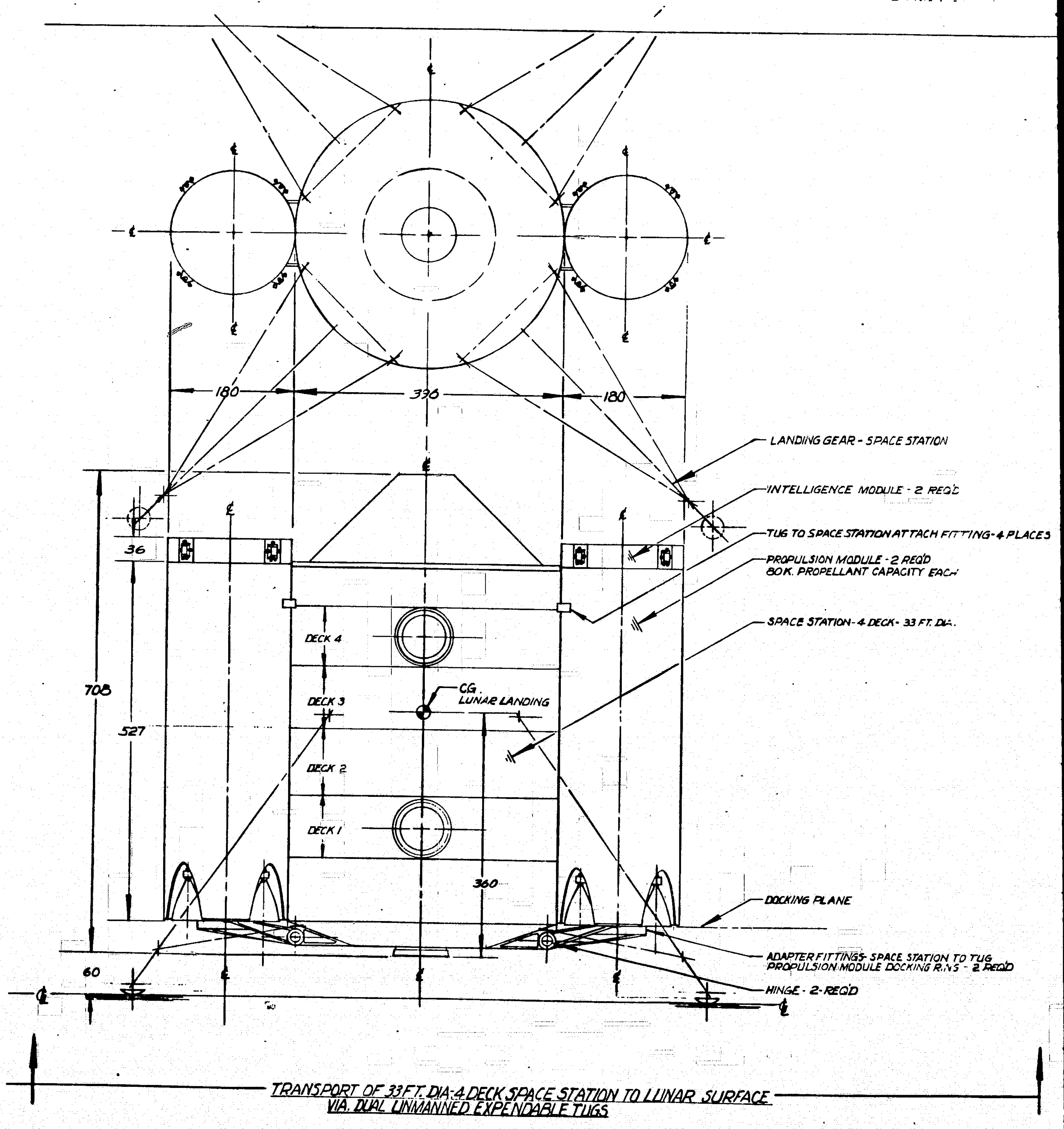
Figure 2-24. Aerospike Engine and Docking Drogue Integral Concept

2.2.9 LUNAR SURFACE BASE PAYLOAD ACCOMMODATION

Conceptual design studies were conducted to examine the configurational approaches that could be used in landing various types of lunar surface base shelter modules by the three selected tug lander concepts. In these studies the spectrum of surface base modules included 15-ft-diameter (4.57-m) modules, which would be representative of single-floor-level shelters similar to the tug CM and larger surface base modules, which included both a two-deck and full four-deck versions of the current NR 33-ft-diameter (10 m) space station core module. (These studies were also of interest to the study treams at NR concurrently engaged in the Lunar Base study for NASA/MSFC and the Orbiting Lunar Station Study for NASA/MSC.)

Three draw' 'rs were generated that illustrate the surface base payload accommodation fo. the single-stage (Concept 1), two-stage (Concept 5), and 1-1/2-stage (Concept 11) tug configurations. They are shown in Figures 2-25, 2-26, and 2-27, respectively.

The largest surface base payload considered was the 33-ft (10-m) NR space station core module. The lunar landing payload capability of a pair of single- or 1-1/2-stage tug PM's is adequate to accommodate the complete four-deck core module. The conceptual approach presented considers that a pair of tug PM's could be attached to the sides of the core module as part of an orbital assembly operation. The normal tug landing gear would not be used because it would be physically incompatible with the core module in the strap-on clustering mode. Also, the gear tread would not satisfy the overturn requirements resulting from the higher landing c.g. of the combined vehicle. A feasible approach is to consider that the landing gear assembly is ground-assembled and launched with the core module. Since the core module would be launched by a derivative of the Saturn V launch vehicle, the landing gear could be retracted to stow within the aft interstage adapter, and the longitudinal struts of the gear folded along the side of the core module could be covered with jettisonable boost fairing covers. In addition, it is considered feasible to incorporate the structural support system for the tug propulsion modules into the aft skirt area of the core module with the landing gear. These two support arms would be hinged to stow within the basic diameter of the core module and would be deployed in orbit along with the landing gear. Each support arm assembly would incorporate an active docking mechanism compatible with the selected passive system in the aft end of the tug propulsion module. To facilitate the assembly operation, the docking interface on the support arm would be hinged to permit the tug propulsion module to be brought in at an angle to the core module. After it



TRANSPORT OF 33 FT. DIA. 4 DECK SPACE STATION TO LUNAR SURFACE VIA. DUAL UNMANNED EXPENDABLE TUGS

Figure 2-25. 80

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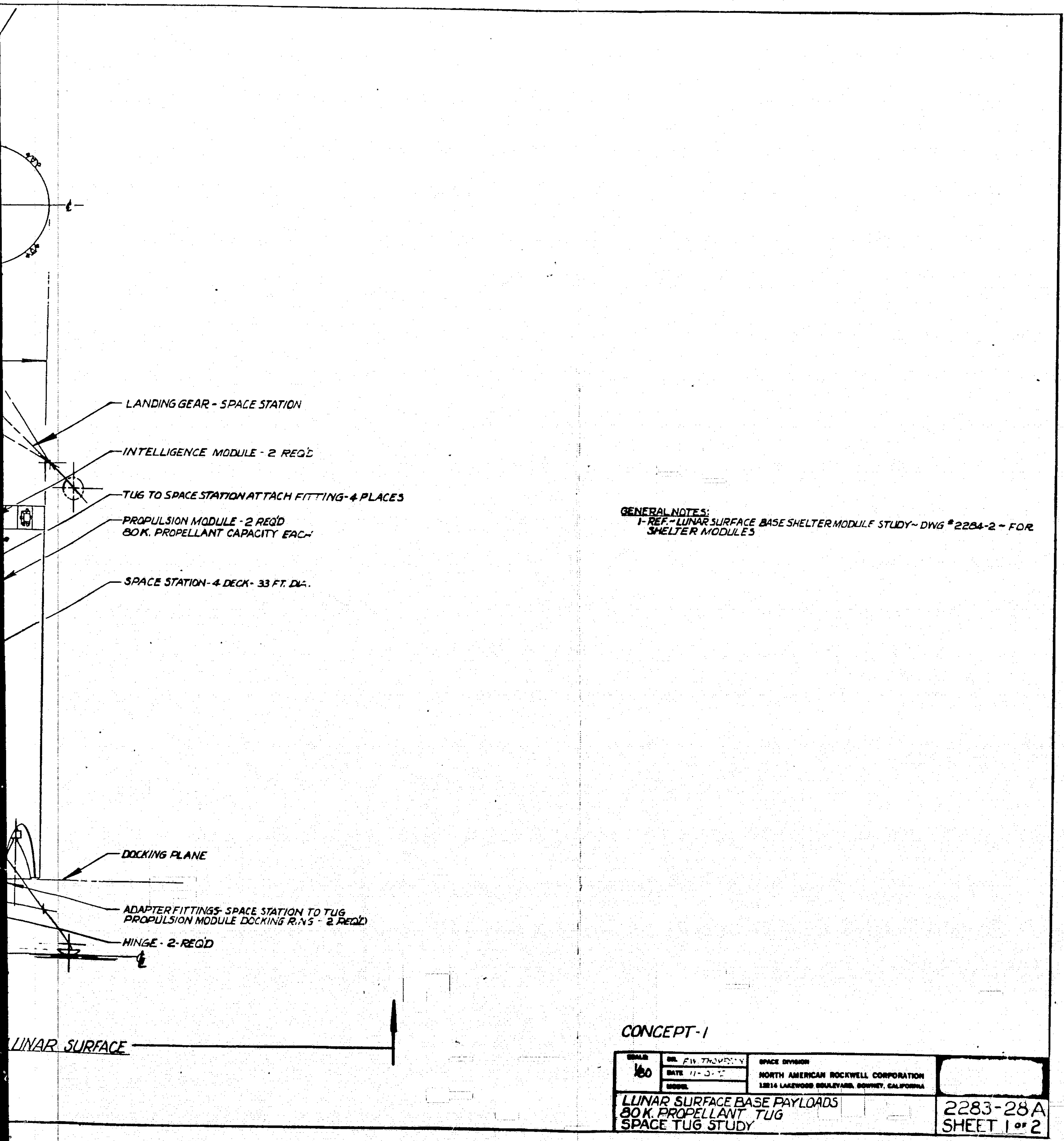

Figure 2-25. 80K Pound Propellant Tug Lunar Surface Base Payloads (Sheet 1 of 2)

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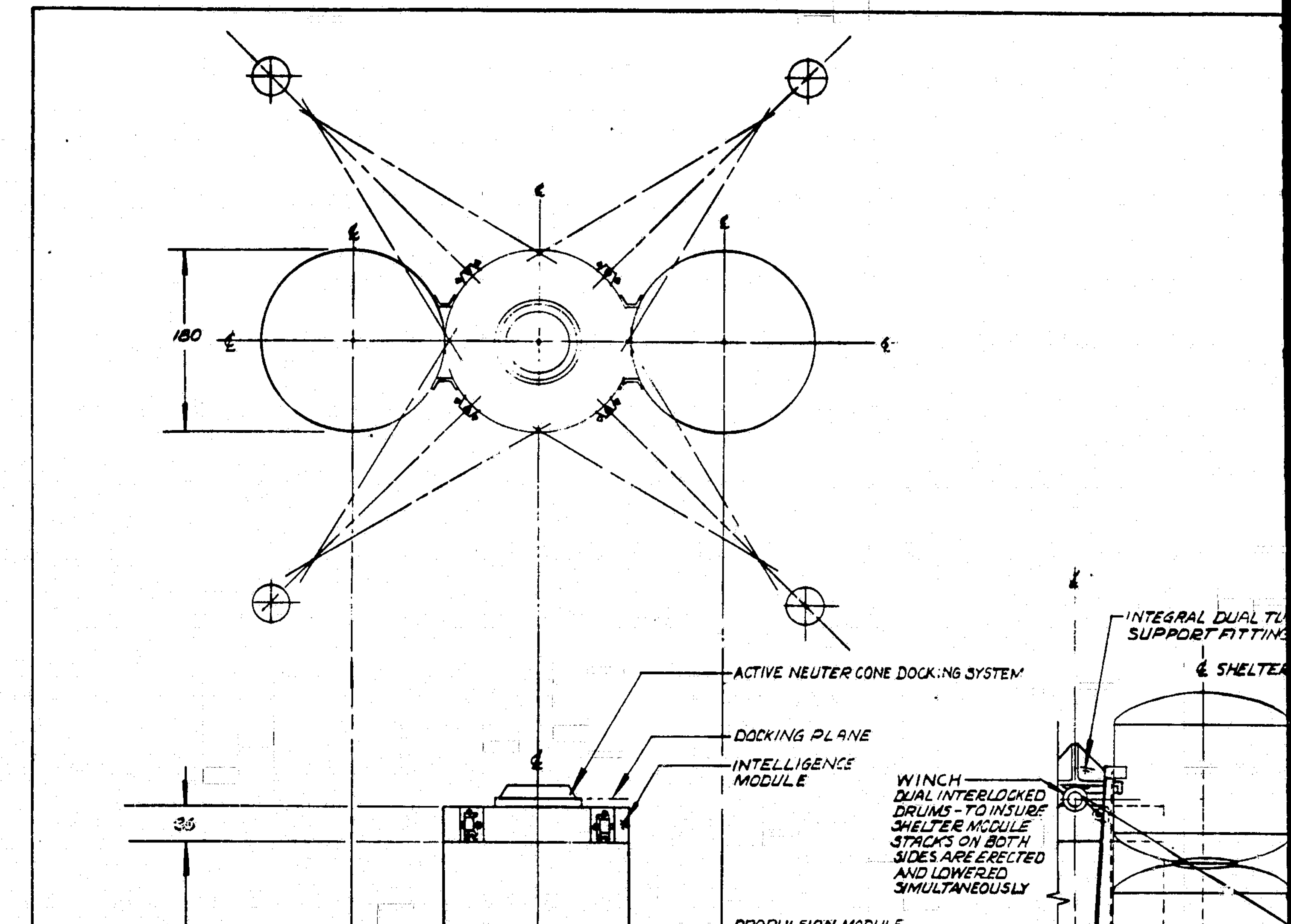

180
ACTIVE NEUTER CONE DOCKING SYSTEM
DOCKING PLANE
INTELLIGENCE MODULE
36
PROPULSION MODULE
80K PROPELLANT CAPACITY
ATTACH FITTINGS-8 REQ'D
SHELTER MODULES TO TUG
527
167 TYP
MATING PLANE
SHELTER MODULES
60
WINCH
DUAL INTERLOCKED DRUMS-TO INSURE SHELTER MODULE STACKS ON BOTH SIDES ARE ERECTED AND LOWERED SIMULTANEOUSLY
BOOM
CRADLE HINGE
TRANSPORT OF 4 SHELTER MODULES TO LUNAR SURFACE VIA UNMANNED RECOVERABLE TUG

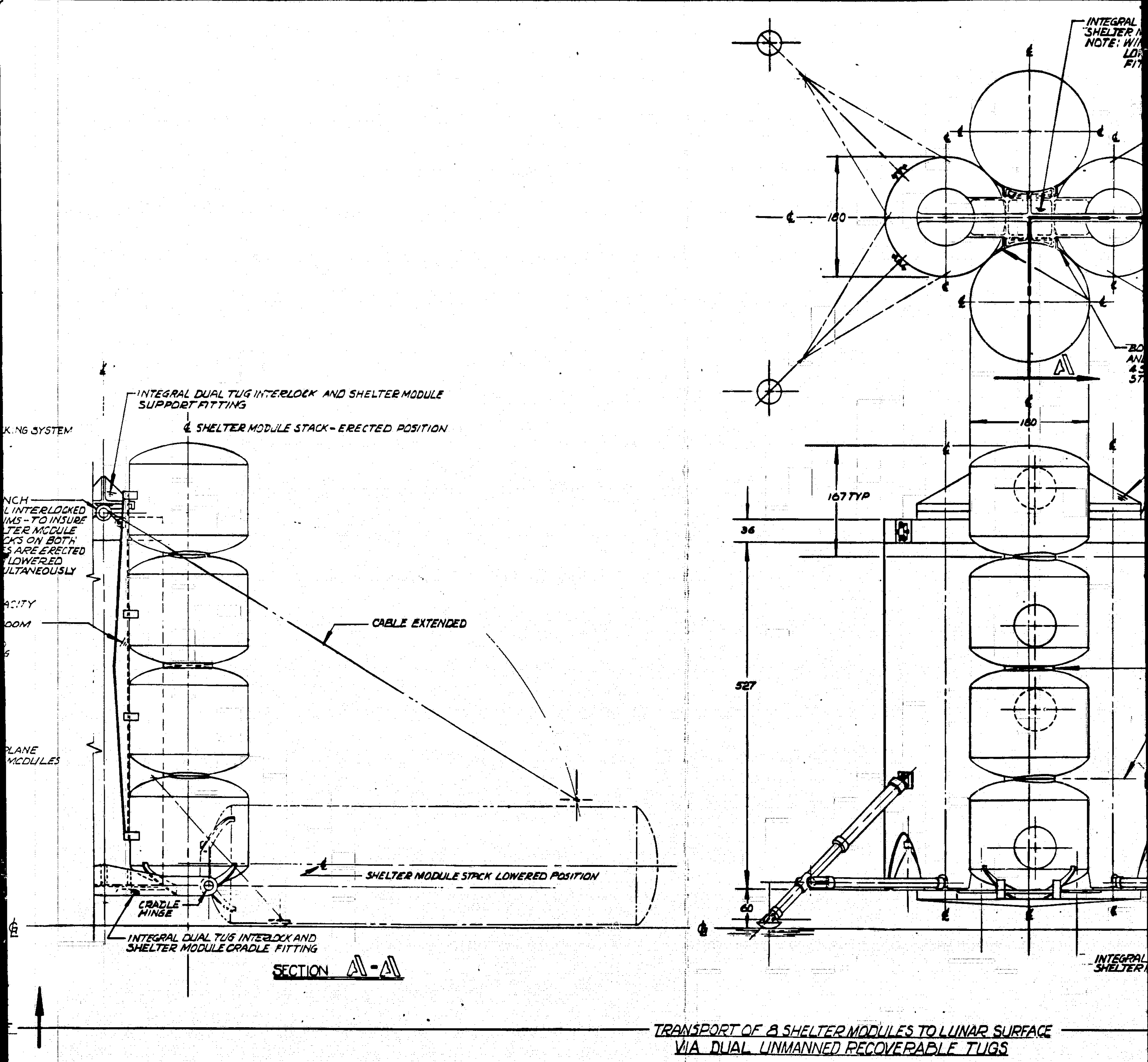

Figure 2-25. 80K Pound Propellant Tug Lu
(Sheet 2 of 2)

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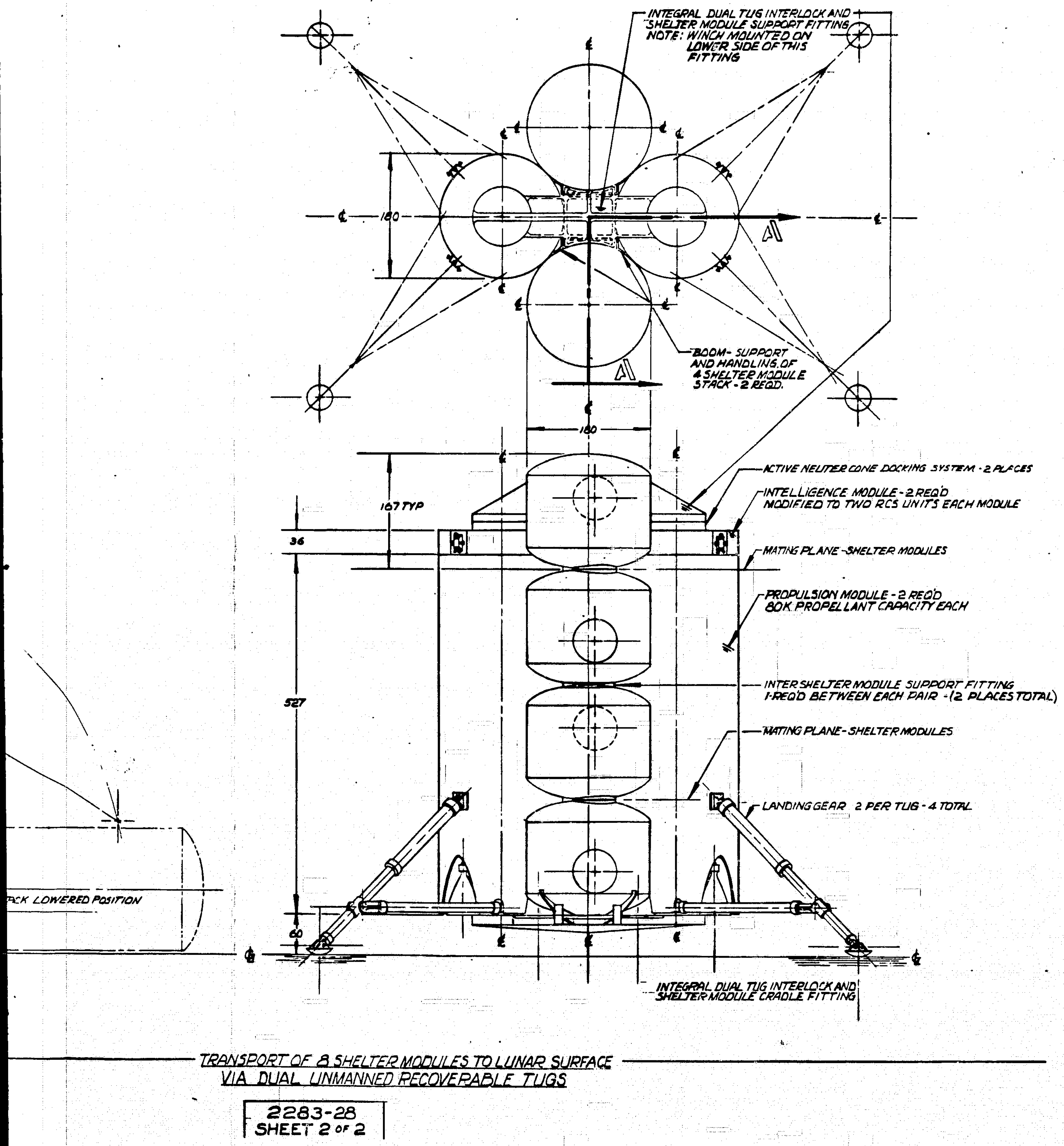
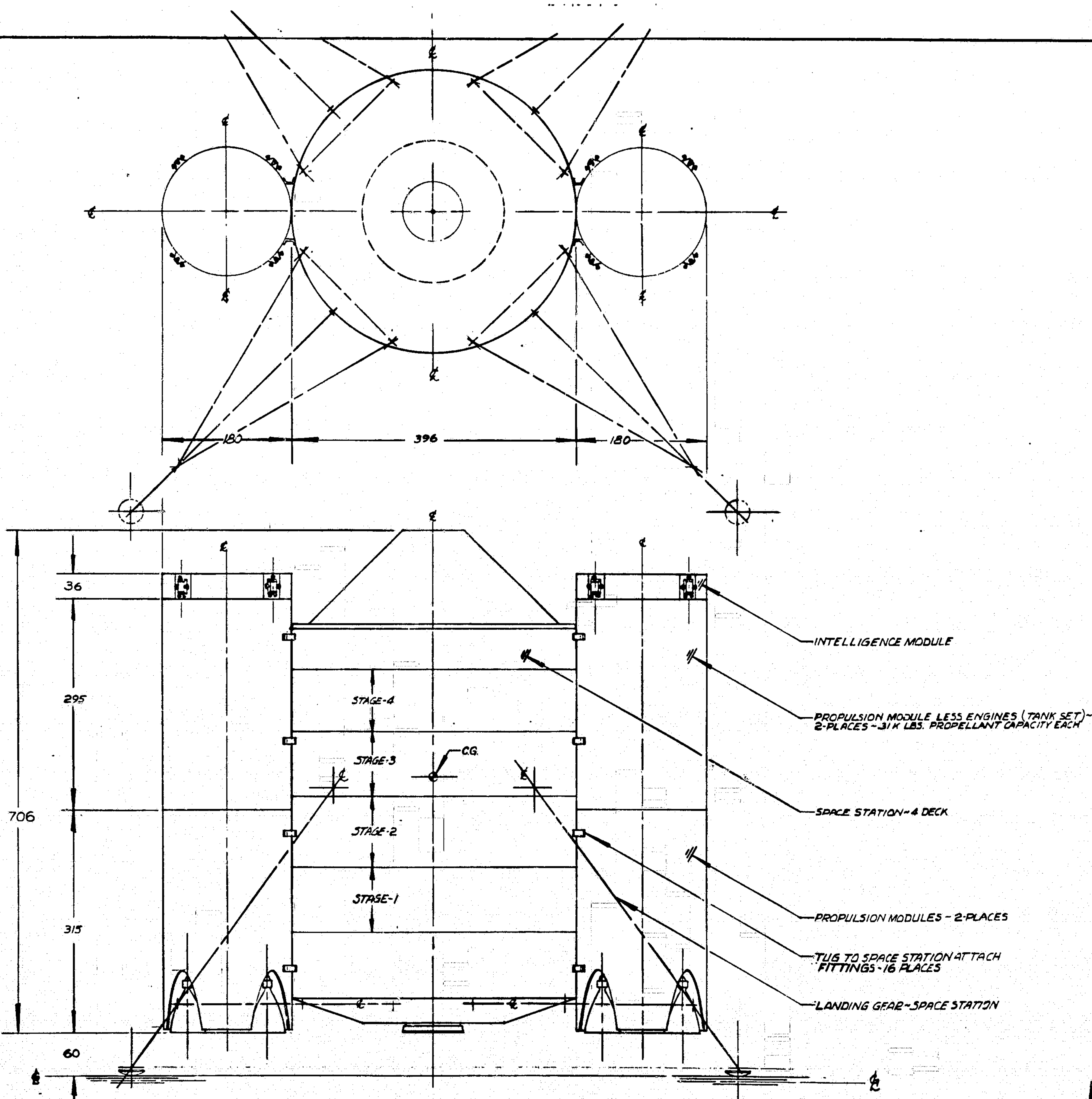


Figure 2-25. 80K Pound Propellant Tug Lunar Surface Base Payloads (Sheet 2 of 2)



TRANSPORT OF 33-FT DIA 4-DECK SPACE STATION TO LUNAR SURFACE VIA DUAL UNMANNED EXPENDABLE TUGS WITH TANDEM TANK SETS

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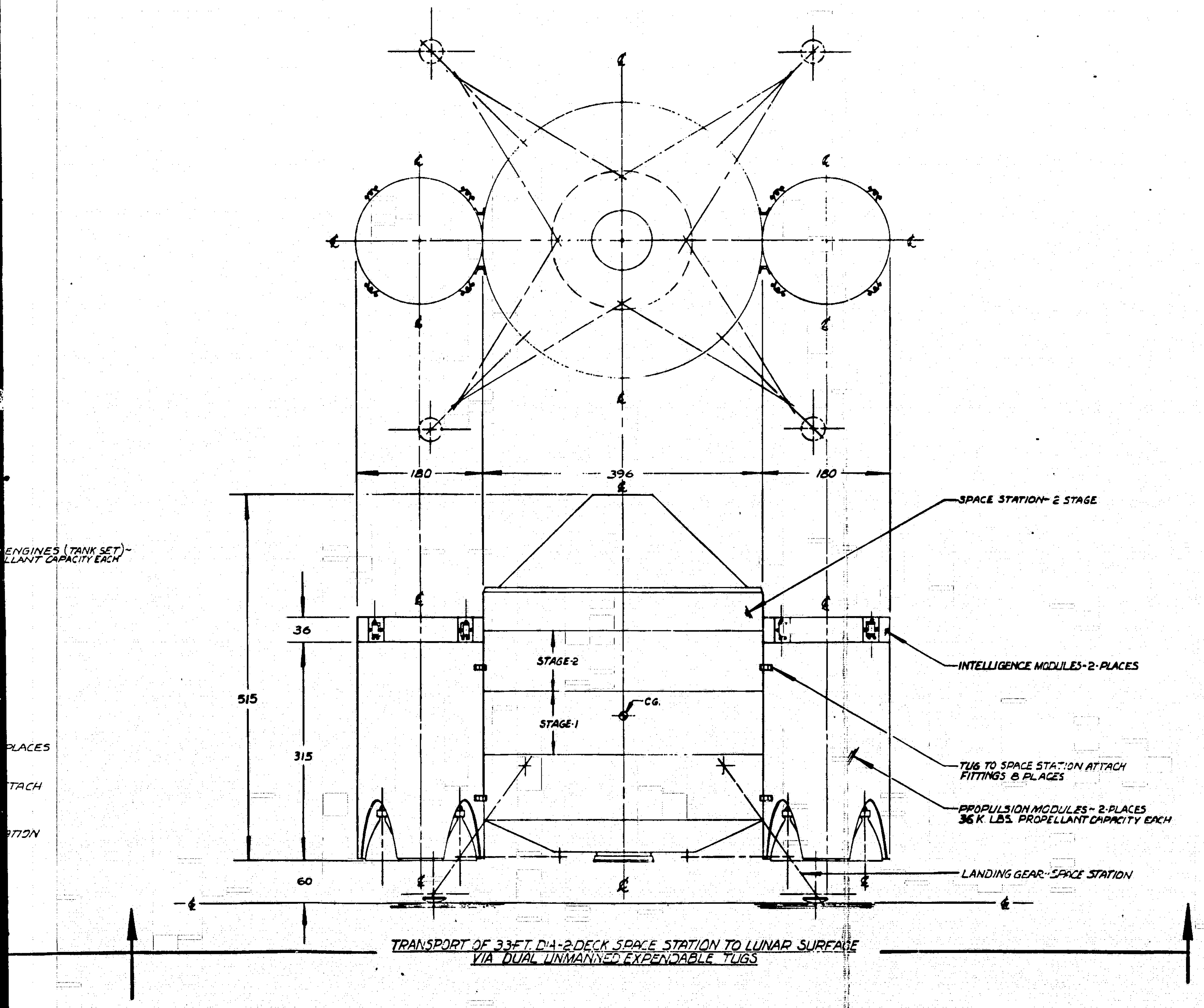

Figure 2-26. 36K Pound Propella
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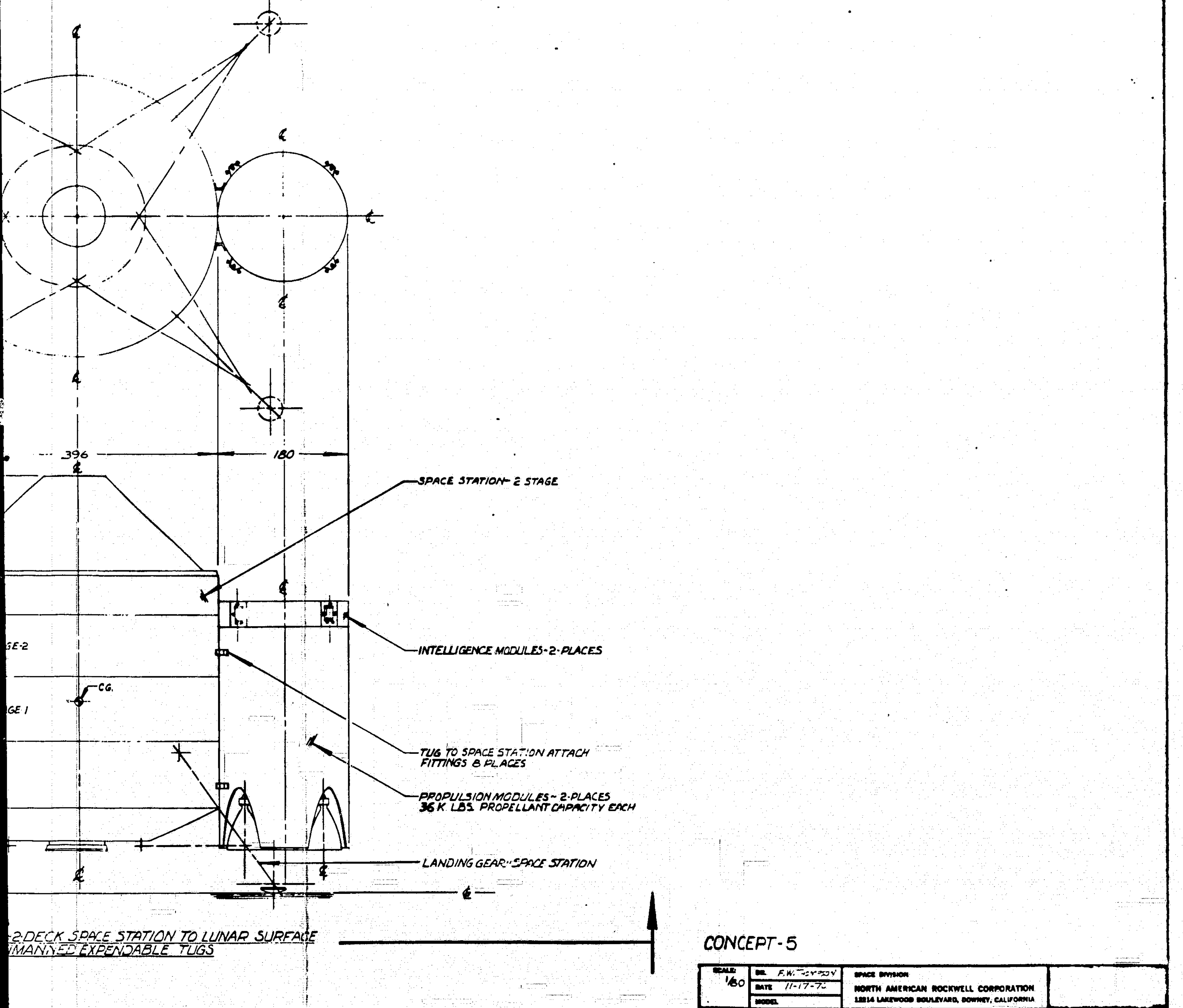

Figure 2-26. 36K Pound Propellant Tug Lunar Surface Base Payloads (Sheet 1 of 2)

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NOTES:
1-REMOVE TE
2-LATCHES E
3-UNLOADIN
a- DISENG
b- LOWER
c- UNLOAD
d- RAISE BO
e- LOWER S

TRANSPORT / LUNAR LANDING OF 8 SHELTE

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SHEET 3 OF 3

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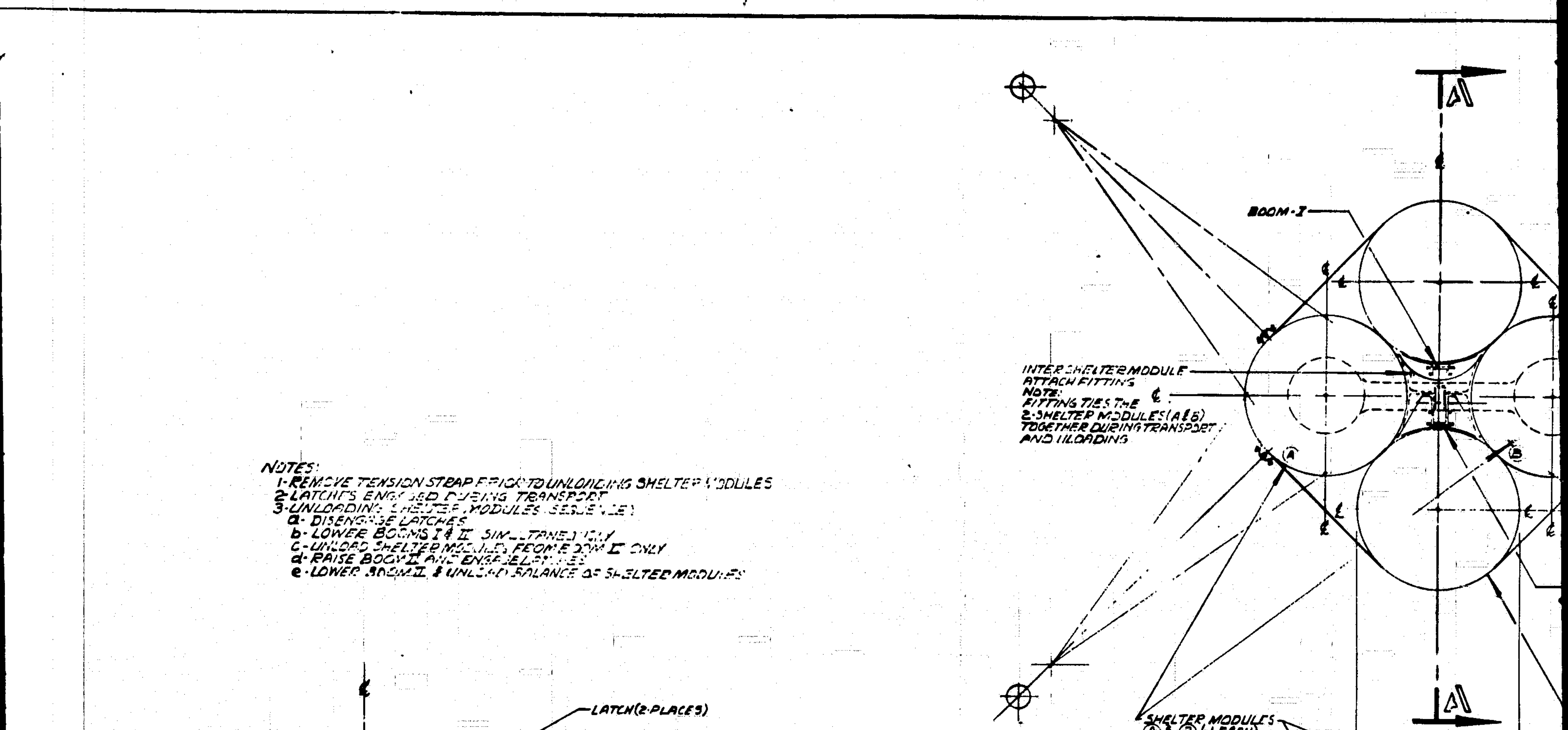

Figure 2-26. 36K Pound Propellant Tu
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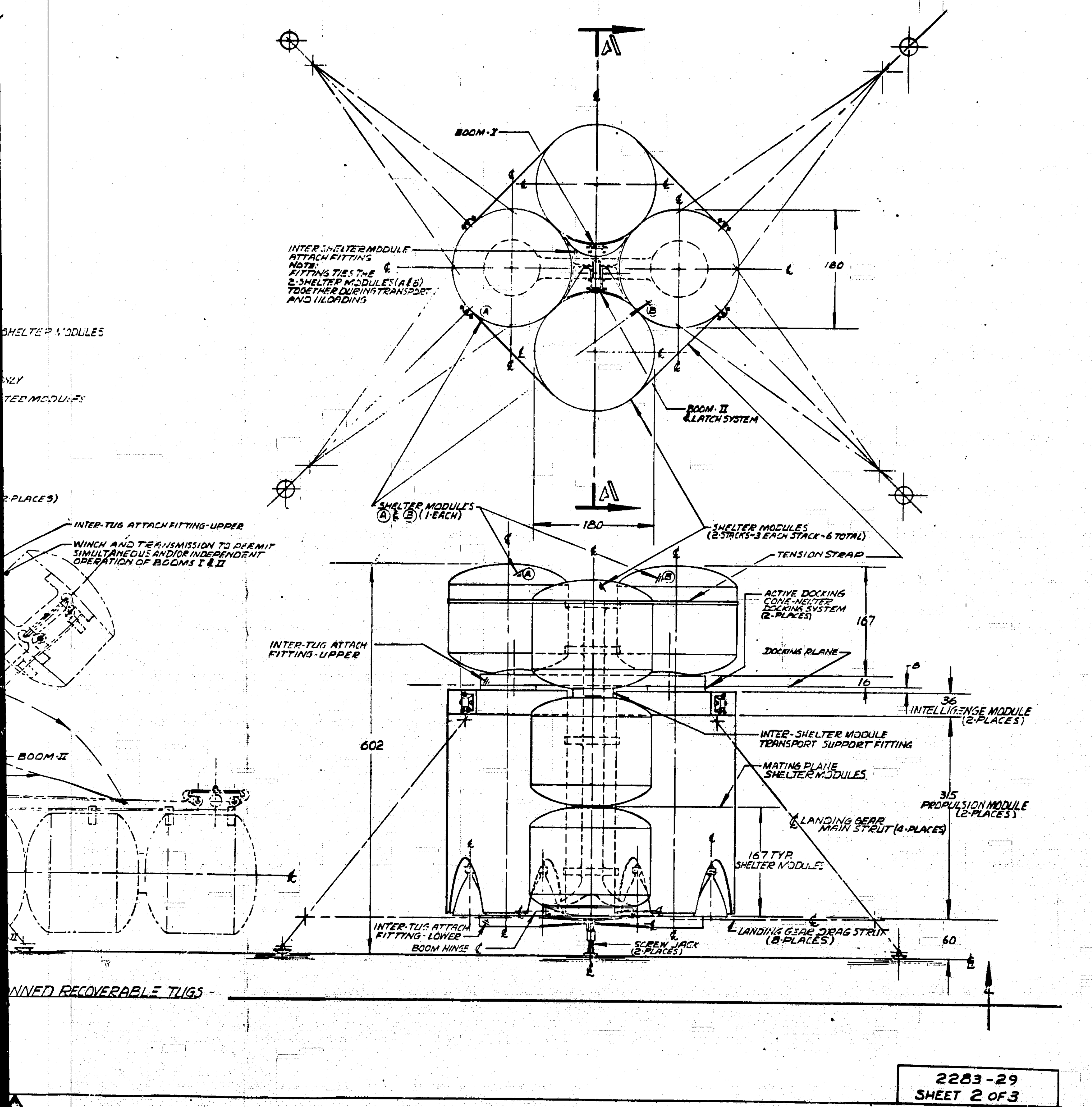

Figure 2-26. 36K Pound Propellant Tug Lunar Surface Base Payloads (Sheet 2 of 2)

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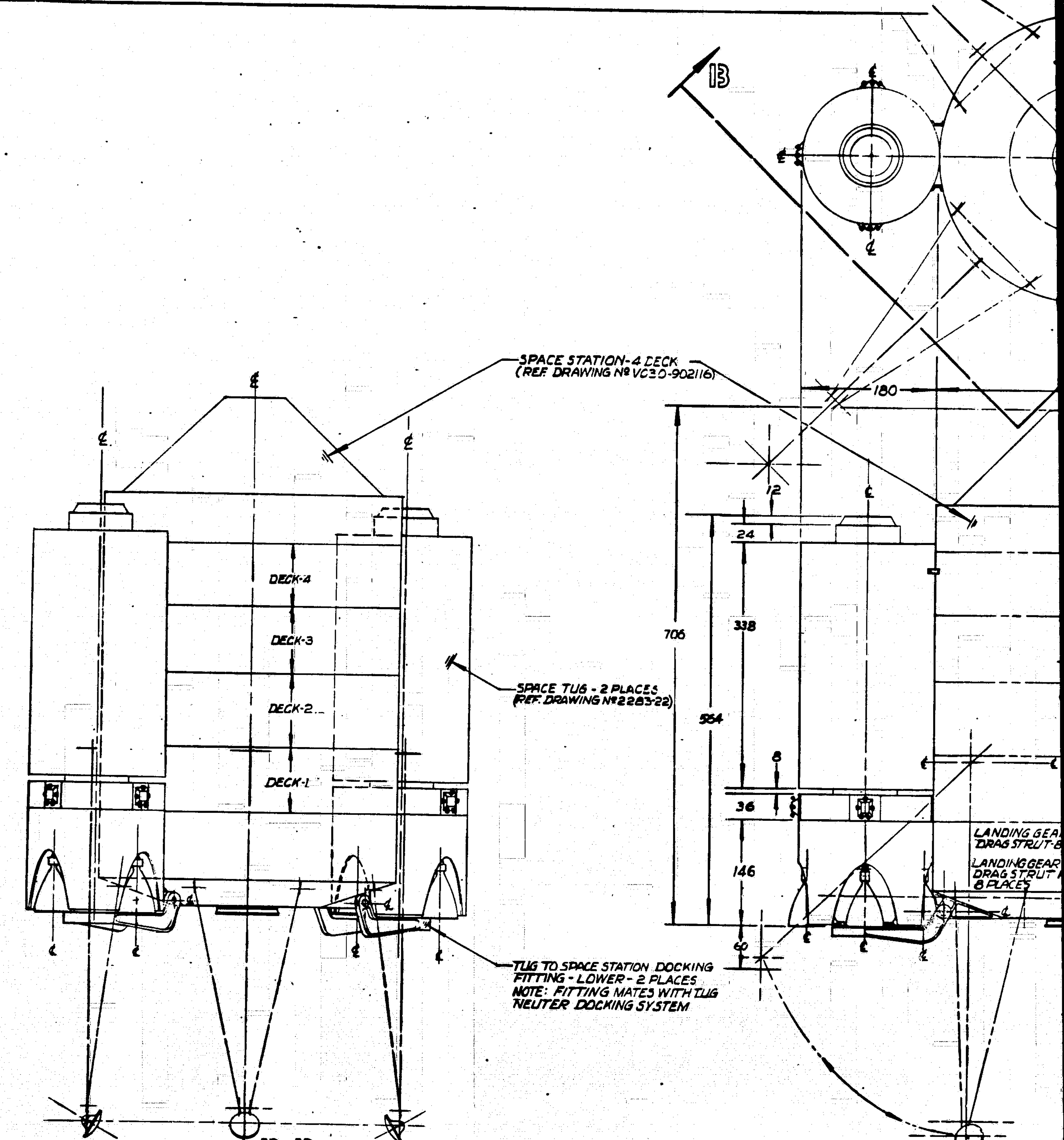

VIEW 13-13

LANDING GEAR SHOWN IN RETRACTED POSITION

SPACE PICK-UP AND TRANSPORT TO LUNAR SURFAC
VIA DUAL UNMANNED EXPENDABLE TUGS ~ 56K PRO

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SHEET 2 OF 3

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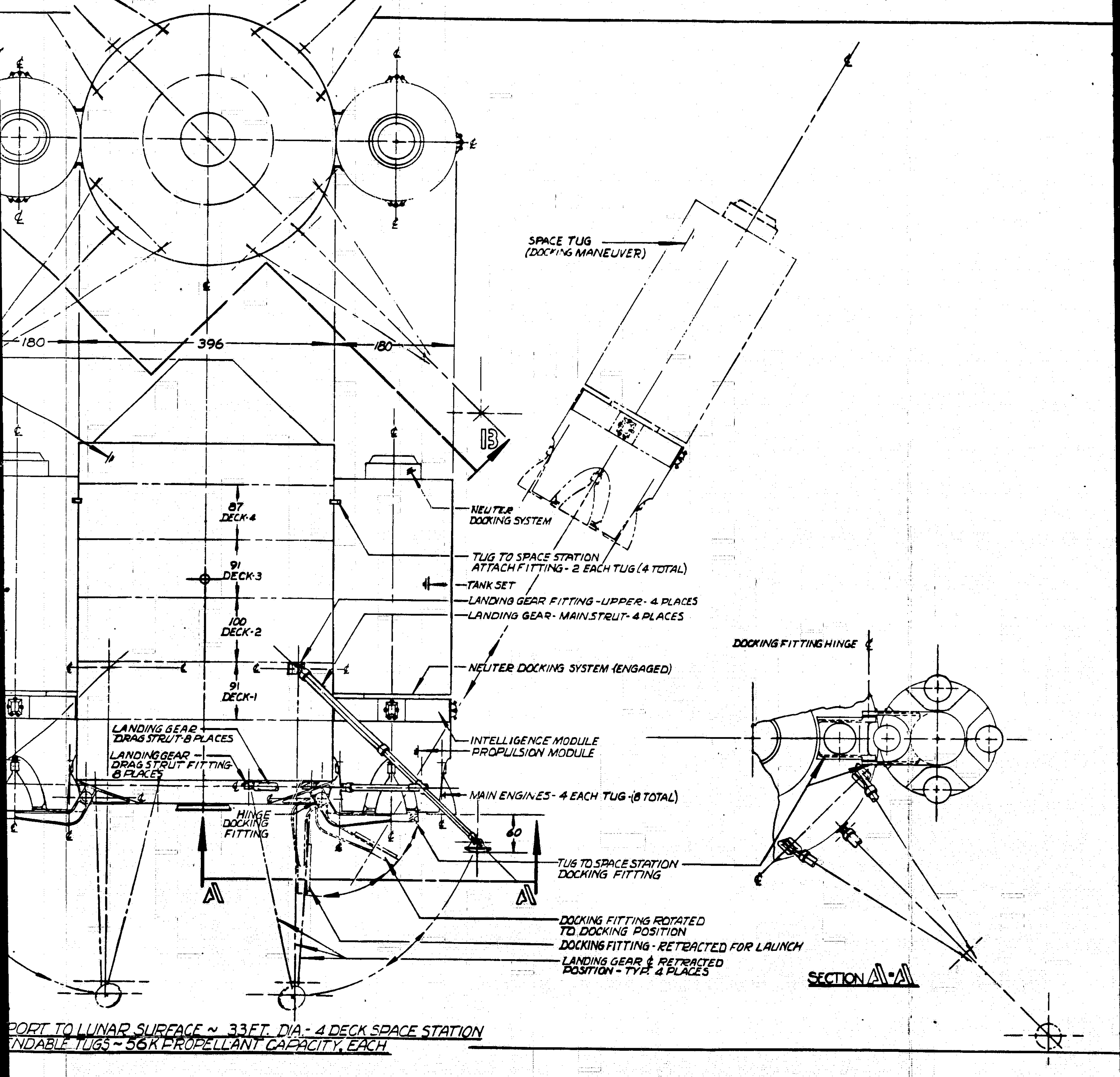

Figure 2-27. 56K Po

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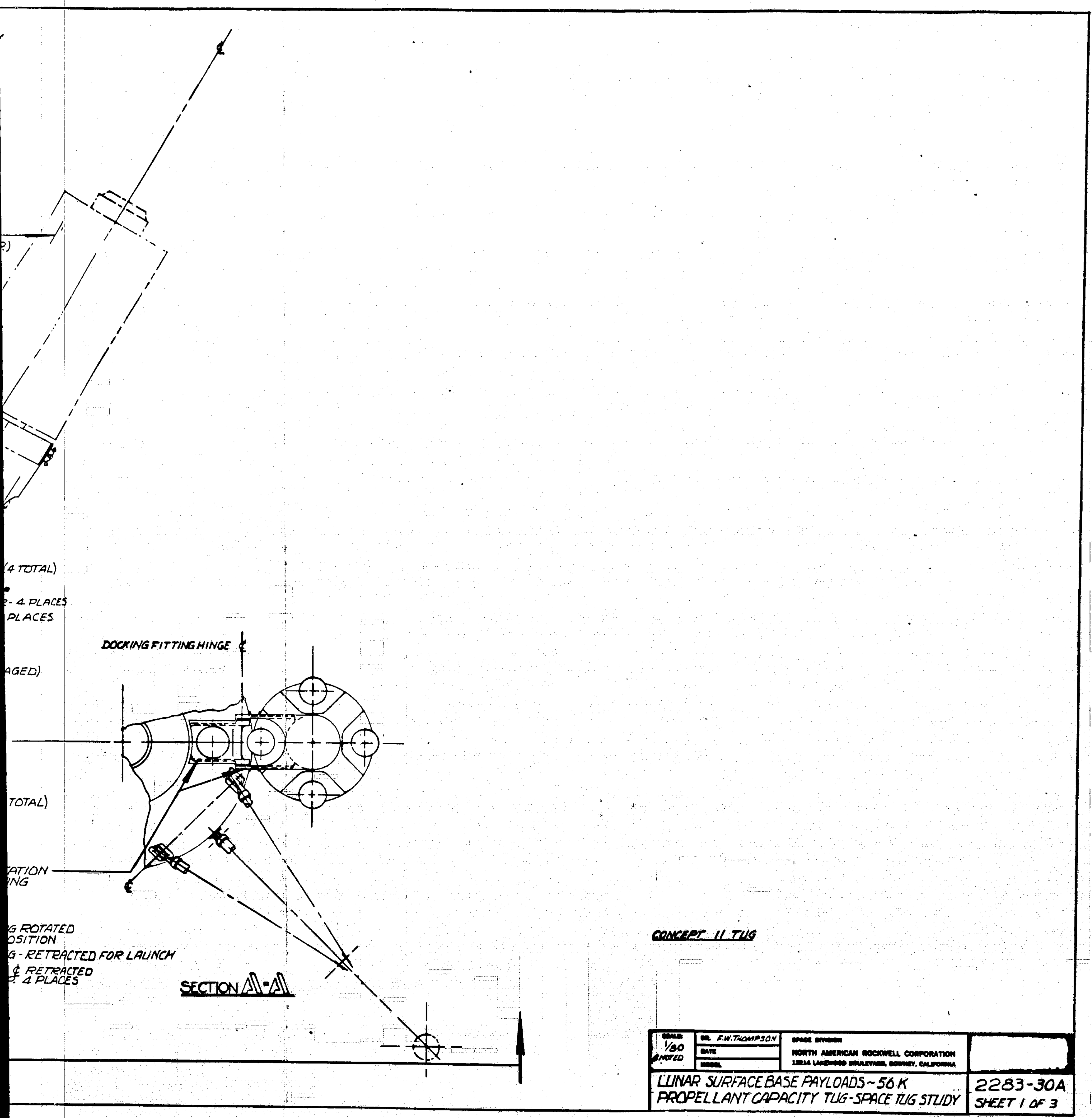
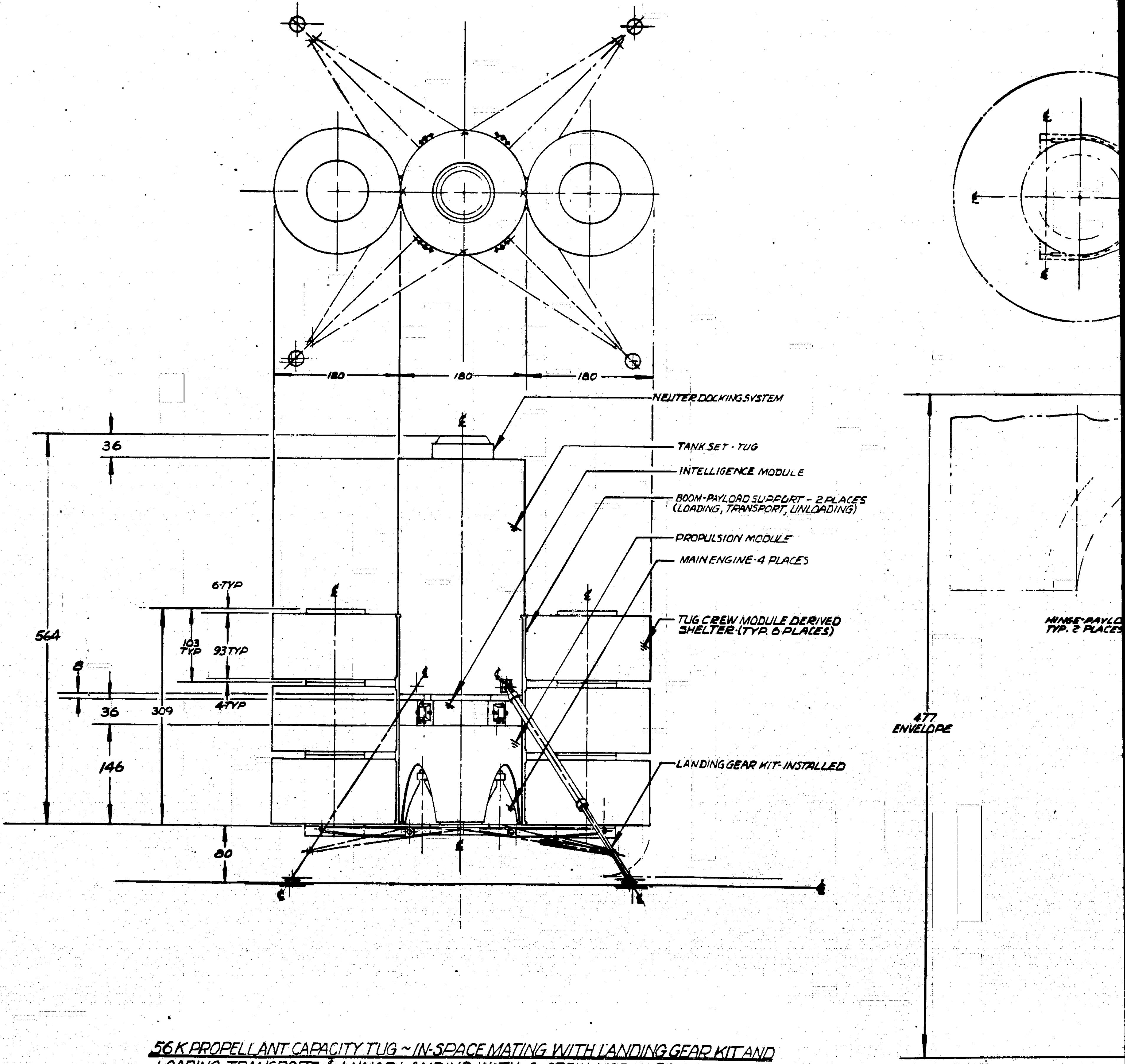


Figure 2-27. 56K Pound Propellant Tug Lunar Surface Base Payloads (Sheet 1 of 2)



56 K PROPELLANT CAPACITY TUG ~ IN-SPACE MATING WITH LANDING GEAR KIT AND LOADING TRANSPORT & LUNAR LANDING WITH 6 CREW MODULES

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SHEET 3 OF 3

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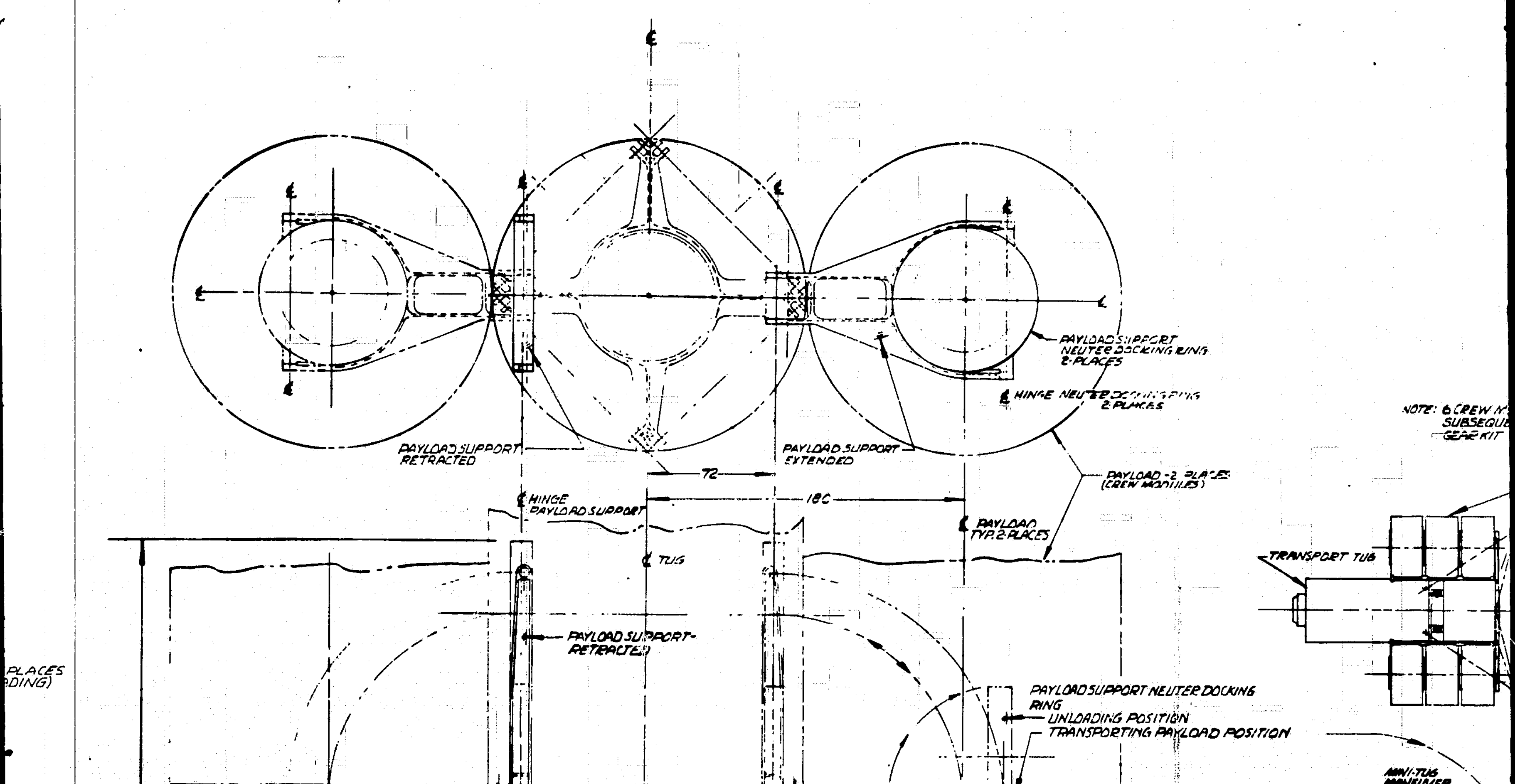

Figure 2-27. 56K Pound

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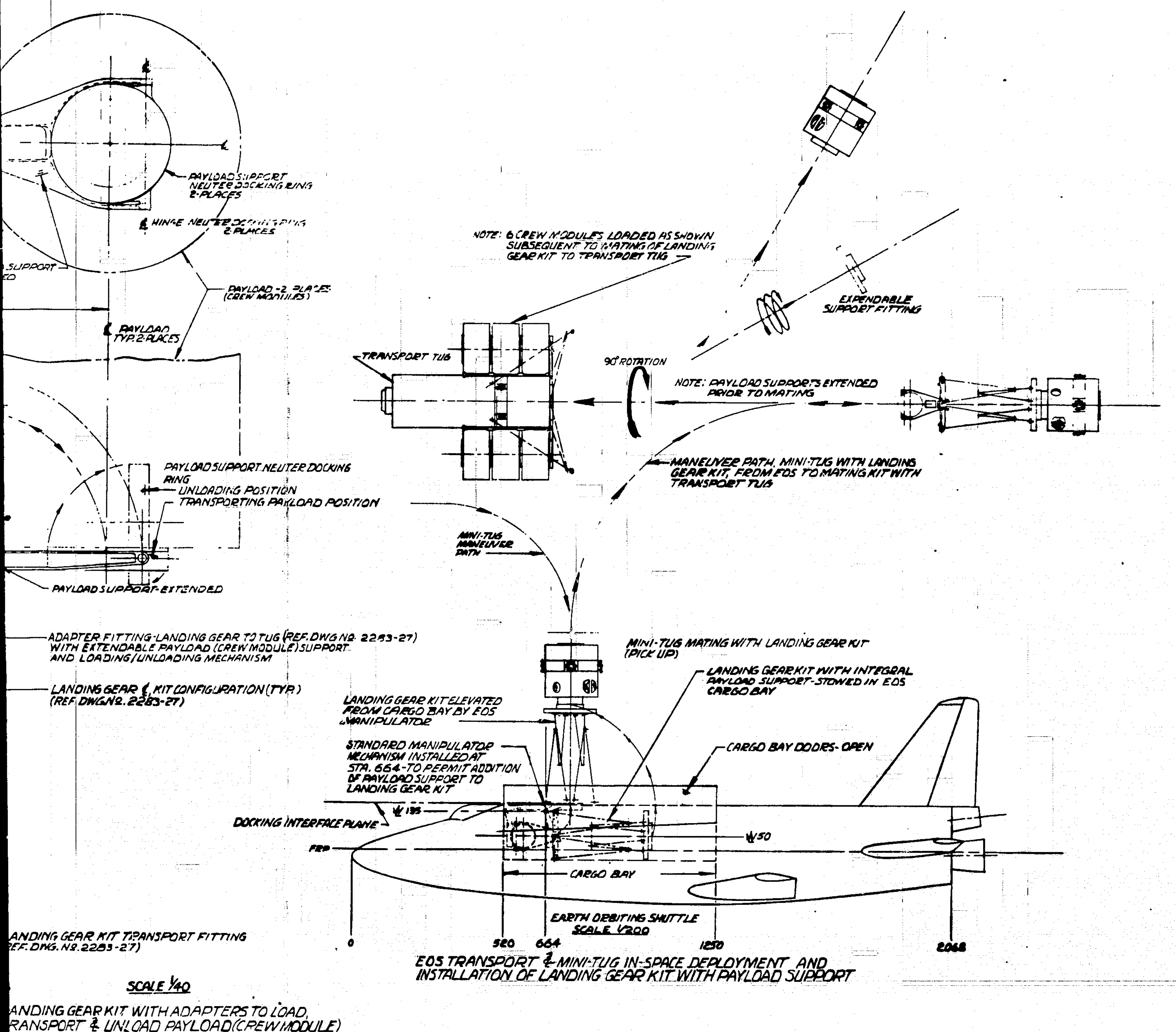


Figure 2-27. 56K Pound Propellant Tug Lunar Surface Base Payloads (Sheet 2 of 2)

was docked it would be rotated into the parallel position, and the forward stabilizing attachments would be made. For the two-stage concept shown in Figure 2-26, the lander concept is the same except that only a two-deck section of the core module is used since the payload capability of the two smaller stages is less than either the single- or 1-1/2-stage vehicles clustered in parallel. The four-deck core module could be landed if a propellant tank set were added to each single stage, as shown.

Lunar surface base modules of the 15-ft-diameter (4.57-m) type can be accommodated in various clustering concepts shown on the drawings. The modules could be carried in clusters stacked end to end and attached to the sides of the tug lander in quantities compatible with the lander payload capability and symmetrically arranged from c.g. considerations. Various structural attach systems were considered, but the hinged-boom concept offered several advantages. It eliminated attach interfaces between each module and the lander and provided a convenient deployment system for lowering the modules to the ground, as shown on the three drawings.

As a result of this conceptual study, it was concluded that there are feasible approaches equally compatible with each of the candidate tug pm concepts that permit various lunar surface base shelter modules to be landed. The strap-on accommodation approach is compatible with small, as well as large, shelter module concepts.

2.2.10 MODULE AND KIT FEASIBILITY

For a basic unmanned vehicle capability, yet to retain provisions for manned operation, functions must be carefully distributed between modules. The IM contains all astrionics for unmanned flight. The duration of flight is limited only by the stored propellants. Additional propellants are contained in the PM main propulsion tanks and are drawn upon for auxiliary propulsion, fuel cell reactant, and crew oxygen during a mission. The CM contains a complete life support system, as well as inputs from the IM for electrical power, water, oxygen, and astrionics interfaces. Cargo is stored in hemipods attachable in space, primarily for lunar surface delivery. A manipulator attachment to either the IM or CM enables the tug to service or retrieve satellites or to maneuver small cargo components.

It has been shown that the geosynchronous orbit mission requires very high performance, which tends to preclude PM compromises for the multipurpose use. However, these compromises do not appear to be substantial until consideration is given to the lunar lander version wherein landing loads, cargo pods, landing gear attachments, and special visibility/access requirements are imposed on the configuration. The PM is generally optimized for the geosynchronous (high energy) mission and is offloaded for other

applications. Changes such as structure beef-up, retractable engines, and various kits are usually required for the lunar lander version.

A single basic CM design appears feasible for transport of up to six men to and from a space station, four men for a 28-day lunar surface stay, and a six- to 12-man rescue. Docking adapters on top or bottom provide crew ingress and egress for the various mission applications.

A CAM can be nearly any reasonable shape for use in earth orbital missions. In general, it should provide for standard docking interfaces at each end and should be pressurizable for compatibility with a space station. The special constraints of the lunar landing configuration of tug (i.e., low c.g., easy loading and surface access, maintaining vehicle balance) result in choice of hemipods or cynlindrical halves that can be EOS-launched and readily attached laterally to tug landers low on opposite sides. These hemipods can serve double duty as a single cylinder module if the two halves are bolted together and docking adapters are added.

The IM can incorporate the basic subsystems required for unmanned tug flight. Add-on kits can provide the necessary additional power, life support, communications, etc. for longer duration and manned flight. While options exist as to the specific location of some of these subsystem elements (within PM or within IM), the differences do not now appear large.

The variety of missions to which the IM may be subjected does indicate that a separate module is desirable. If the IM is combined with the PM, module structure, weight is saved but some weight must be added to the PM since components must still be mounted and interconnected. The structural weight, as well as vehicle length, would decrease slightly. However, the PM forward end must be closed out to support a docking port. If the IM is combined with the PM the missions that do not require a PM (shuttle/space station short-range cargo and crew transfer, space station remote experiment module control) would be penalized. Integration of the IM with the CM would severely penalize all unmanned missions.

Many potential equipment divisions within the IM are possible. If the IM is to be capable of flying alone, it must have, as a minimum, all unmanned functions. Beyond this requirement it may also include certain manned functions. Inclusion of manned electrical power sources in the IM seems attractive since fewer IM/CM interfaces are required, space maintenance of EPS is centralized, and crew safety may be slightly improved. The presence of auxiliary control thrusters on the IM precludes using that area for active thermal control space radiators. The IM radiatiors would therefore be mounted on the PM or on a separate skirt when the IM flies alone.

Internally, the IM arrangement consists of four ACS support elements (tanks, valves, electrical, and fluid lines), between which is mounted all other equipment. The ACS support elements, as well as the other equipment, are mounted to the exterior IM walls for ease of removal and to promote passive thermal control.

Kits, including radiator, landing gear, manipulators, and long-range extensible antenna kits may be either eliminated or integrated into modules. Basic radiators for space operation are mounted on the vehicle periphery. The lunar surface radiator is added to the top of the tug to reduce reradiation heating. The top radiator is initially folded against the vehicle sides when docked and during landing. If a 22-foot-diameter (6.7-m) tug is landed, no deployable radiator area is required. A landing gear kit could either be an integral component or a separate assembly for each leg. Either method would involve EVA to attach. A manipulator kit may be attached at the docking port, inside the CM, if manned. Additional assembly would contain all necessary controls and displays. An antenna kit would be required if high data rates are to be transmitted from lunar orbit to earth. This antenna should be approximately six feet (1.8 meters) in diameter and could be attached to the tug or could be ground-deployed after landing, depending on mission requirements.

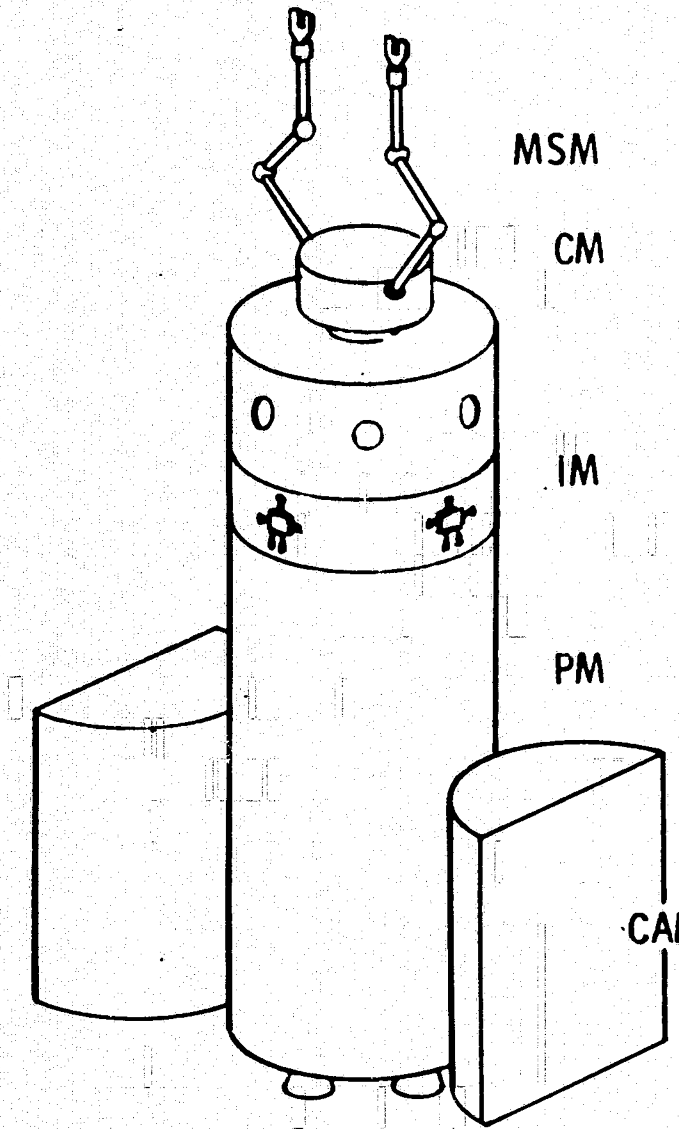
Although the organization of modules and kits in the vehicle stack depends somewhat on the mission requirements, these differences will be minimized as the tug concept becomes more formalized. A preliminary description of the module and kit orientation for a space mission is shown in Figure 2-28.

Another aspect of the case for modular feasibility involves assembly. A large number and types of interfaces must be reliably joined. And, to accomplish the task in space, may be beyond anticipated technical capability. The cryogenic propellant connections, in particular, require meticulous cleanliness, zero leakage in inclosed areas, and shielding from heat shorts. It is recommended, therefore, that the PM-to-IM and IM-to-CM interfaces be considered ground-assembled only, until such time as a capable space base facility becomes available to support these operations or until sufficient technology advances are made in this area.

2.2.11 CONCEPTUAL WEIGHT ANALYSIS

Preliminary Weight Sizing

Use of the design synthesis models as described in Appendix E permitted the sixing of the ten basic concepts (cases). Case 11 was added after the original ten cases were defined. Table 2-5 again summarizes the concepts



	CAPABILITIES	FUNCTIONS
MSM	• MANIPULATOR SUB-MODULE	• SAT. RETRIEVAL. SERVICE • CARGO HANDLING
CM	• 4 MEN-28 DAY LUNAR STAY 6 MEN-7 DAYS SPACE 12 MEN-1 DAY RESCUE	• EC/LSS • I/O FOR G&N. COMM • PWR DISTRIB & THERM CONT • PRESSURIZED ACCESS
IM	• UNMANNED FLIGHT CM SUPPORT AUTO & REMOTE OPS	• G&N. COMM & DATA MGT • ELECT. PWR & THERM CONT • AUX FLT CONTROL SYS • EXT ACCESS (EVA)
PM	• MAIN PROPULSION	• PROPULSION SYSTEM • PROPELLANT FEED • PWR DISTRIB & THERMAL CONT
CAM	• CARGO HEMI-PODS	• EXPENDABLES • INSTRUMENTS & TOOLS • EXPERIMENTS • MOBILITY AIDS

Figure 2-28. Current Philosophy of Module Organization

Table 2-5. Summary of Candidate Multipurpose Approaches

Case	PM Max Capability 1000 lb/(kg)	Staging Modes* Geosynchronous Mission	Lunar Mission Mode A	Lunar Mission Mode B	Lunar Mission Mode C and D
1	80/(36	A-1**	A-1	A-1	A-1
2	52/(24)	D-1, B-3	A-1**		
3	45/(20)	D-1**	C-1, D-1	A-1	A-1
4	41/(19)	B-3		A-1**	
5	36/(16)	B-3**	C-1, D-1	C-1, D-1	A-1**
6	31/(14)	C-4**	B-3**, C-1, D-1	C-1, D-1	C-1, D-1
7	27/(12)	A-2**	C-1, D-1	C-1, D-1	C-1, D-1
8	23/(10.5)	B-5		B-3**	
9	21/(9.5)	B-5		C-1**, D-1**	B-3**, C-1**, D-1**
10	15/(7)	B-5**	C-1***	C-1***	C-1***

*(A-1) - Single-stage, recovered
(A-2) - Single-stage, expended
(B-3) - Two-stage, both recovered
(B-5) - Two-stage, second expended
(C-1) - Two-propulsion modules, one IM, parallel operation
(C-4) - Two-stage, second stage recovered, IM on second stage only
(D-1) - One propalsion module, one tank set, one IM operating as single stage
**Mission(s) from which concept originated
***Also has tank sets on each propulsion module

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and modes considered. Tables 2-6 through 2-8 summarize the weight data resulting from these analyses.

Final Weight Analyses

The final computation of the performance characteristics of the three selected concepts is based on the parametric weight-scaling equations derived from detailed analyses of the required modules. These modules include the IM, PM, tank set, (a special case of a PM without engines), CM, and CAM and several kits such as lunar landing gear and manipulator.

Propellant Module (PM)

The PM contains the impulsive propellant, attitude control propellant, and interfaces with the IM and CM. The PM consists of a single LH_2 tank, four LO_2 tanks, and four engines with a 15-ft-diameter cylindrical non-integral external shell. The forward end assembles with an IM, while the aft end contains an Apollo-type passive docking capability. The structural weight of the PM is based on the data presented in Appendix A
A detail analysis of the weight of this configuration and several primary configurational alternatives is presented in Appendix C. Parametric weight evaluation of the PM structure resulted in this equation:

$$W_{STR} = \text{PM structure}$$
$$= 1660.0 + 0.0336*W_p, \text{ lb}$$
$$= (753.0 + 0.0336*W_p, \text{ kg})$$

Structure here includes tank cryogenic insulation and meteoroid protection for a seven-day mission.

The parametric engine weight is based on the engine weight data presented in Appendix E.

$$W_{ENG} \text{ (engine weight)} = 110.0 \text{ NE} + \text{ft}/87.5, \text{ lb}$$
$$= (50.0 \text{ NE} + \text{ft}/87.5, \text{ kg})$$
$$W_{TVC} \text{ (thrust vector control)} = 25.0 \text{ NE} + \text{ft}/100.0, \text{ lb}$$
$$= (11.3 \text{ NE} + \text{ft}/100.0, \text{ kg})$$
$$W_{MTG} \text{ (engine mounts)} = 20.0 \text{ NE} + \text{ft}/50.0, \text{ lb}$$
$$= 9.1 \text{ NE} + \text{ft}/50.0 \text{ kg})$$

Engine system (feed, fill/drain, and pressurization) parametric weight data are based on current in-house studies and the latest configurational drawings:

$$W_{SYS} \text{ (engine systems)} = 25.0\ XN_{FT} + 40.0\ XN_{OT} + 0.0010\ XW_P, \text{ lb}$$

$$= (11.3\ XN_{FT} + 18.1\ XN_{OT} + 0.0010\ XW_P, \text{ kg}$$

$$W_{RES} \text{ (residual propellants)} = 735.0 + 0.0032 * W_P, \text{ lb}$$

$$= (333.0 + 0.0032 * W_P, \text{ kg})$$

W_{LOS} (two starts, two shutdowns, and seven days of boiloff) = 370 lb

= (168 kg)

where:

f_t = total thrust in pounds (kg)

NE = number of engines

N_{OT} = number of oxidizer tanks

N_{FT} = number of fuel tanks

W_P = total usable propellant (impulsive plus auxiliary)

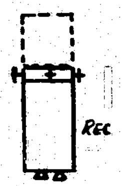
For a tug PM having four engines, four LO_2 tanks, one LH_2 tank, and an engine burn of 1000 seconds (ft = 0.463 W_P), the total of the above elements is

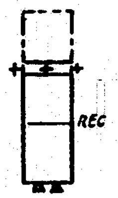
$$W_{PS} = 1890.0 + 0.0109 * W_P, \text{ lb}$$

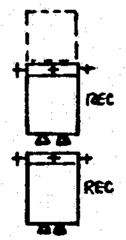
$$= (857.3 + 0.0109 * W_P, \text{ Kg})$$

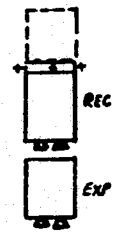
Table 2-6. Concept Weight Summary (1000 Pounds)

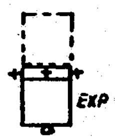
Concept No.	1	2	3	4	5	6	7	8	9	10
NR Staging Mode	A1*	D1	D1*	B3	B3*	C4*	A2*	B5	B5	B5*
IM - 1	3.3	3.3	3.3	3.3	3.3	3.3	3.3	3.3	3.3	3.3
- 2	-	-	-	3.3	3.3	-	-	3.3	3.3	3.3
CM	-	-	-	-	-	-	-	-	-	-
PM - 1	7.9	5.5	5.0	4.4	4.1	3.9	3.5	3.1	3.0	2.4
- 2	-	(TS)4.5	(TS)3.9	4.4	4.1	3.9	-	3.1	3.0	2.4
Landing gear	-	-	-	-	-	-	-	-	-	-
Payload	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
Total inert	21.2	23.4	22.2	25.4	24.8	21.1	16.8	22.8	22.6	21.4
ΔV propellant	80.4	92.2	90.3	74.6	73.0	63.3	27.1	28.3	29.3	30.5
Aux. consumables	0.6	0.6	0.6	1.2	1.2	0.6	0.6	1.2	1.2	1.2
Gross S/C weight (10^3 kg)	102.2 (46.4)	116.2 (52.7)	113.1 (51.3)	101.2 (45.9)	99.0 (44.9)	85.0 (38.6)	44.5 (20.2)	52.3 (23.7)	53.1 (24.1)	53.1 (24.1)

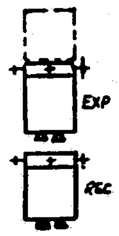






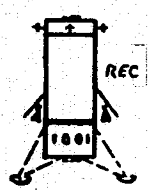
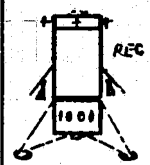
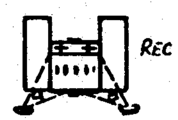






*Concept orgin

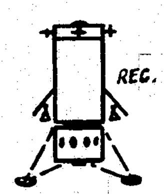
Table 2-7. Concept Weight Summary Manned Lunar Landing
NASA Mode A (1,000 Pounds)

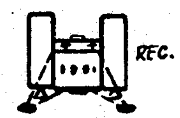
Concept No.	1	2	3	4	5	6	7	8	9	10
NR staging mode	A1	A1*	C1	C1	C1	C1	C1	-	-	C1**
IM - 1	3.8	3.8	3.8	3.8	3.8	3.8	3.8			3.8
- 2	-	-	-	-	-	-	-			-
CM	9.2	9.2	9.2	9.2	9.2	9.2	9.2			9.2
DM - 1	7.9	5.6	5.0	4.4	4.1	3.9	3.5			2.4+1.9
- 2	-	-	5.0	4.4	4.1	3.9	3.5			2.4+1.9
Landing gear	6.0	3.4	3.2	3.1	3.0	3.0	2.9			2.7
Payload	10.0	10.0	10.0	10.0	10.0	10.0	10.0			10.0
Total inert	36.9	32.0	36.2	34.9	34.2	33.8	32.9			32.3
Propellant	56.5	48.7	56.1	54.2	53.1	52.5	51.2			50.4
Aux. Consumable	3.0	3.0	3.0	3.0	3.0	3.0	3.0			3.0
Gross S/C weight (10^3 kg)	96.4 (43.7)	83.7 (38.0	95.3 (43.2)	92.1 (41.8)	90.3 (41.0)	89.3 (40.5)	87.1 39.5)			85.7 (38.9)
										

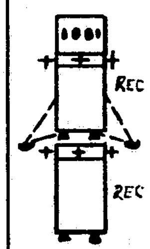
*Concept origin
**(PM + tank set)

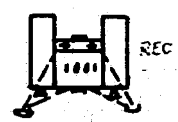
Table 2-8. Concept Weight Summary Manned Lunar Landing— NASA Mode B (1, 000 Pounds)

Concept	1	2	3	4	5	6	7	8	9	10
NR staging mode	A1	A1	A1	A1*	C1	C1	C1	B3*	C1	C1**
IM - 1	3.8	3.8	3.8	3.8	3.8	3.8	3.8	3.3	3.8	3.8
-2	-	-	-	-	-	-	-	3.8	-	-
CM	9.2	9.2	9.2	9.2	9.2	9.2	9.2	9.2	9.2	9.2
PM - 1	7.9	5.6	5.0	4.4	4.1	3.9	3.5	3.1	3.0	2.4+1.9
- 2	-	-	-	-	4.1	3.9	3.5	3.1	3.0	2.4+1.9
Landing gear	6.0	3.4	3.2	2.6	3.0	3.0	2.9	2.4	2.8	2.7
Payload	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
Total inert	36.9	32.0	31.2	30.0	34.2	33.8	32.9	34.9	31.8	32.3
ΔV propellant	48.0	40.4	39.4	37.9	44.8	44.2	42.9	43.4	40.5	42.1
Aux. consumables	3.0	3.0	3.0	3.0	3.0	3.0	3.0	3.7	3.0	3.0
Gross S/C weight (10^3 kg)	87.9 (39.9)	75.4 (34.2)	73.6 (33.4)	70.9 (32.2)	82.0 (37.2)	81.0 (36.7)	78.8 (35.7)	82.0 (37.2)	75.3 (34.2)	77.4 (35.1)









*Concept origin
**2 PM 1 tank set

This parametric number, when added to the equation of the structural portion of the PM, yields a resultant inert weight equation of the PM:

$$WIPM(1) = \text{(inert PM weight)}$$

$$= 3550.0 + 0.0445 * W_P, \text{ lb}$$

$$= (1610.3 + 0.0445 * W_P, \text{ kg})$$

where PM has (1) four engines at total thrust of 0.463 W_P, (2) four LO_2 tanks and one LH_2 tank, and (3) aft docking provisions. This weight is applicable to the single-stage concept (No. 1) configuration.

The parametric weight equation for the two-stage concept (No. 5) configuration is identical to Concept 1 except that the thrust of each stage doubles with respect to its propellant load; i.e., $F_T = 0.926 * W_P$. The inert PM weight equation then becomes

$$W_{IPM}(5) = 3550.0 + 0.0532 * W_P, \text{ lb}$$

$$= (1610.3 + 0.0532 * W_P, \text{ kg})$$

Concept 11 configuration is composed of a tank set (a PM less thrust structure and engines) and a small specialized PM containing two LO_2 tanks and two LH_2 tanks. The stage thrust is truly a function of the propellant load of the tank set and PM. However, for separate equations to be formed for the two stages, a relationship of the thrust to the PM propellant load only is used.

These parametric PM weight data were used in the final performance computations presented later in this report. A plot of the PM inert weights as a function of usable propellant loading is presented in Figure 2-29. Use of these equations in the performance evaluation of the geosynchronous missions resulted in a deterimation of the mission propellant for the three basic configurations. A weight summary of these PM modules is presented in Tables 2-9 and 2-10.

Intelligence Module (IM)

The IM contains all the equipment essential to the operation of a tug mission except the primary impulsive propellant and storage for the main portion of the auxiliary propellant. When coupled with a PM, the IM/PM forms the basic mission spacecraft. The IM equipment is housed in an annular structure of 15.0 feet (4.57 meters) outside diameter, 6.7 feet (2.04 meters) inside diameter, and 3.0 feet (0.91 meter) long. Insulation is provided on the total surface. Meteoroid protection is provided on the

The ratio of the propellant of the TS to the PM loading is approximately 4.75, therefore the thrust of the first stage is approximately (4.75 + 1.0) x (0.463) = 2.66 times the PM propellant load. The residuals noted previously are distributed by the number of tanks and propellant load. The strat-shutdown boiloff losses are split between the TS and the PM. The resulting parametric weights for Concept 11 are given below:

WITS = inert tank set weight

$= 2,880.0 + 0.0294 + W_{po}$, lb

$= (1306.4 + 0.0294 + W_{po}$, kg)

WIPM (11) = inert PM for Concept 11.

$= 2,245.0 + 0.0865 \pm W_{pr}$, lb

$= (1018.7 + 0.0865 + W_{pr}$, kg)

W_{po} = outbound propellant

W_{pr} = return propellant

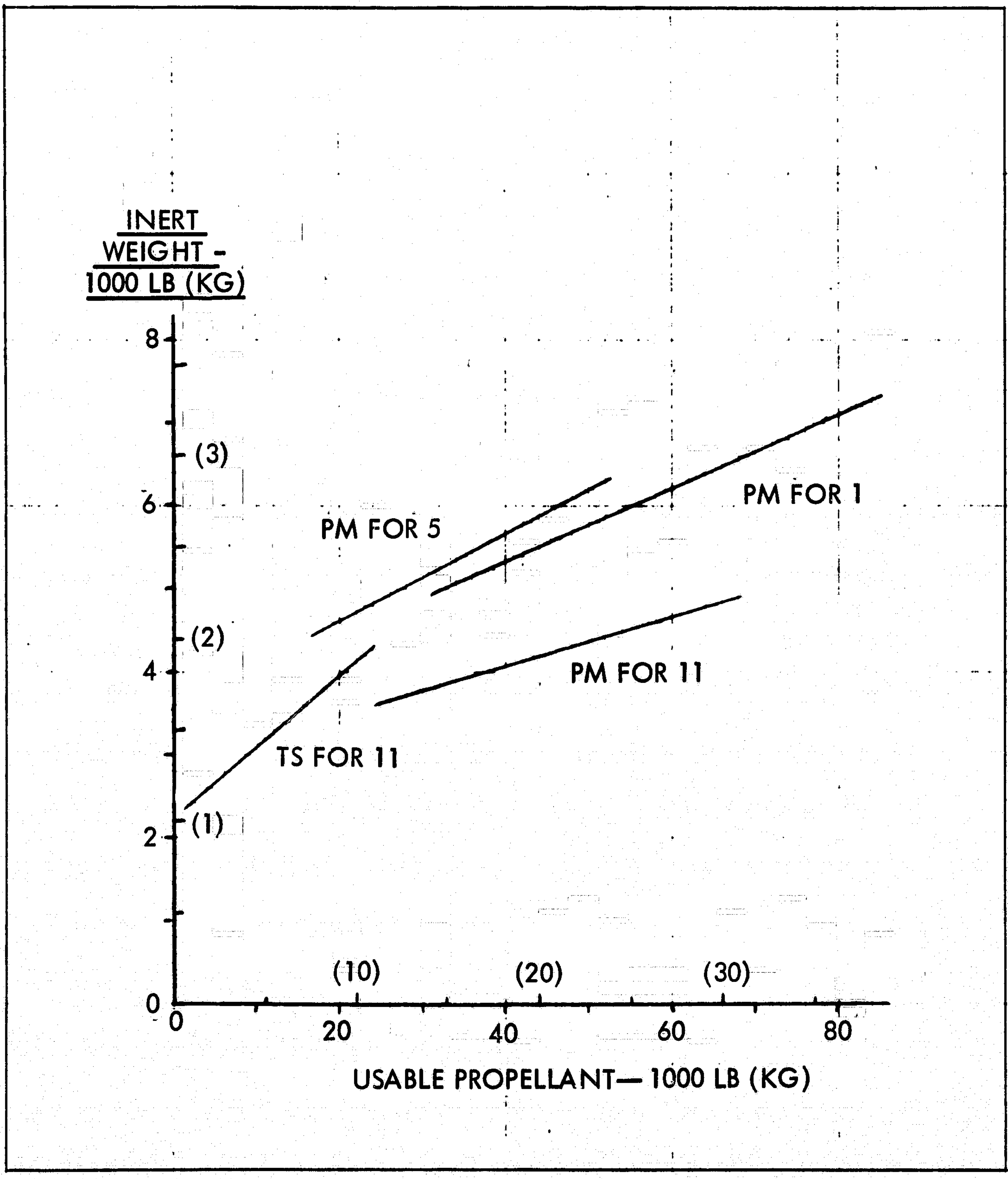


Figure 2-29. PM Inert Weight Versus Propellant Load

Table 2-9. Propulsion Module Summary Weight Statement - English Units

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

SPACECRAFT SUMMARY WEIGHT STATEMENT

CONFIGURATION	BY	DATE
PROPELLANT MODULES	FGC	January, 1971

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	U
1.0	AERODYNAMIC SURFACES								
2.0	BODY STRUCTURE	3,015	2,070	915	2,380				
3.0	INDUCED ENVIR PROT	690	535	140	580				
4.0	LNCH RECOV & DKG	75	75	75	75				
5.0	MAIN PROPULSION	1,599	1,564	1,484	274				
6.0	ORIENT CONTROL SEP & ULL								
7.0	PRIME POWER SOURCE								
8.0	POWER CONV & DISTR				46				
9.0	GUIDANCE & NAVIGATION	17	17	17	-				
10.0	INSTRUMENTATION								
11.0	COMMUNICATION								
12.0	ENVIRONMENTAL CONTROL	◆ 29	◆ 29	◆ 29	-				
13.0	GROWTH ALLOWANCE	270	215	130	165				
14.0	PERSONNEL PROVISIONS								
15.0	CREW STA CONTRL & PAN								
16.0	RANGE SAFETY & ABORT								
SUBTOTALS (DRY WEIGHT)		5,695	4,505	2,790	3,520				
17.0	PERSONNEL								
18.0	CARGO								
19.0	ORDNANCE								
20.0	BALLAST								
21.0	RESID PROP & SERV ITEMS	975	870	220	715				
SUBTOTALS (INERT WEIGHT)		6,670	5,375	3,010	4,235				
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES	250	250	125	125				
24.0	THRUST DECAY PROPELLANT	40	40	20	20				
25.0	FULL THRUST PROPELLANT	(77,795)	(41,210)		(10,980)	(52,375)			
26.0	THRUST PROP BUILDUP	9-	9-	40	40				
27.0	PRE-IGNITION LOSSES								
-	AUX PROPELLANT*	(595)	(495)	(100)	(495)				
TOTALS (GROSS WEIGHT) (LB)		7,040	5,745	3,195	4,420				
DESIGN ENVELOPE VOLUME (FT^3)									
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA (FT^2)									
PRESSURIZED SURF AREA (FT^2)									
DESIGN q, MAX (LB/FT^2)									
DESIGN g, MAX									
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS THRUST (LB)		36,200	38,200	29,300	-				

DESIGNATIONS:

CODE, SYSTEM; REF. MIL-M-38310A OR SP-6004

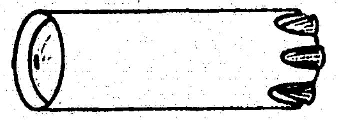
ITEM OR MODULE

A	PM for Concept 1	Unmanned
B	PM for Concept 5	Or Space
C	DM for Concept 11	Manned
D	TS for Concept 11	Operation
E		
◆ F	Incr by 40 lb for Lunar Mission	

SPACECRAFT

- M MANNED LAUNCH
- U UNMANNED LAUNCH
- * Ref Tank Propellant Capacity

NOTES & SKETCHES:



Propellant Modules e

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Table 2-10. Propulsion Module Summary Weight Statement - Metric Units

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

SPACECRAFT SUMMARY WEIGHT STATEMENT

CONFIGURATION: PROPELLANT MODULES | BY: FGC | DATE: January, 1971

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	U
1.0	AERODYNAMIC SURFACES								
2.0	BODY STRUCTURE	1,368	939	415	1,080				
3.0	INDUCED ENVIR PROT	313	243	64	263				
4.0	LNCH RECOV & DKG	34	34	34	34				
5.0	MAIN PROPULSION	725	709	672	124				
6.0	ORIENT CONTROL SEP & ULL								
7.0	PRIME POWER SOURCE								
8.0	POWER CONV & DISTR				21				
9.0	GUIDANCE & NAVIGATION	8	8	8	-				
10.0	INSTRUMENTATION								
11.0	COMMUNICATION								
12.0	ENVIRONMENTAL CONTROL	♦ 13	♦ 13	♦ 13	-				
13.0	GROWTH ALLOWANCE	123	98	59	75				
14.0	PERSONNEL PROVISIONS								
15.0	CREW STA CONTRL & PAN								
16.0	RANGE SAFETY & ABORT								
SUBTOTALS (DRY WEIGHT)		2,584	2,044	1,265	1,597				
17.0	PERSONNEL								
18.0	CARGO								
19.0	ORDNANCE								
20.0	BALLAST								
21.0	RESID PROP & SERV ITEMS	442	395	100	324				
SUBTOTALS (INERT WEIGHT)		3,026	2,439	1,365	1,921				
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES	113	113	57	57				
24.0	THRUST DECAY PROPELLANT	18	18	9	9				
25.0	FULL THRUST PROPELLANT	(35,369)	(18,693)	(4,981)	(23,757)				
26.0	THRUST PROP BUILDUP	36	36	18	18				
27.0	PRE-IGNITION LOSSES								
-	AUX PROPELLANT•	(270)	(225)	(45)	(225)				
TOTALS (GROSS WEIGHT) (KG)		3,193	2,606	1,449	2,005				
DESIGN ENVELOPE VOLUME (FT^3)									
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA (FT^2)									
PRESSURIZED SURF AREA (FT^2)									
DESIGN q, MAX (LB/FT^2)									
DESIGN g, MAX									
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS THRUST - KG		16,400	17,300	13,300					

DESIGNATIONS:

CODE, SYSTEM; REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

A	PM for Concept 1	Unmanned
B	PM for Concept 5	or Space
C	DM for Concept 11	Manned
D	TS for Concept 11	Operation
E		
♦ F	Incr by 18 Kg for Lunar Mission	

SPACECRAFT

M	MANNED LAUNCH
U	UNMANNED LAUNCH
•	Ref Tank Propellant Capacity

NOTES & SKETCHES:

Propellant Modules m

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external surface and the exposed upper surface. Structure to accommodate the swivel RCS pods is provided, and an Apollo-type docking probe attached to the inner structure is required when the IM/PM operates as a single spacecraft element. When the IM/PM is used in conjunction with a CM the docking probe is not required. A summary weight statement of three configurations of the IM are presented in Tables 11 and 12. All the equipment weight data are derived from the FO/FS listings in Section 4.2

Crew Module (CM)

The CM is a flat ended cylinder 15.0 feet (4.57 meters) in diameter and 7.75 feet (2.36 meters) high. One end is provided with a NASA pressurized passive neuter docking ring. The other has provisions on the 15.0-ft (4.57-m) diameter for attachment to an IM (if top mounted) or to a PM (if bottom mounted). Insulation is provided on all external surfaces. Meteoroid protection is provided on the external cylinder and the exposed end bulkhead. A summary weight statement for the basic six-man/seven-day space operation CM is presented in Tables 2-13 and 2-14. Also illustrated is the CM modified to fulfill the four-man/45-day lunar surface operation.

Cargo Module (CAM)

The configuration of the CAM used in the lunar landing missions consists of two semicircular pods (with a radius of 7.5 feet (2.28 m) and 8.33 feet (2.54 meters) long. These pods will be mounted flat side toward the core module 180 degrees apart at fitting points provided in the primary structure. The weight of a minimum access, unpressurized CAM of this size is approximately 1400 pounds (635 kilograms) a pair. Installation and tie-down equipment for the installed cargo are not included in this weight. The installed cargo must include its own installation hardware and insulation weight.

Lunar Landing Gear (LLG)

The lunar landing gear is a four-pad, reusable, tubular network attached to the landed core module tug. The four compression links are mounted to hard points of the core module at their upper ends and to the landing pads at the lower end the eight tension links are attached in pairs to the pads at the lower end around the perimeter of a ring (which mates with the aft docking provisions of the core module) at the upper end. The weight of the gear was estimated from available in-house information, including the LM landing gear. A parametric weight equation was formed that related the weight of the gear to the gross weight at lunar landing and the distance of the c.g. about the ground. A 5.0-ft (1.53-m) ground clearance and a core

Table 2-11. Intelligence Module Summary Weight Statement - English Units

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

SPACECRAFT SUMMARY WEIGHT STATEMENT

CONFIGURATION: INTELLIGENCE MODULES | BY: FGC | DATE: January, 1971

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	U
1.0	AERODYNAMIC SURFACES								
2.0	BODY STRUCTURE	520		520		520			
3.0	INDUCED ENVIR PROT	140		140		140			
4.0	LNCH RECOV & DKG (2)	200		-		-			
5.0	MAIN PROPULSION	-		-		-			
6.0	ORIENT CONTROL SEP & ULL	358		358		358			
7.0	PRIME POWER SOURCE	406		861		861			
8.0	POWER CONV & DISTR	100		100		100			
9.0	GUIDANCE & NAVIGATION	276		276		410			
10.0	INSTRUMENTATION (1)	101		-		-			
11.0	COMMUNICATION	935		935		935			
12.0	ENVIRONMENTAL CONTROL	184		331		762			
13.0	GROWTH ALLOWANCE	160		174		204			
14.0	PERSONNEL PROVISIONS								
15.0	CREW STA CONTRL & PAN								
16.0	RANGE SAFETY & ABORT								
SUBTOTALS (DRY WEIGHT)		3,380		3,695		4,290			
17.0	PERSONNEL								
18.0	CARGO								
19.0	ORDNANCE								
20.0	BALLAST								
21.0	RESID PROP & SERV ITEMS	15		15		15			
SUBTOTALS (INERT WEIGHT)		3,395		3,710		4,305			
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES								
24.0	THRUST DECAY PROPELLANT								
25.0	FULL THRUST PROPELLANT								
26.0	THRUST PROP BUILDUP								
27.0	PRE-IGNITION LOSSES								
-	AUX PROPELLANT	45		45		45			
TOTALS (GROSS WEIGHT) (LB)		3,440		3,755		4,350			
DESIGN ENVELOPE VOLUME (FT^3)									
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA (FT^2)									
PRESSURIZED SURF AREA (FT^2)									
DESIGN q, MAX (LB/FT^2)									
DESIGN g, MAX									
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS									

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

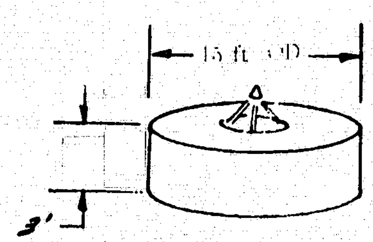
- A Unmanned IM
- B
- C Manned Space IM
- D
- E Manned Lunar Landing IM
- F (Incl G&N - Thermal Control Kits)

SPACECRAFT

- M MANNED LAUNCH
- U UNMANNED LAUNCH

NOTES & SKETCHES:

(1) For Unmanned Docking Only
(2) Not Required for Manned Operation



Intelligence Modules, e

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Table 2-12. Intelligence Module Summary Weight Statement - Metric Units

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

SPACECRAFT SUMMARY WEIGHT STATEMENT

CONFIGURATION: INTELLIGENCE MODULES
BY: FGC
DATE: January, 1971

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	U
1.0	AERODYNAMIC SURFACES								
2.0	BODY STRUCTURE	236		236		236			
3.0	INDUCED ENVIR PROT	64		64		64			
4.0	LNCH RECOV & DKG	91							
5.0	MAIN PROPULSION								
6.0	ORIENT CONTROL SEP & ULL	162		162		162			
7.0	PRIME POWER SOURCE	184		391		391			
8.0	POWER CONV & DISTR	45		45		45			
9.0	GUIDANCE & NAVIGATION	125		125		186			
10.0	INSTRUMENTATION	46							
11.0	COMMUNICATION	424		424		424			
12.0	ENVIRONMENTAL CONTROL	83		150		346			
13.0	GROWTH ALLOWANCE	73		79		92			
14.0	PERSONNEL PROVISIONS								
15.0	CREW STA CONTRL & PAN								
16.0	RANGE SAFETY & ABORT								
SUBTOTALS (DRY WEIGHT)		1,533		1,676		1,946			
17.0	PERSONNEL								
18.0	CARGO								
19.0	ORDNANCE								
20.0	BALLAST								
21.0	RESID PROP & SERV ITEMS	7		7		7			
SUBTOTALS (INERT WEIGHT)		1,540		1,683		1,953			
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES								
24.0	THRUST DECAY PROPELLANT								
25.0	FULL THRUST PROPELLANT								
26.0	THRUST PROP BUILDUP								
27.0	PRE-IGNITION LOSSES								
-	AUX PROPELLANT	20		20		20			
TOTALS (GROSS WEIGHT) (KG)		1,560		1,703		1,973			
DESIGN ENVELOPE VOLUME (FT^3)									
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA (FT^2)									
PRESSURIZED SURF AREA (FT^2)									
DESIGN q, MAX (LB/FT^2)									
DESIGN g, MAX									
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS									

DESIGNATIONS:

CODE, SYSTEM, REF., MIL-M-38310A OR SP-6004

ITEM OR MODULE
- A Unmanned IM
- B
- C Manned Space IM
- D
- E Manned Lunar Landing IM
- F (Incl G&N + Thermal Control Kits)

SPACECRAFT
- M MANNED LAUNCH
- U UNMANNED LAUNCH

NOTES & SKETCHES:

Intelligence Modules, m

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Table 2-13. Crew Module Summary Weight Statement - English Units

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

SPACECRAFT SUMMARY WEIGHT STATEMENT

CONFIGURATION	BY	DATE
CREW MODULES	FGC	January, 1971

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	U
1.0	AERODYNAMIC SURFACES								
2.0	BODY STRUCTURE	2,535		2,535					
3.0	INDUCED ENVIR PROT	340		340					
4.0	LNCH RECOV & DKG	480		480					
5.0	MAIN PROPULSION	-		-					
6.0	ORIENT CONTROL SEP & ULL								
7.0	PRIME POWER SOURCE								
8.0	POWER CONV & DISTR	50		50					
9.0	GUIDANCE & NAVIGATION	190		•210					
10.0	INSTRUMENTATION								
11.0	COMMUNICATION	300		300					
12.0	ENVIRONMENTAL CONTROL	190		•190					
13.0	GROWTH ALLOWANCE	585		605					
14.0	PERSONNEL PROVISIONS	1,640		1,835					
15.0	CREW STA CONTRL & PAN	155		155					
16.0	RANGE SAFETY & ABORT								
SUBTOTALS (DRY WEIGHT)		6,465		6,700					
17.0	PERSONNEL @ 200 LB EACH	1,200		800					
18.0	CARGO FOOD, ETC.	485		1,620					
19.0	ORDNANCE N_2 AND TK	20		20					
20.0	BALLAST EVA	-		360					
21.0	RESID PROP & SERV ITEMS								
SUBTOTALS (INERT WEIGHT)		8,170		9,500					
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES								
24.0	THRUST DECAY PROPELLANT								
25.0	FULL THRUST PROPELLANT								
26.0	THRUST PROP BUILDUP								
27.0	PRE-IGNITION LOSSES								
	META AND EPS O_2	515		2,605					
	EPS H_2	45		270					
TOTALS (GROSS WEIGHT) (LB)		8,730		12,375					
DESIGN ENVELOPE VOLUME (FT^3)									
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA (FT^2)									
PRESSURIZED SURF AREA (FT^2)									
DESIGN q, MAX (LB/FT^2)									
DESIGN g, MAX									
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS									

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

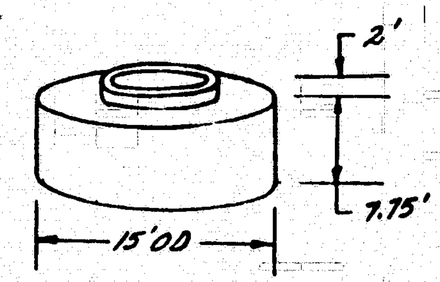
ITEM OR MODULE

A	CM - 6 Men/7 Days, Space
B	
C	CM - 4 Men/45 Days, Lunar Surface
D	
E	
F	

SPACECRAFT

M	MANNED LAUNCH
U	UNMANNED LAUNCH
•	SYS INCR DUE TO LUN SURF INCL IN LM WT

NOTES & SKETCHES:



Crew Modules, e

FORM 3942-Z NEW 2-70

Space Division
North American Rockwell

Table 2-14. Crew Module Summary Weight Statement - Metric Units

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

SPACECRAFT SUMMARY WEIGHT STATEMENT

CONFIGURATION: CREW MODULES | BY: [illegible] | DATE: January, [illegible]

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	U
1.0	AERODYNAMIC SURFACES								
2.0	BODY STRUCTURE	1,150		1,150					
3.0	INDUCED ENVIR PROT	154		154					
4.0	LNCH RECOV & DK@218	218		218					
5.0	MAIN PROPULSION								
6.0	ORIENT CONTROL SEP & ULL								
7.0	PRIME POWER SOURCE								
8.0	POWER CONV & DISTR	23		23					
9.0	GUIDANCE & NAVIGATION	86		95					
10.0	INSTRUMENTATION								
11.0	COMMUNICATION	136		136					
12.0	ENVIRONMENTAL CONTROL	86		86					
13.0	GROWTH ALLOWANCE	265		274					
14.0	PERSONNEL PROVISIONS	745		832					
15.0	CREW STA CONTRL & PAN	70		70					
16.0	RANGE SAFETY & ABORT								
SUBTOTALS (DRY WEIGHT)		2,933		3,038					
17.0	PERSONNEL @ 91 KG EACH	544		363					
18.0	CARGO FOOD, ETC.	220		735					
19.0	ORDNANCE N_2 AND TK	9		9					
20.0	BALLAST EVA	-		108					
21.0	RESID PROP & SERV ITEMS								
SUBTOTALS (INERT WEIGHT)		3,706		4,308					
22.0	RES PROP & SERV ITEMS								
23.0	INFLIGHT LOSSES								
24.0	THRUST DECAY PROPELLANT								
25.0	FULL THRUST PROPELLANT								
26.0	THRUST PROP BUILDUP								
27.0	PRE-IGNITION LOSSES								
	META AND EPS O_2	234		1,182					
	EPS H_2	20		133					
TOTALS (GROSS WEIGHT) (KG)		3,960		5,613					
DESIGN ENVELOPE VOLUME (FT^3)									
PRESSURIZED VOLUME (FT^3)									
DESIGN ENVEL SURF AREA (FT^2)									
PRESSURIZED SURF AREA (FT^2)									
DESIGN q, MAX (LB/FT^2)									
DESIGN g, MAX									
DESIGN POWER, MAX (KW)									
DESIGN NO. MEN/DAYS									

DESIGNATIONS:

CODE, SYSTEM, REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

- A CM - 6 Men/7 Days, Space
- B
- C CM - 4 Men/45 Days, Lunar Surface
- D
- E
- F

SPACECRAFT

- M MANNED LAUNCH
- U UNMANNED LAUNCH

NOTES & SKETCHES:

Crew Modules, m

FORM 3942-Z NEW 2-70

module 15.0 feet (4.57 meters) in diameter are assumed. The gear semispan is $\sqrt{2}$ times the c.g. distance. The gear weight equation is

$$WLLG = 118.2\ (S) - 21.7\ (W_L/10^3) + 151.3/SXW_L/10^5) - 1.060$$

where S is the semispan in feet and W_L is the landing weight in pounds. A plot of this equation is presented in Figure 2-30. At a landing weight of 50,000 pounds (22,730 kilogram), the landing gear weight is

$$WLLG = 274.0 \times CG - 2{,}145\ \text{lb (CG in ft)}$$

$$= (407.0 \times CG - 963\ \text{kg) (CG in m)}$$

At a c.g. distance from the ground of 20.0 feet (6.1 meter), the landing gear weight is

$$WLLG = 3{,}335\ \text{lb (1510 Kg)},\ CG = 20.0\ \text{ft (6.1 m)}$$

From the data ploted in the figure the sensitivity of the weight is

$$(WLLG/W_{LAND}) = (10\ \text{to}\ 40)\ \text{lb}/1000\ \text{lb}$$

$$= ((10\ \text{to}\ 40)\ \text{kg}/1000\ \text{kg})$$

$$(WLLG/CG) = (260\ \text{to}\ 330)\ \text{lb/ft}$$

$$= ((386\ \text{to}\ 490)\ \text{kg/m})$$

It would appear, therefore, that attention to the cg will result in a lighter gear weight than the same degree of attention to the spacecraft landing weight.

Manipulator Submodule (MM)

The manipulator submodule or kit consists of a pressurized module 9.0 feet (2.74 meter) in diameter and 3.0 feet (0.91 meter) long, with docking systems configured to be compatible with the systems selected for the other major elements of the space program, and the tug derivatives. Attached to this module is a set of articulating mechanical arms operated by controls within the MM. Pressurized hatches are installed on both ends, to permit crew access to the manipulator drive systems from either side of the MM. The gross weight of this module kit is 2200 pounds (998 kg). A weight breakdown is shown in Table 2-15.

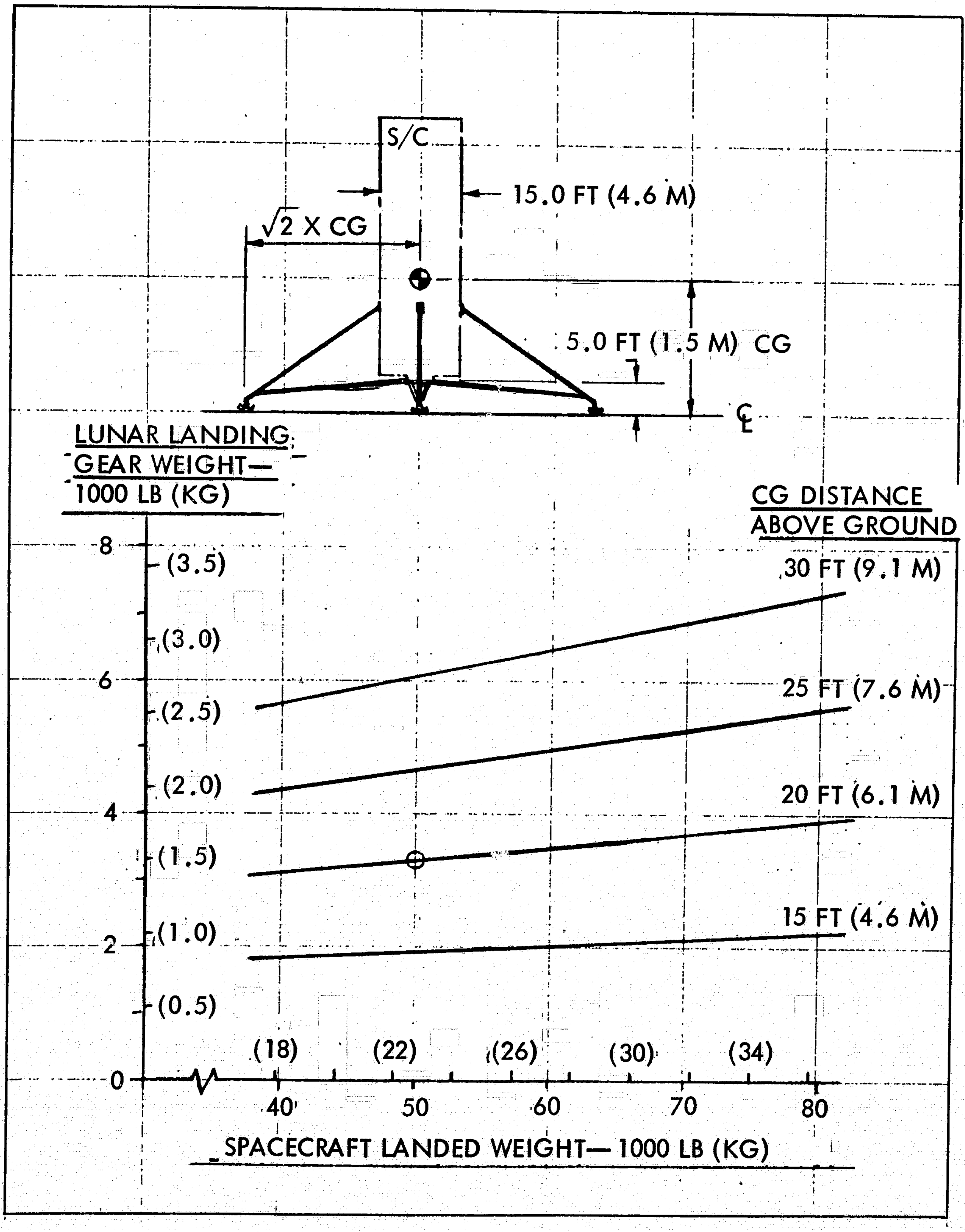


Figure 2-30. Landing Gear Weight Versus Spacecraft Weight and Center of Gravity

Table 2-15. Estimated Manipulator Module Description and Delta Weights

Category	Weight (lb)	Power (w)
Exterior manipulator kit	(486)	(205)
Manipulator arm assemblies (2)	280	40 (400 peak)
Torque and force restraining arms (2)	20	5
Tool attachments and fixtures	46	25
Manipulator/docking port attachment bracket	20	—
Control electronics	50	100
Video cameras (2)	50	35
Electrical power distribution	20	—
Crew interface kit	(60)	(115)
Manual controls	20	30
Binocular video viewer	20	70
Electrical power distribution	10	—
Communications and data management	10	15
Electrical power system at 1 lb/watt	(320)	
Module structure	(254)	
Module insulation and meteoroid protection	(100)	
Docking provisions, pressurized	(980)	
1 Active docking system	520	
1 Passive docking system	280	
1 Pressure hatch	180	
Total estimated module weight	2,200 pounds (998 kilograms)	

Docking Provisions

Two systems were considered. The NASA neuter docking system and the Apollo probe-drogue system. The weight of these systems is presented in Table 2-16. A comparison of these concepts is illustrated in Figures 2-31 and 2-32.

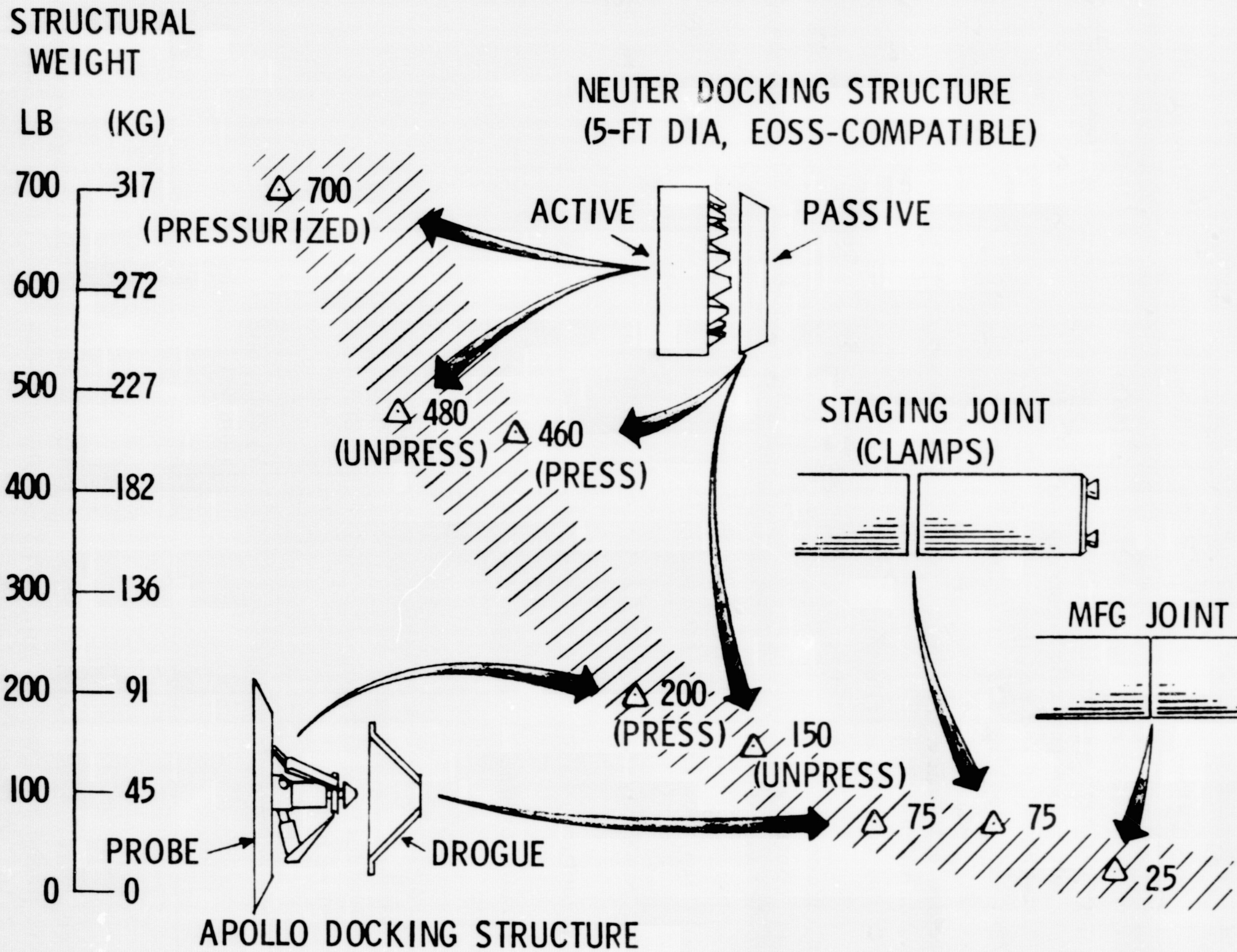
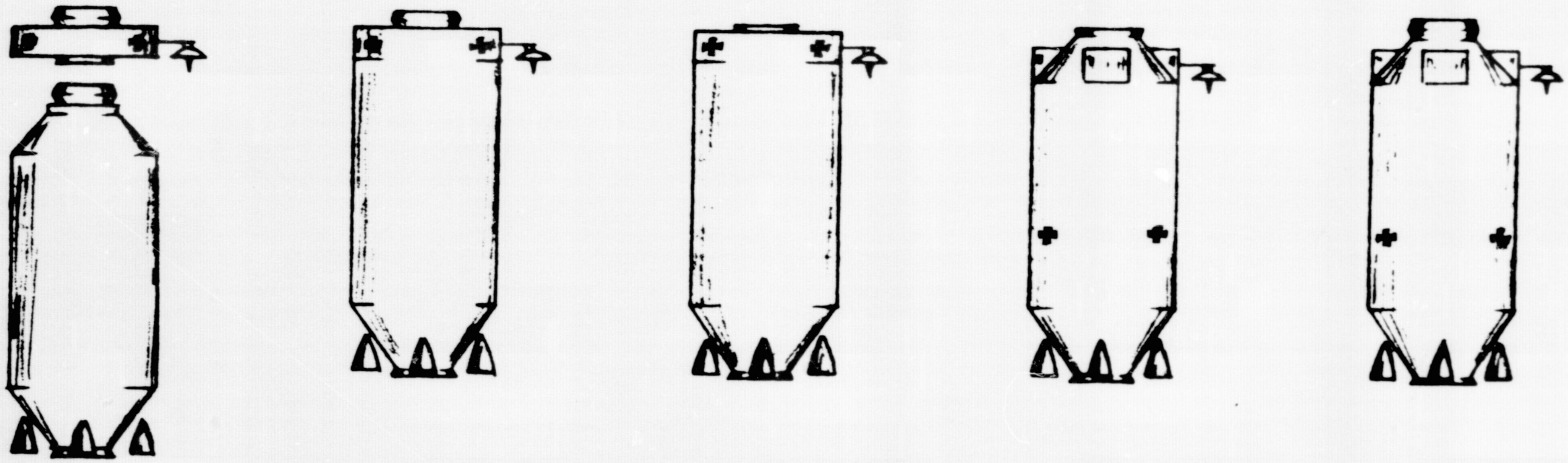


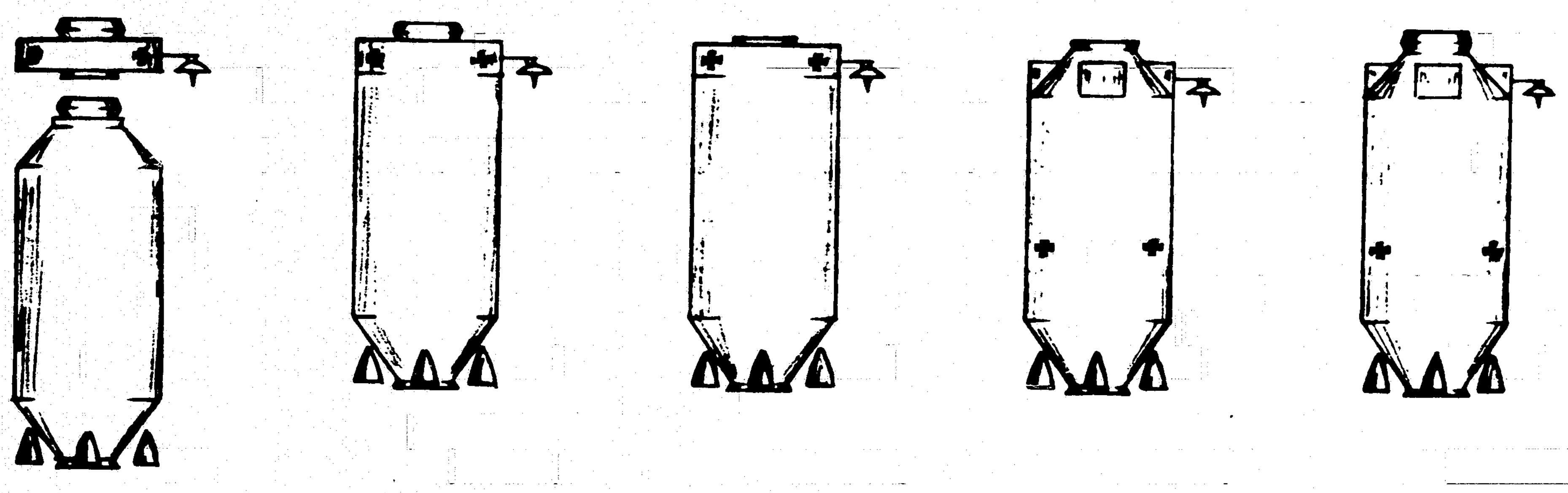
Figure 2-31. Docking Concept Weight Comparison



• SPACE ASSY OF IM TO PM • ACTIVE DOCKING	• NOMINAL CONFIGURATION • GROUND ASSY • ACTIVE DOCKING	• NOMINAL CONFIGURATION • GROUND ASSY • PASSIVE DOCKING	• INTEGRATE IM COMP INTO PM • PASSIVE DOCKING	• INTEGRATE IM COMPONENTS INTO PM • ACTIVE DOCKING
		BASE POINT*		
IM + 620 LB PM + 680 LB	IM + 330 PM + 0	IM 2,575 PM 5,350	IM - 280 PM + 280	IM - 280 PM + 610
TOT + 1,300 LB (+ 590 KG)	TOT + 330 LB (+ 150 KG)	TOT 7,925 LB (3,590 KG)	TOT 0	TOT + 330 LB (+ 150 KG)

*REF HEO 15/80 DRY WEIGHT LESS GROWTH ALLOWANCE

Figure 2-32. IM/PM/Docking Interface Weight Trends



• SPACE ASSY OF IM TO PM • ACTIVE DOCKING	• NOMINAL CONFIGURATION • GROUND ASSY • ACTIVE DOCKING	• NOMINAL CONFIGURATION • GROUND ASSY • PASSIVE DOCKING	• INTEGRATE IM COMP INTO PM • PASSIVE DOCKING	• INTEGRATE IM COMPONENTS INTO PM • ACTIVE DOCKING
		BASE POINT*		
IM + 620 LB PM + 680 LB	IM + 330 PM + 0	IM 2,575 PM 5,350	IM - 280 PM + 280	IM - 280 PM + 610
TOT + 1,300 LB (+ 590 KG)	TOT + 330 LB (+ 150 KG)	TOT 7,925 LB (3,590 KG)	TOT 0	TOT + 330 LB (+ 150 KG)

*REF HEO 15/80 DRY WEIGHT LESS GROWTH ALLOWANCE

Figure 2-32. IM/PM/Docking Interface Weight Trends

Table 2-16. Docking Provisions Weight Summary

Item	Active lb	Active kg	Passive lb	Passive kg
NASA neuter docking				
Pressurized	(700)	(318)	(460)	(209)
Docking port	520	236	280	127
Pressure hatch (5 ft, 1.5 meters)	180	82	180	82
Unpressurized	(480)	(218)	(150)	(68)
Docking port	480	218	150	68
Apollo docking				
Probe	75 lb (34 kilograms)			
Drogue with hatch (2.5 ft, 0.76 meters)	200 lb (91 kilograms)			

Diameter and Length Comparisons

The weight and length effects on the tug spacecraft because of a change in outer diameter is primarily concentrated in the PM sizing. Appendix C develops the parametric sizing of the PM with respect to outer diameter and number of LO_2 tanks (one and four), as well as comparing the initial parametric weight. Two figures from this appendix are duplicated here to summarize these effects. Figure 2-33 illustrates the PM structural weight as a function of outer diameter and propellant capacity while Figure 2-34 illustrates the length of the PM as a function of its contained propellant load.

At low propellant capacities, the smaller-diameter PM's are lighter. However, increasing propellant loads cause the structural loads to increase rapidly, creating a condition in which the stage weight of the smaller-diameter PM is larger than a more moderate diameter PM. At PM diameters greater than 20 feet (6 meters), the packaging efficiency of the propellant is low, and the stage weight is therefore larger than the more moderate diameters. The figure shows that selection of stage diameters of 14 feet (4 meters) to 17 feet (5 meters) for propellant loads of 50,000 to 80,000 pounds (14,000 to 36,000 kilogram) result in the lightest module weights.

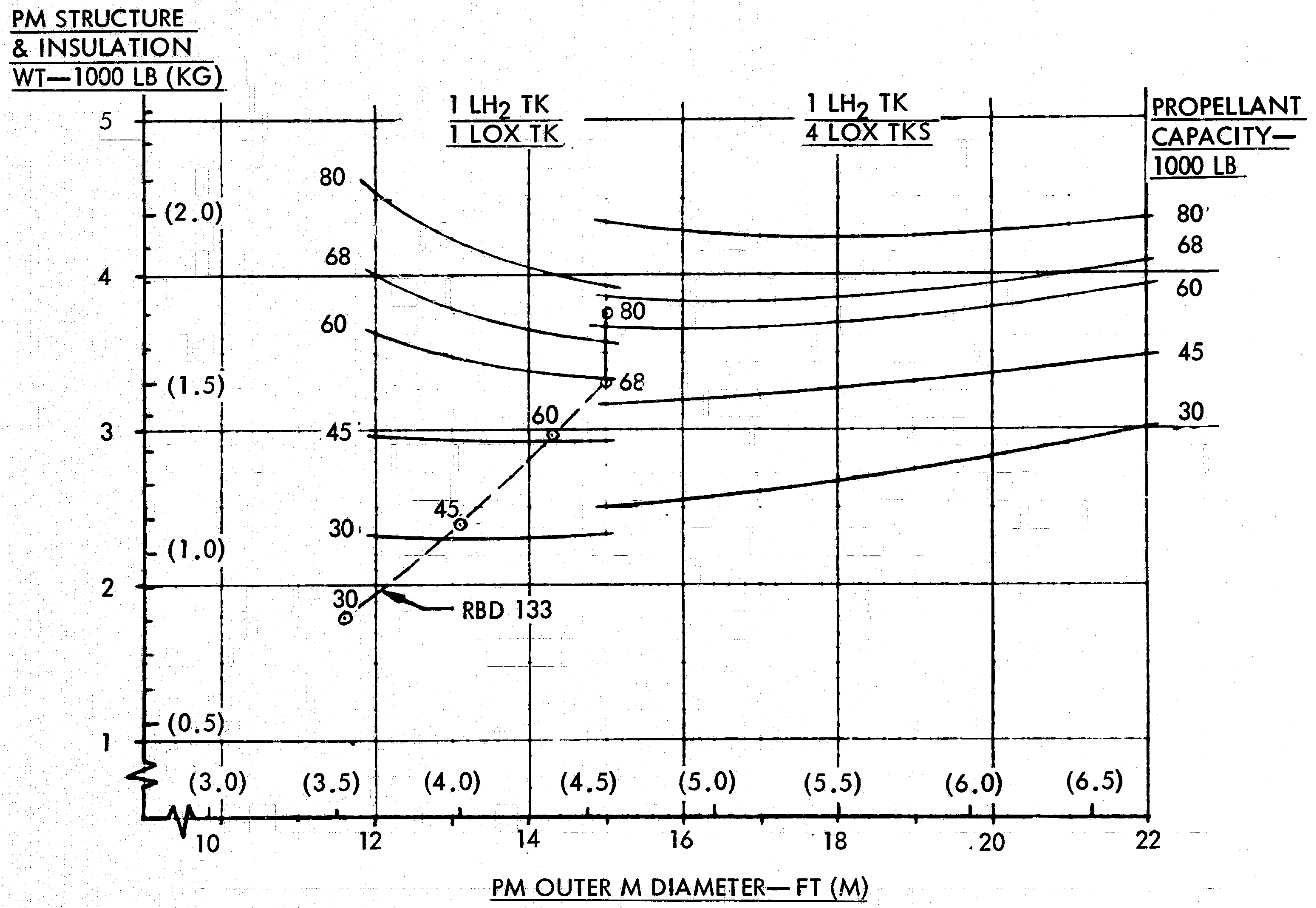


Figure 2-33. PM Structure Weight Versus PM Diameter

Space Division
North American Rockwell

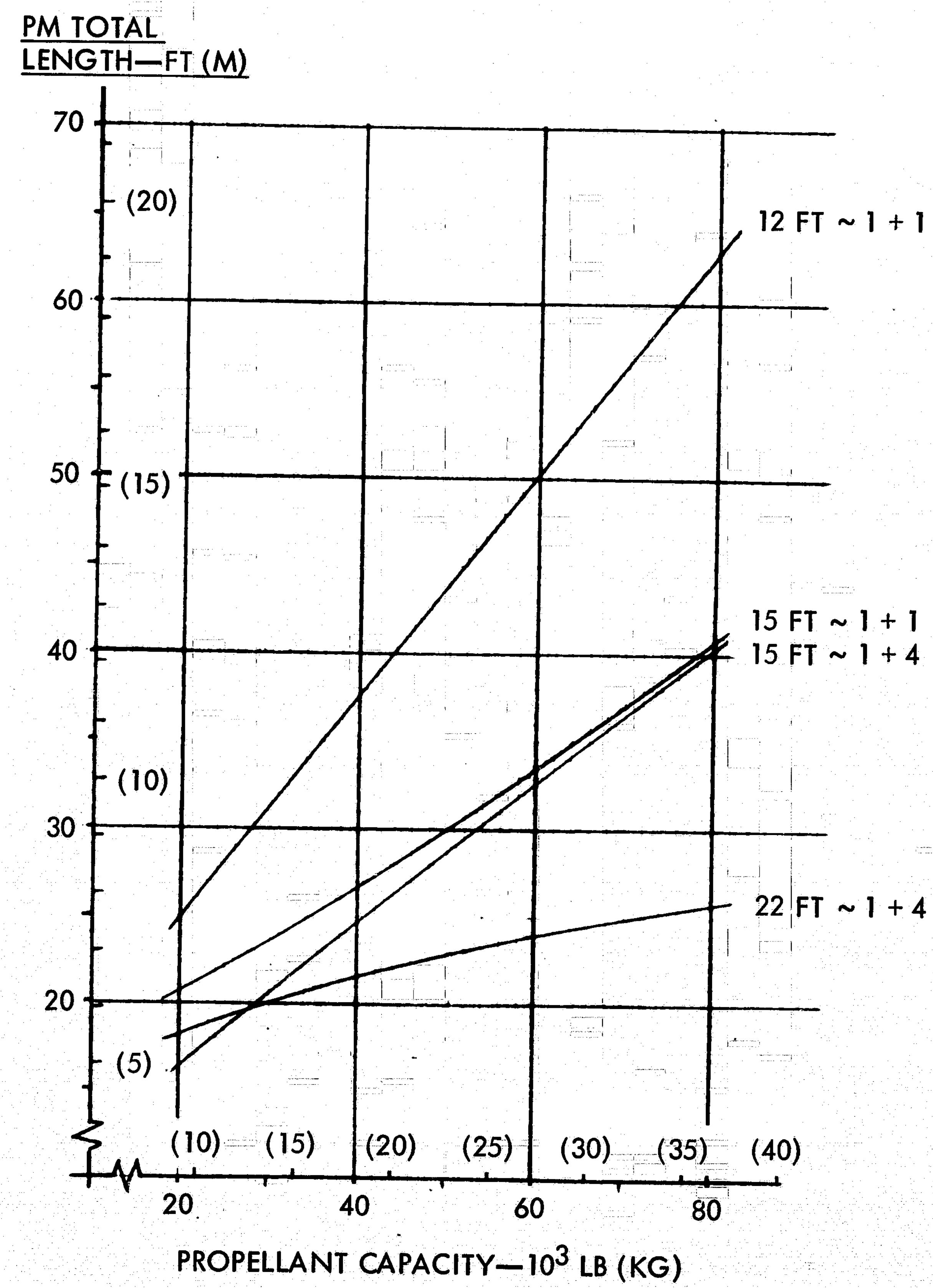


Figure 2-34. PM Length Versus Propellant Weight

IM Weight Comparisons

Figure 2-35 presents results of comparing three IM diameters, their resulting module characteristics, and their structure weight trend. The 12-ft-(3.66-m) diameter is good for low weight and for use of the internal volume.

The 15-ft-diameter (4.57-m) module also uses space well, and the interior hole provides a good base for partial enclosure of docking adapter structure.

The 22-ft diameter (6.71-m) is too large to permit good volume use, and the docking support structure does not integrate well because of a too-large diameter hole. However, the module could be pressurized for access from the CM via the interior diameter.

PM Aft End Design Weight Comparison

Table 2-17 summarizes the weights associated with three configurations of the four-engine/neuter docking ring and a one-engine/no docking configuration. Compared to the fixed four-engine configuration, the swing-out or swing-out-and-down engine configurations add 600 to 700 pounds (270 to 320 kilogram) of additional mechanism and structural beefup. When the thermal protection on the CM is included, the weight penalty increases to approximately 800 to 950 pounds (360 to 430 kilogram).

Landing Gear Weight Comparisons

Based on the parametric lunar landing gear weight, and on the conceptual designs developed during the study, a weight comparison was made of the landing gear based on outer diameters of 12, 15 and 22 feet (3.66, 4.57, and 6.71 meters) and crew module location (top or bottom). A diagram of these six configurations and the resulting center of gravity and landing gear weight is presented in Figures 2-36 and 2-37.

Crew Module Structural Weight Trend

The primary structural weight of the CM as a function of gross volume is shown in Figure 2-38. The gross module height of a single floor is about eight feet (2.5 meters) and two floors about seventeen feet (5 meters). For a four-man crew with a free volume of 250 cubic feet, (7 meters^3) each and

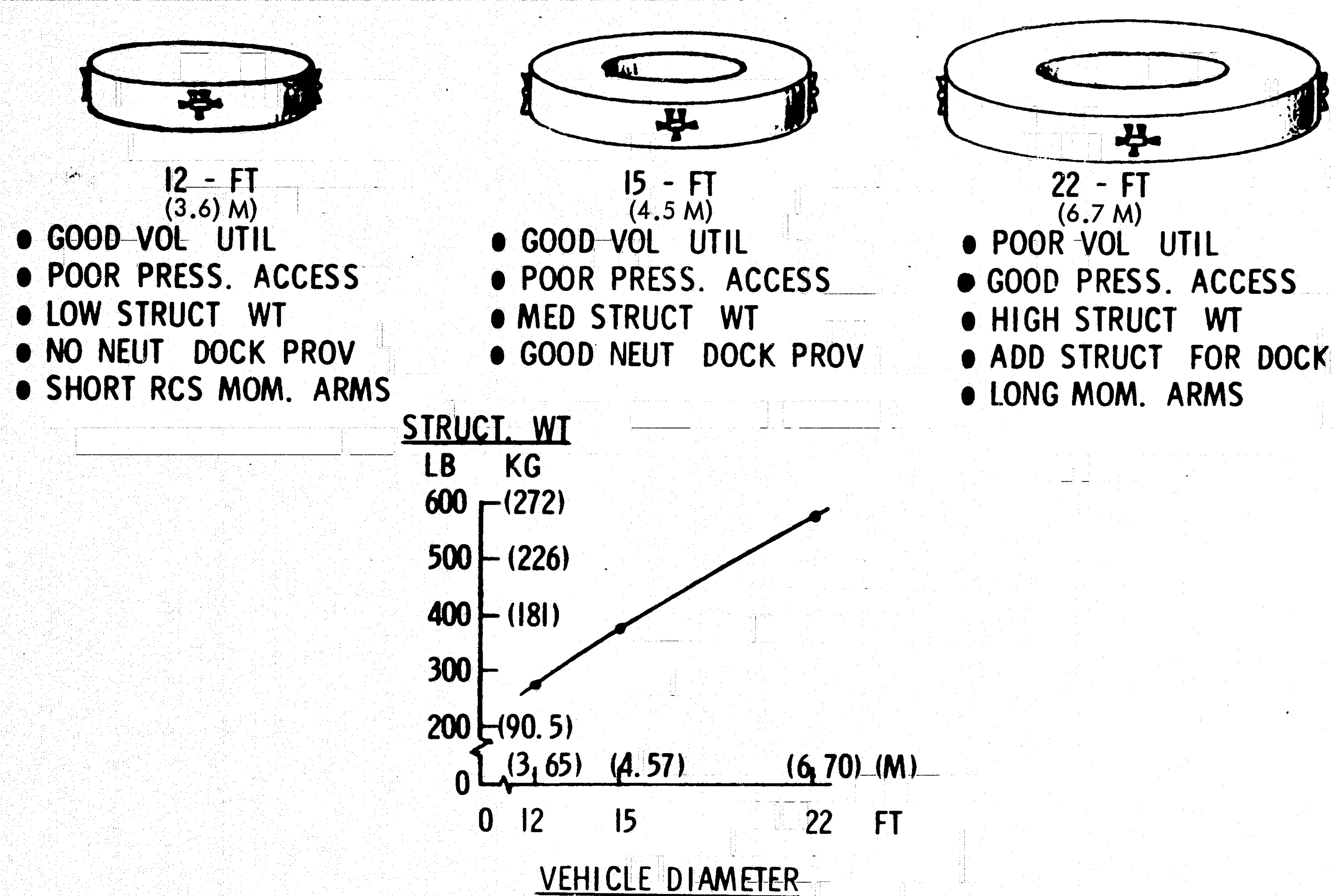


Figure 2-35. Intelligence Module Diameter Influences

Table 2-17. PM Aft End Design Weight Comparison
(15-ft Outside Diameter)

Component	1 Engine, No Dock		4 Engines ~ Passive Neuter Docking: Fixed		4 Engines ~ Passive Neuter Docking: Swing Out		4 Engines ~ Passive Neuter Docking: Out & Down	
LO_2 Support frame	1	100	1	100	1	100	1	100
LO_2 Support beams			4	60	4	60	4	60
Cylindrical skirt			1	110	1	110	1	120
Conical thermal structure	1	130						
Support frame(s)	1	50	2	200	3	300	3	300
Aft bulkhead			1	70	1	70	1	70
Thrust fittings	1	80	4	80	4	80	4	80
Docking ring			1	150	1	150	1	150
Links and joints					8	150	16	200
Engine yokes							4	80
Actuators					8	120	8	80
Screwjacks							8	100
Feedlines, valves, etc.	2	160	8	200	8	200	8	200
Additional swivel jets					24	240	16	160
Σ PM, (kg) lb	(236) 520		(440) 970		(717) 1580		(771) 1700	
CM insulation* (kg) lb					80% CM (154) 340		20% CM (41) 90	
ΣPM + CM, (kg) lb	(236) 520		(440) 970		(871) 1920		(812) 1790	
Δwt to fixed four-eng., (kg) lb	(-204) -450		(0) 0		(+431) +950		(+372) +820	

*1/2-inch cork, 1.3 psf, 330 sq et.

an allowance of 250 cubic feet (7 cubic meters^3) for structure and equipment, a gross volume of about 1250 cubic feet (35 cubic meters^3) is required. This coincides with a single-floor module 15 feet (4.6 meters) in diameter. Two floors 12 feet (3.7 meters) in diameter result in a volume increase of 28 percent and a weight increase of 34 percent. A single-floor diameter larger than 15 feet (4.6 meters) or two floors over 12.0 feet (3.7 meters) is not required.

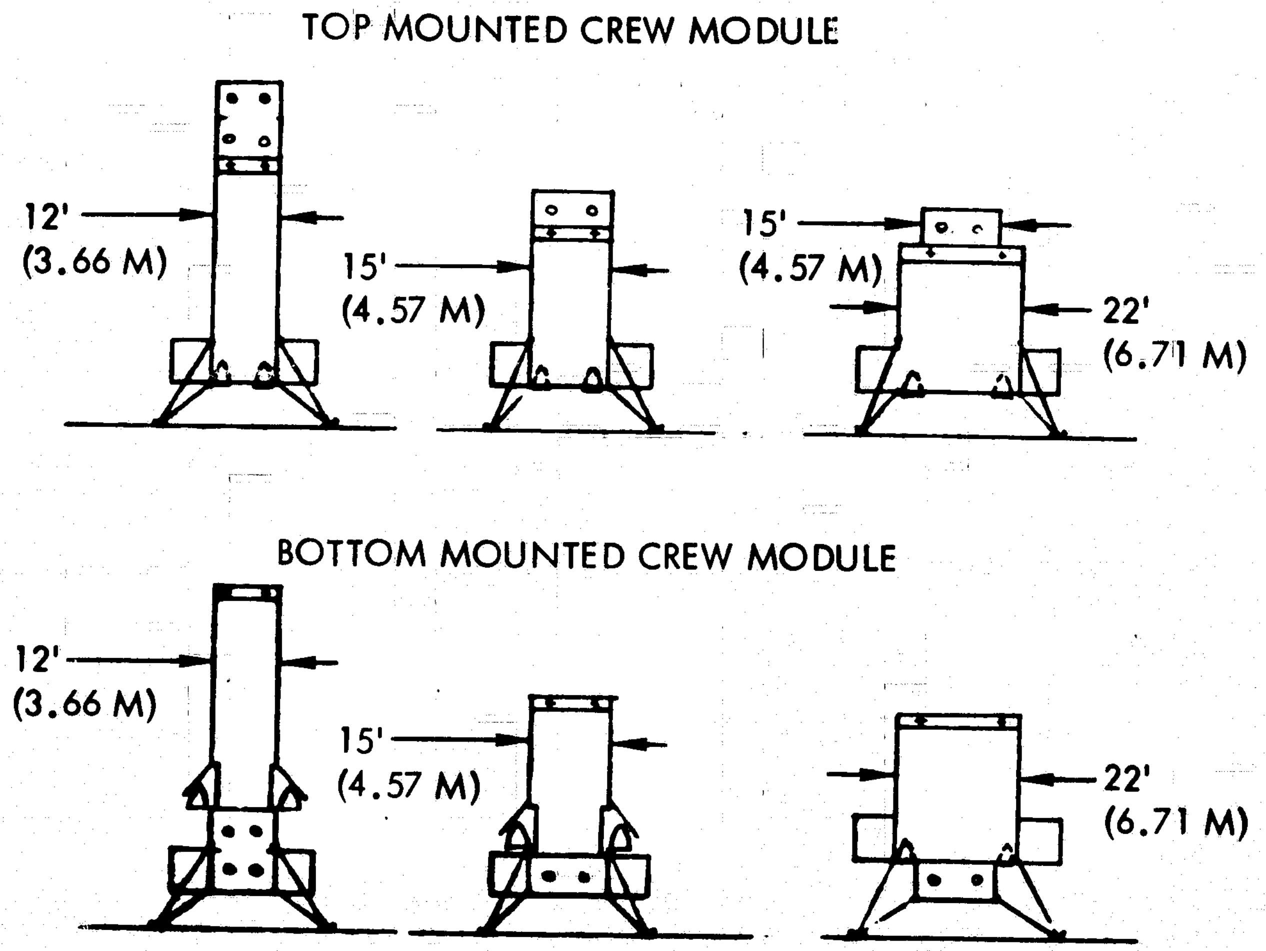


Figure 2-36. Lunar Landing Configuration—Mode A-1

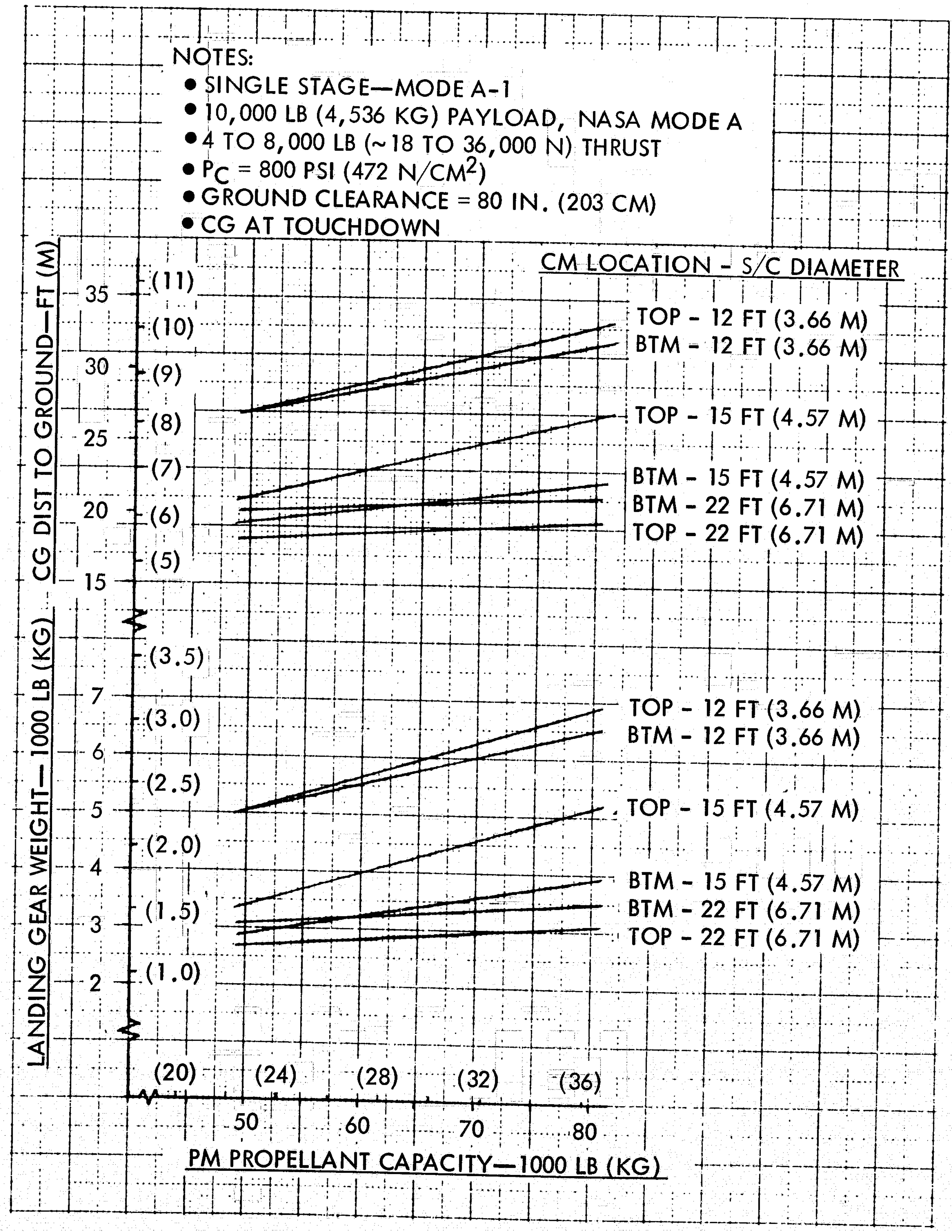


Figure 2-37. Lunar Landing Gear Weight Comparison

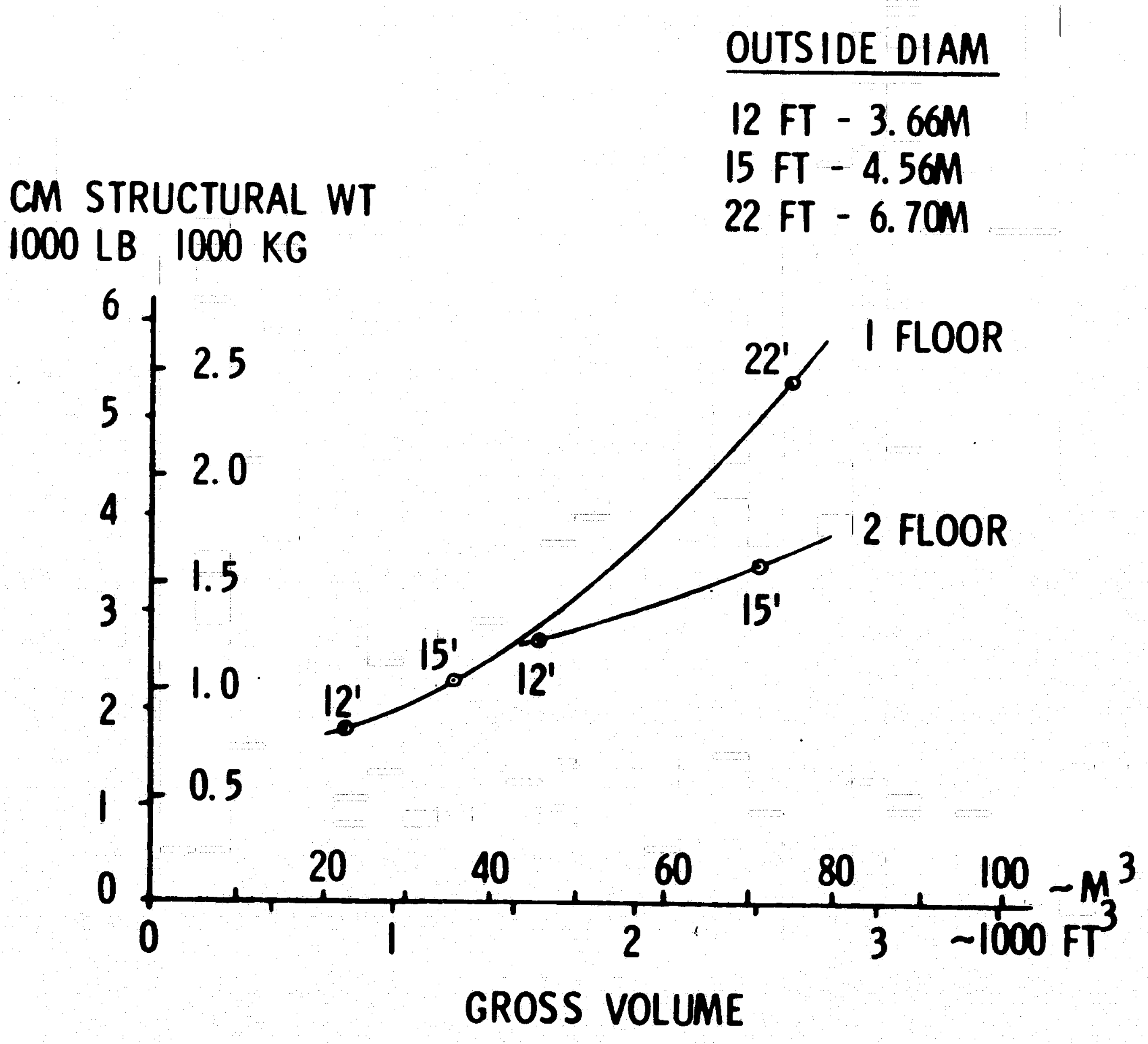


Figure 2-38. Crew Module Structural Weight Trend With Volume

The weight of a one-floor, 15-ft-diameter (4.6 m) crew module designed to accommodate four men for durations up to 45 days is presented in Figure 2-39. The inert weight, including solid expendables (food, clothing, towels, filters, etc.) is little affected by duration. The reactants of the EPS fuel cells (also crew water supply), however, strongly affect CM weight. Addition of the crew, metabolic oxygen, and the required module atmosphere yields the module gross weight. For durations of 14 to 45 days, the total CM weight increases from 10,000 pounds (4540 kilogram) to 13,000 pounds (5800 kilogram), or about 30 percent.

CREW MODULE WEIGHT TREND WITH DURATION

. 1 FLOOR 15 FT (4.6M) DIAMETER
. 4 MAN CREW
. INCL 14 DAY CONTINGENCY OPERATION

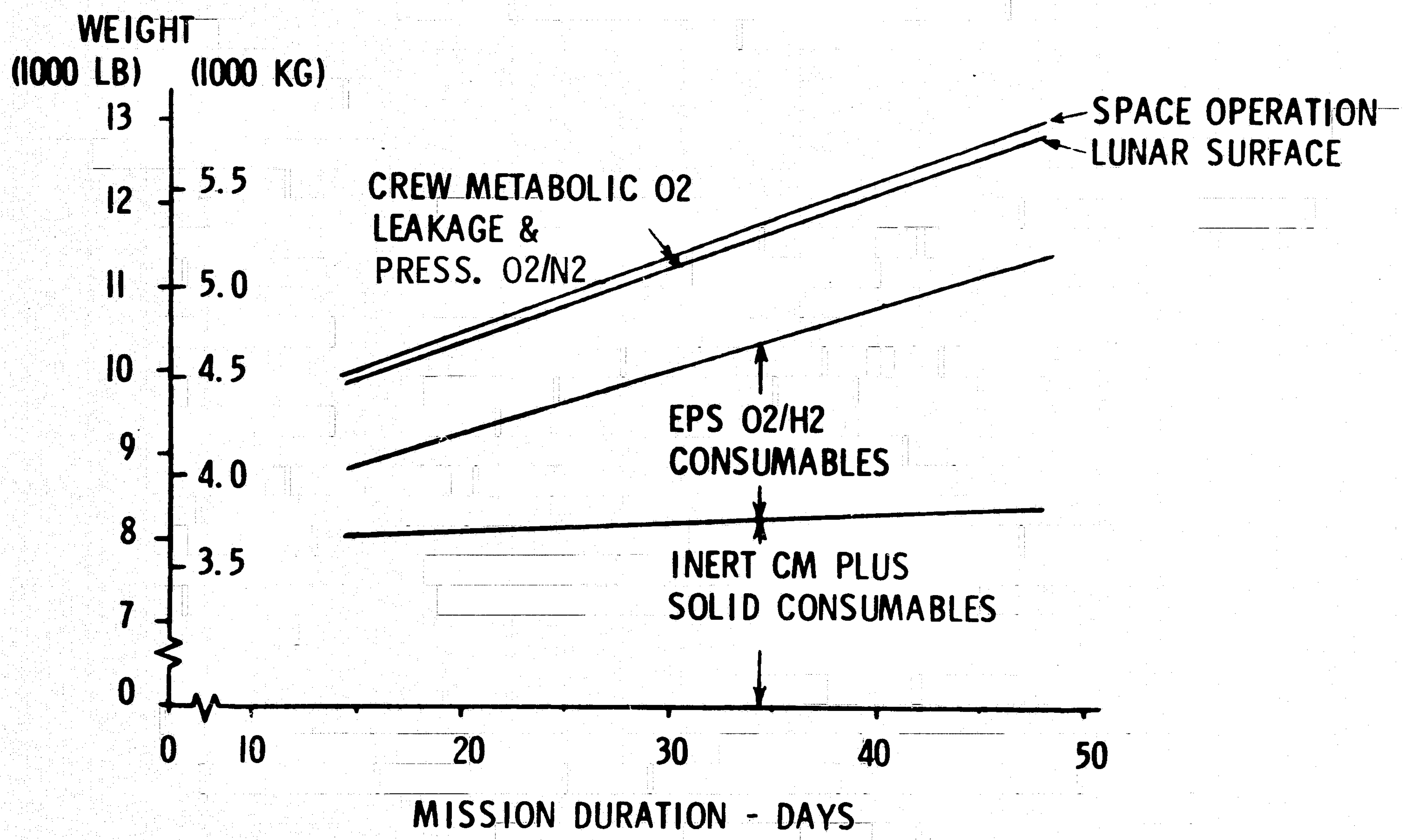


Figure 2-39. Crew Module Weight Trend With Duration

3.0 CONCEPT REFINEMENT STUDIES

3.1 OBJECTIVE AND SCOPE OF ACTIVITY

Activity in the later part of the study consisted primarily of refining each of the three design concepts selected at the midterm point, and investigating several alternate concepts as well as completing a number of studies.

3.1.1 SELECTED CONCEPTS REFINEMENT STUDIES

The three concepts presented in Figure 3-1 were subjected to detailed conceptual design and systems/subsystems analysis. The objective was to refine the designs to insure that the configurations and systems involved were feasible, and the important interfaces were identified, and presented in this report to provide a solid basis for weights analysis and subsequent costing and development planning studies. The resulting weights are somewhat different from those used at the mid-term point as a result of refined systems analyses. Concept 1 is lighter and Concepts 5 and 11 are heavier due to more realistic propulsion and structural weights. All performance sizing for these designs are based on a geosynchronous mission which starts and ends at 100 nautical miles altitude.

3.1.2 SPECIAL STUDIES

A number of studies were undertaken in conjunction with the design refinement effort because of their unusual interest or importance in respect to the baseline selected concepts. These include the effect of ground-based operations on design and reduced system autonomy and flexibility on Tug design characteristics, expendable tug design concepts, minitug concepts, assessment of design sensitivity, and the possibilities for alternate use of tug modules. A study of special lunar lander configurations was conducted earlier, but information concerning it is included in this section for convenience.

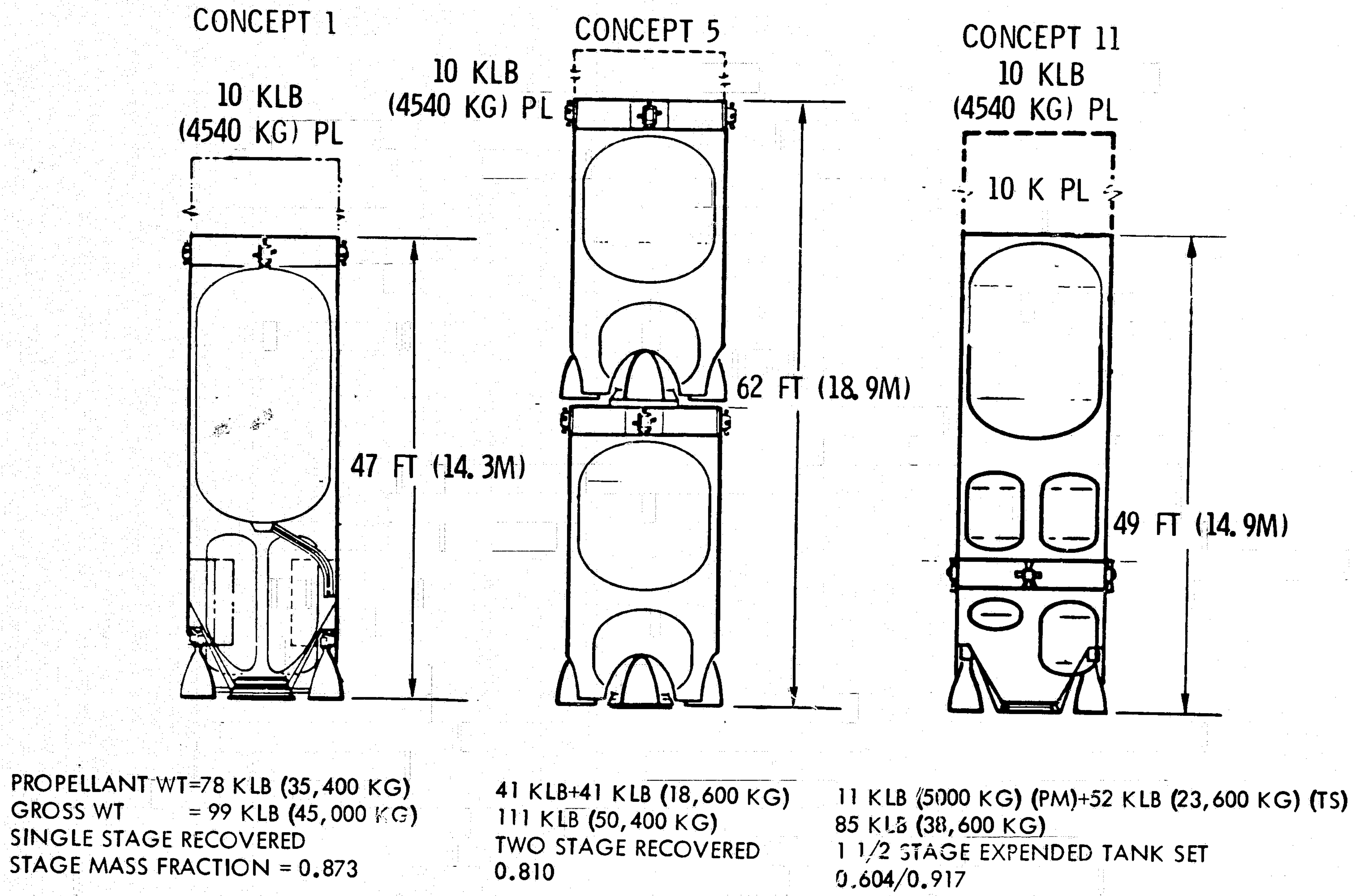


Figure 3-1. Reusable Geosynchronous Mission Space Tug Concepts

3.2 SELECTED CONFIGURATIONS REFINEMENT (FIGURES 3-2, 3-3, AND 3-4)

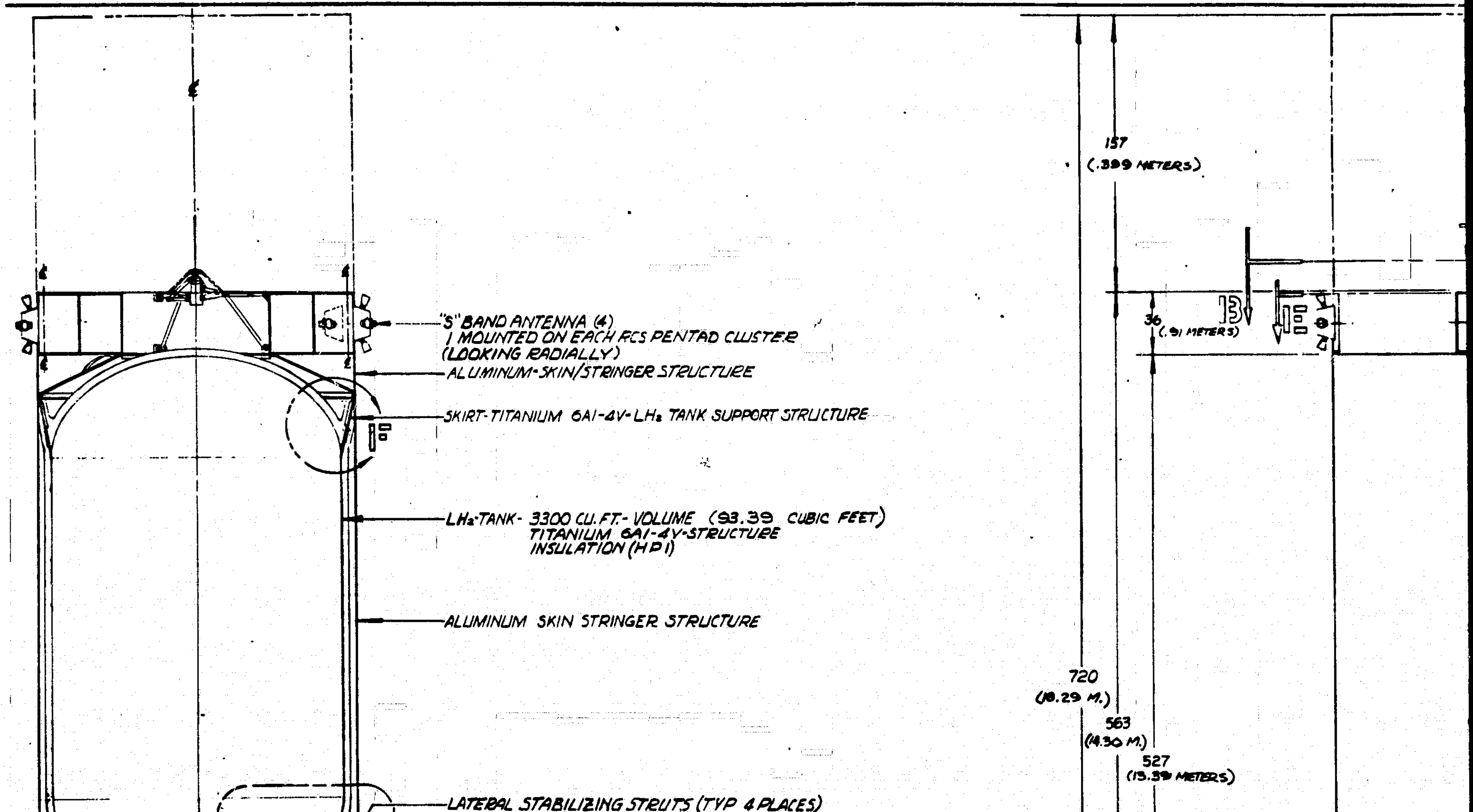
3.2.1 BACKGROUND

Three basic concepts were selected at study mid-term from the eleven baselines of Section 2.1 for further refinement during the remaining study period. The three were Concepts 1, 5, and 11. These concepts are most adaptable to the multipurpose design approach. The other concepts were eliminated from further consideration because of excessive costs and/or inadequate multipurpose capability. Concept 1 is a single-stage vehicle of optimum sizing intended to perform a geosynchronous mission of 10k (4536 kg) payload placement and then return. Concept 5, of optimum size, must perform a geosynchronous mission; it consists of two recoverable stages. Concept 11 is also sized to be optimum for the geosynchronous mission. It consists of a small recoverable propulsion stage and a large tank set which is expended in geosynchronous orbit along with the payload.

During design refinement, the selected baseline concepts were configured to incorporate all latest changes and updates from earlier, preliminary work performed in the various subsystems areas. The vehicles also were resized to be consistent with updated performance and weight synthesis analysis. Each vehicle design was based on a philosophy of maximum multipurpose modularity. The scope of missions included a lunar landing, although scar weights for this application were not included. However, the structural configurations utilized at the base of the vehicle (i.e., four engines, four O_2 tanks, central docking, cross beams) would allow addition of landing gear at a later time.

3.2.2 CONCEPT 1, SINGLE-STAGE RECOVERABLE

The refinement of Concept 1 is shown on Figure 3-2. This single-stage vehicle is comprised of a PM with an IM attached to the forward end. The PM is of sufficient capacity to deliver 80,000 pounds (36,287 kg) of propellant to the main engines. This capacity is based on delivery of 10,000 pounds (4536 kg) to geosynchronous orbit and return to a low earth orbit.



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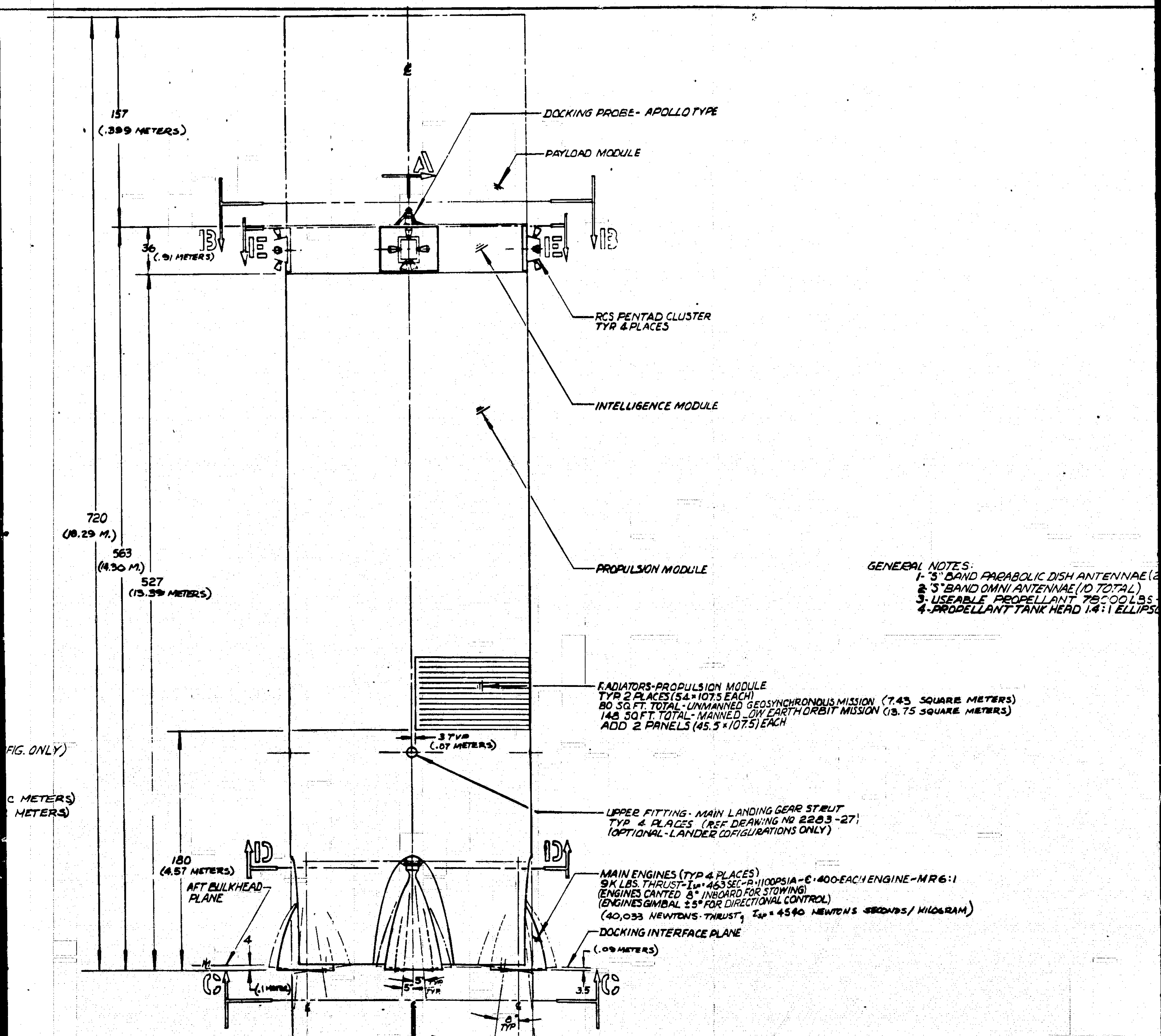

Figure 3-2. Concept 1, Space Tug, 80,000 F

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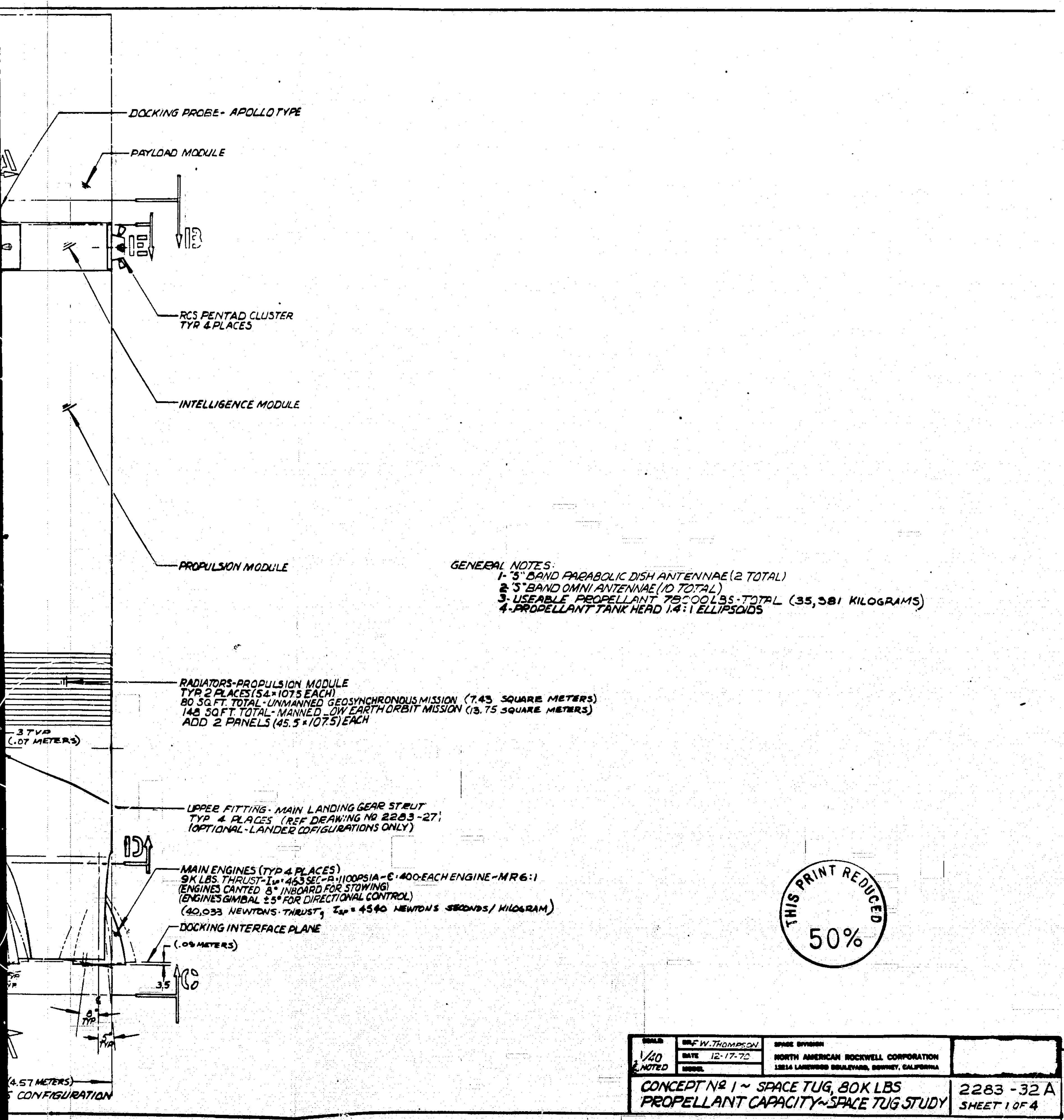
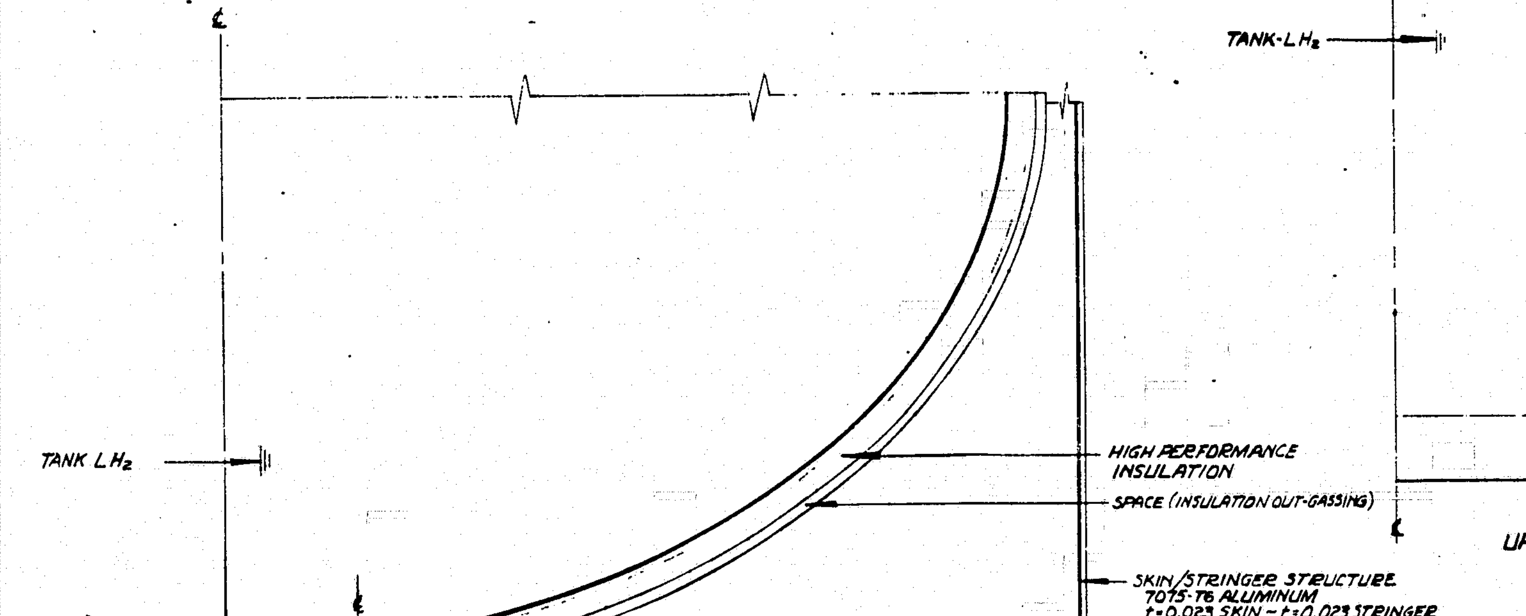


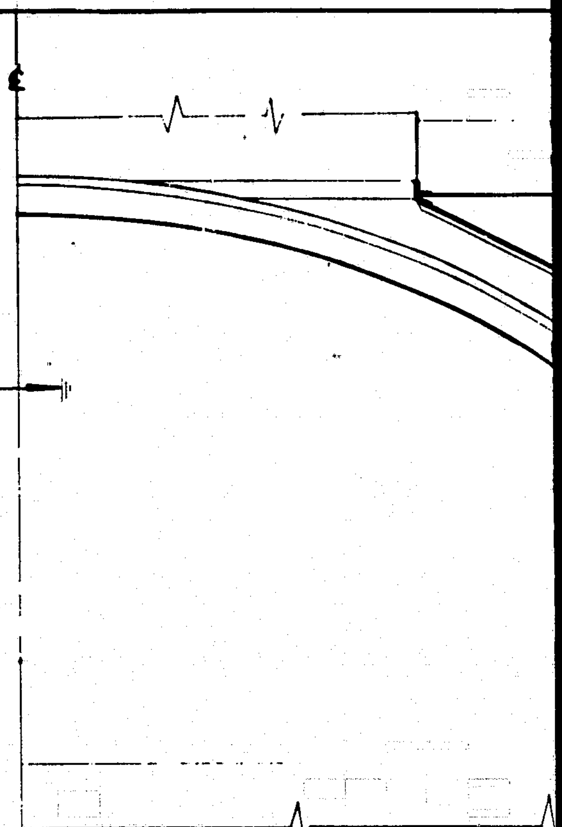
Figure 3-2. Concept 1, Space Tug, 80,000 Pounds Propellant Capacity



VIEW C/A

TYPICAL LATERAL SUPPORT STRUCTURE & INSULATION-LH2 TANK

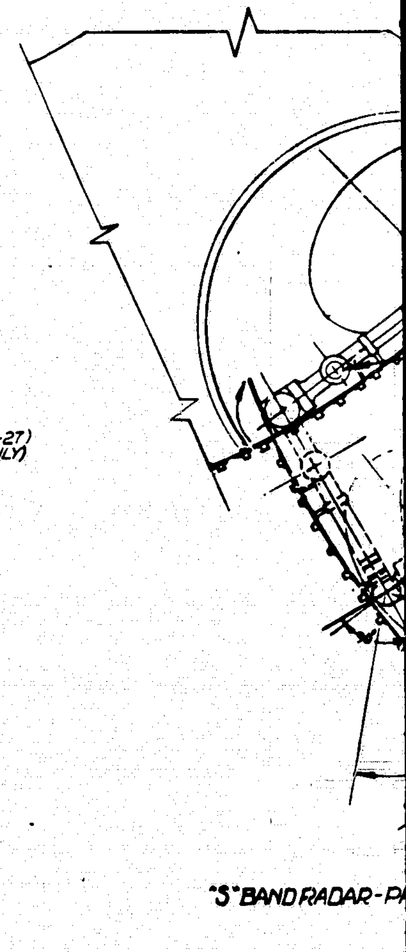
SCALE 1/10



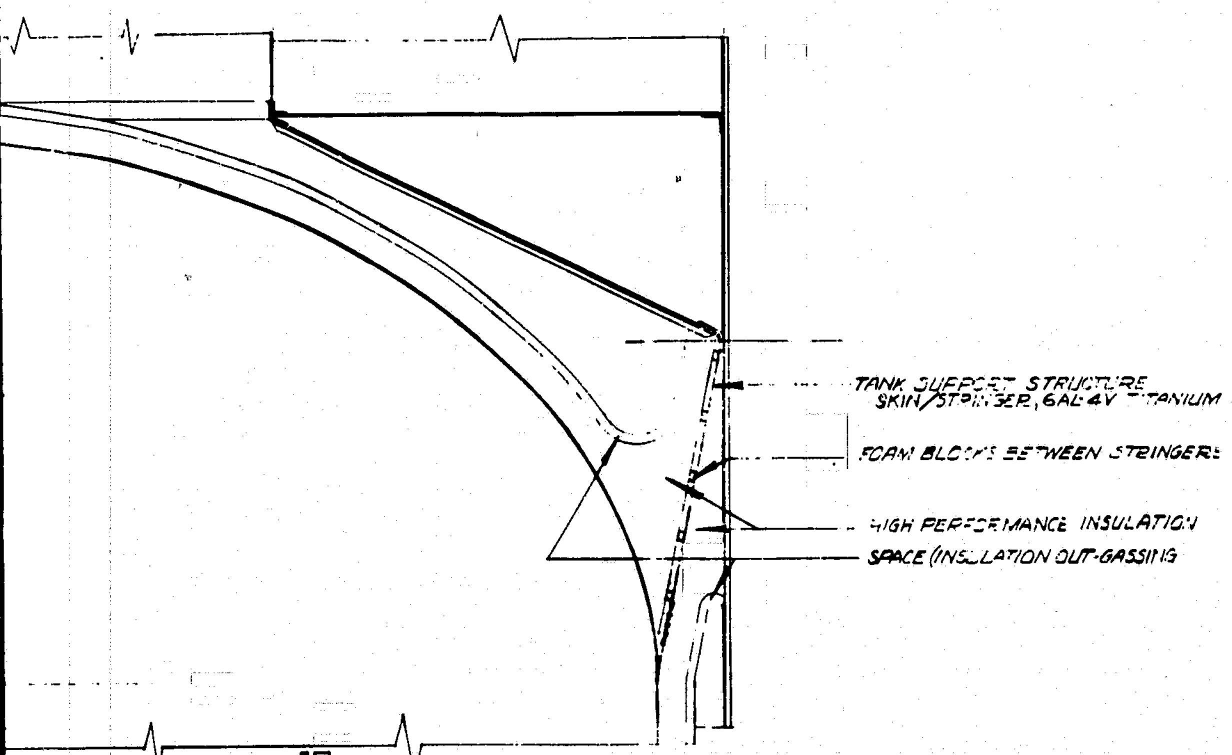
VIEW F

UPPER SUPPORT STRUCTURE & INSULATI

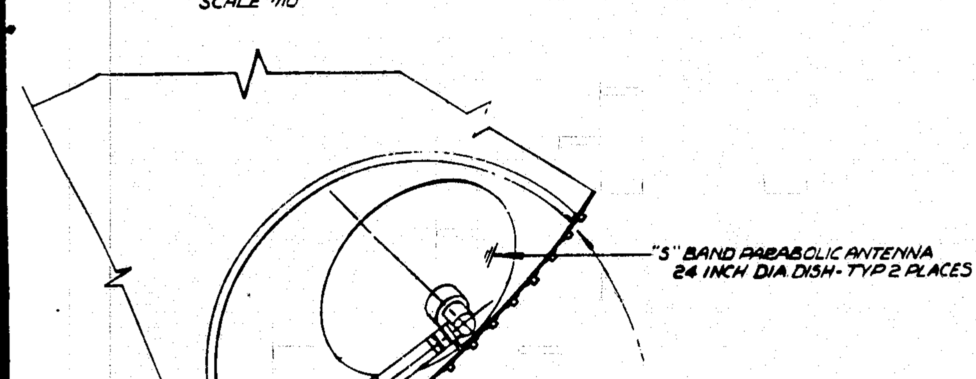
SCALE 1/10



"S" BAND RADAR-P



VIEW I
PER SUPPORT STRUCTURE & INSULATION - LH_2 TANK
SCALE 1/10



VIEW J
"S" BAND RADAR - PARABOLIC ANTENNA STOWED/DEPLOYED
TYPICAL 2 PLACES
SCALE 1/10

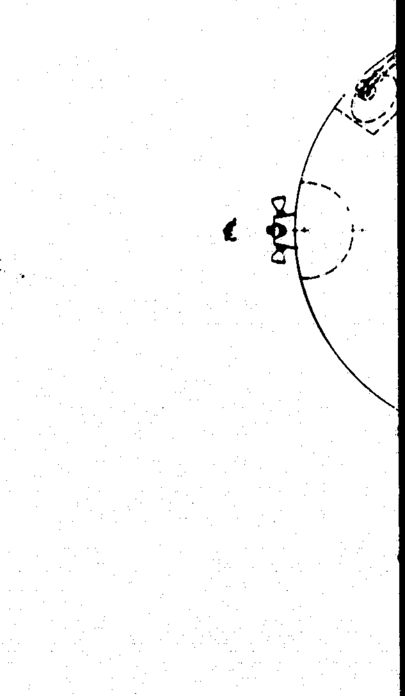


Figure 3-2. Concept 1, S

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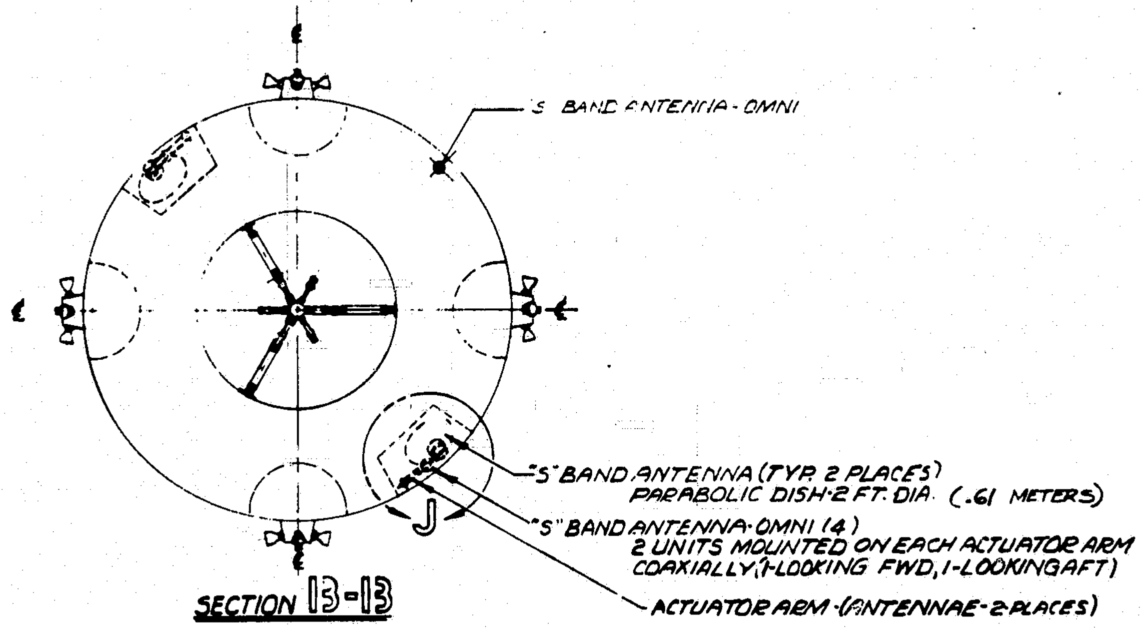

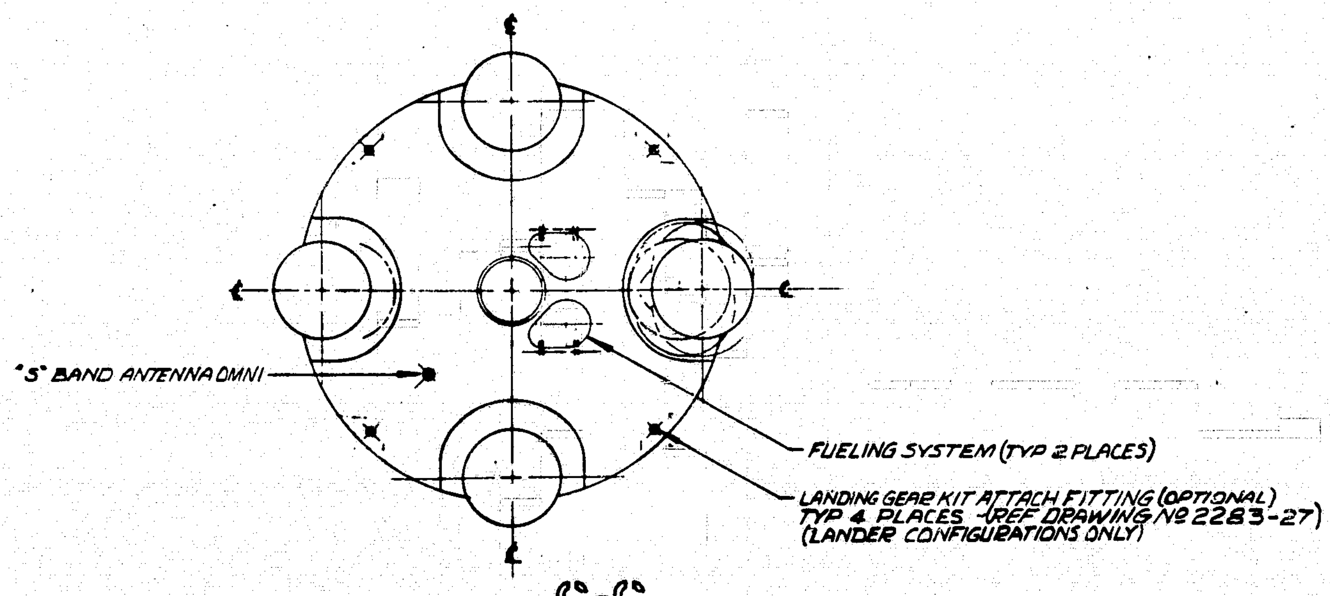

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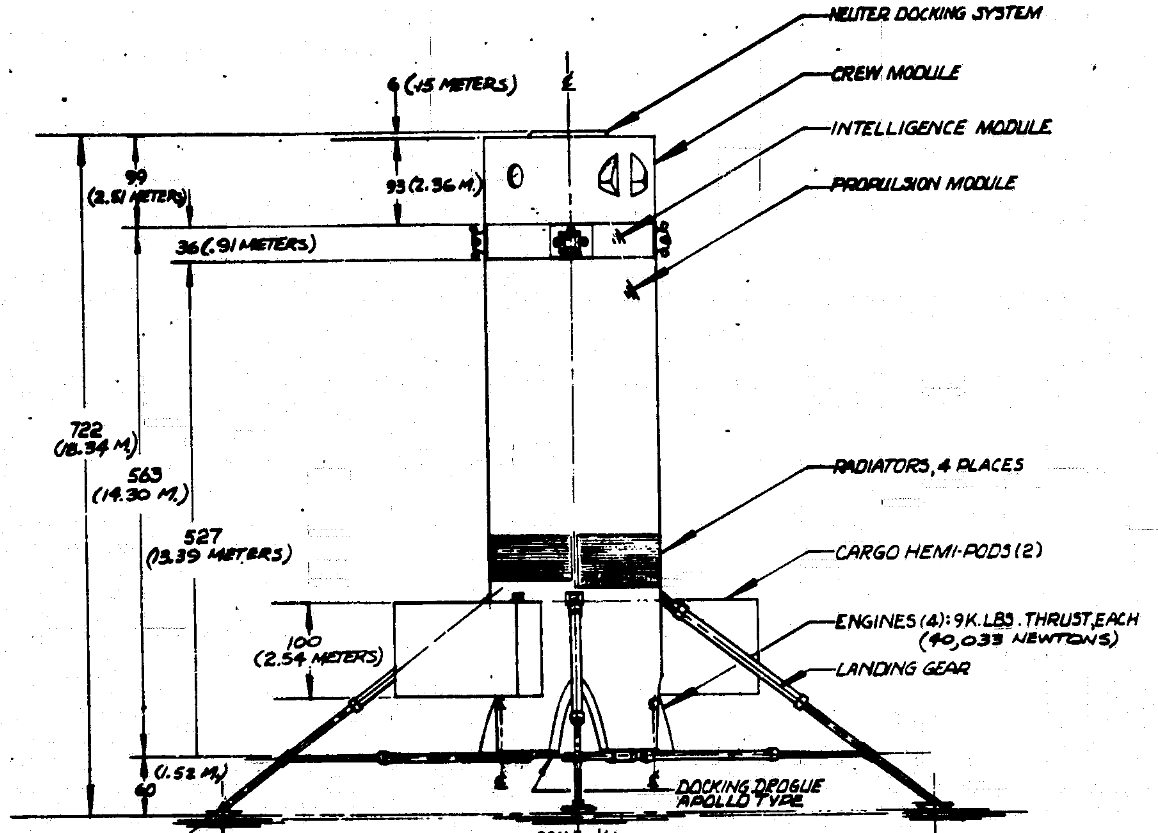
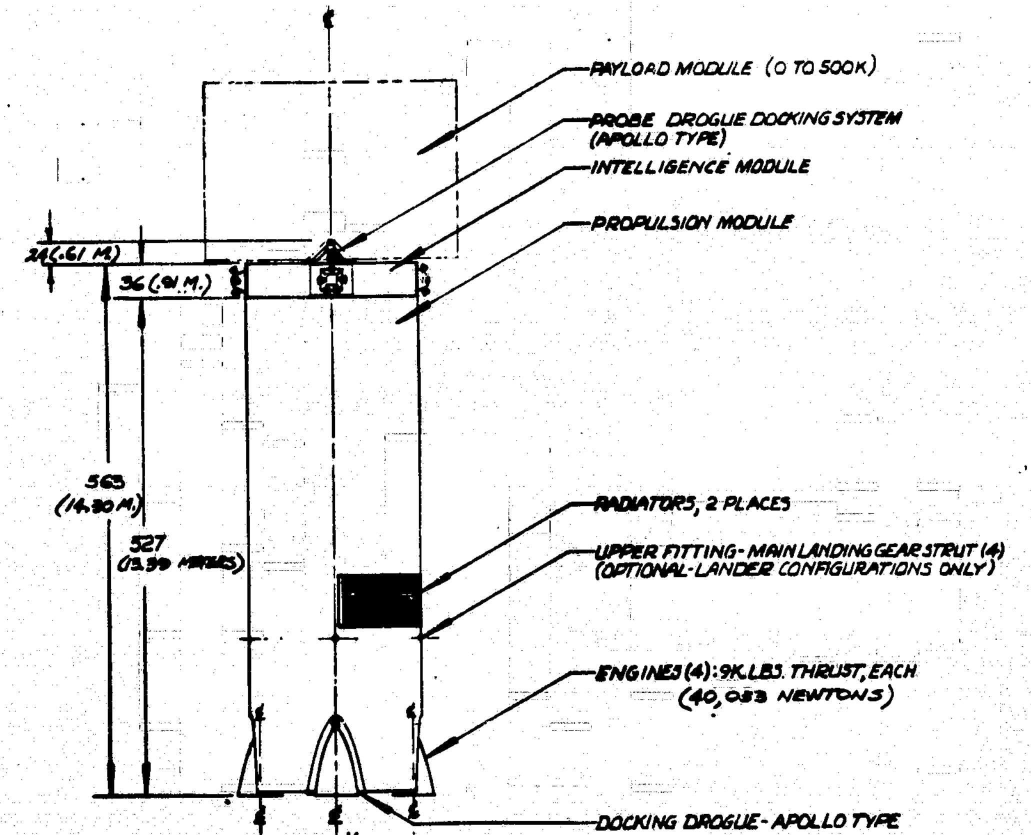
SHEET 2 OF 4

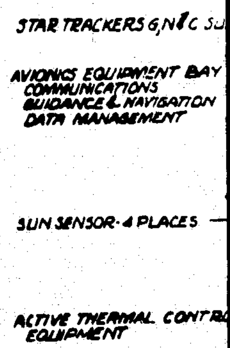
Figure 3-2. Concept 1, Space Tug, 80,000 Pounds Propellant Capacity



SCALE 1/100
LUNAR LANDER - MANNED CONFIGURATION



SCALE 1/100
EARTH ORBIT - UNMANNED CONFIGURATION



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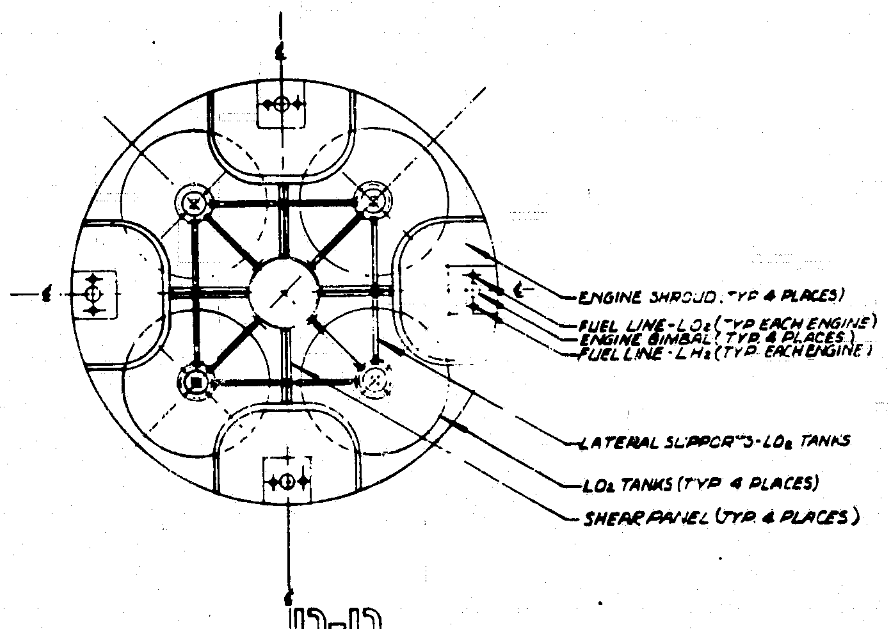
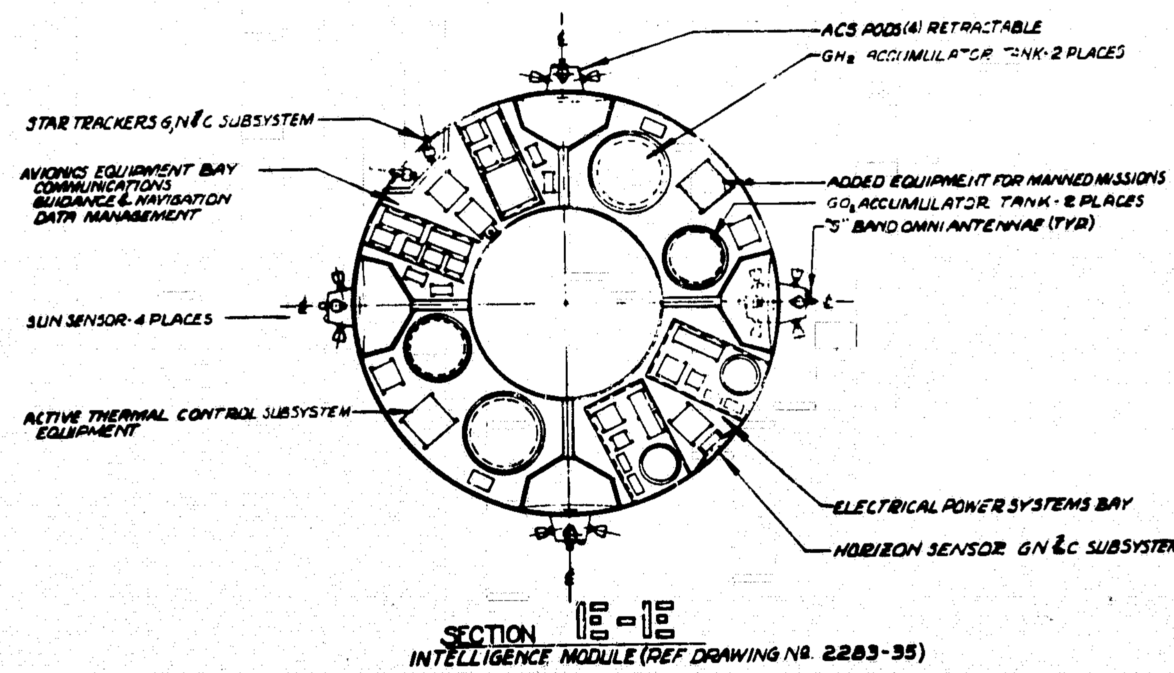
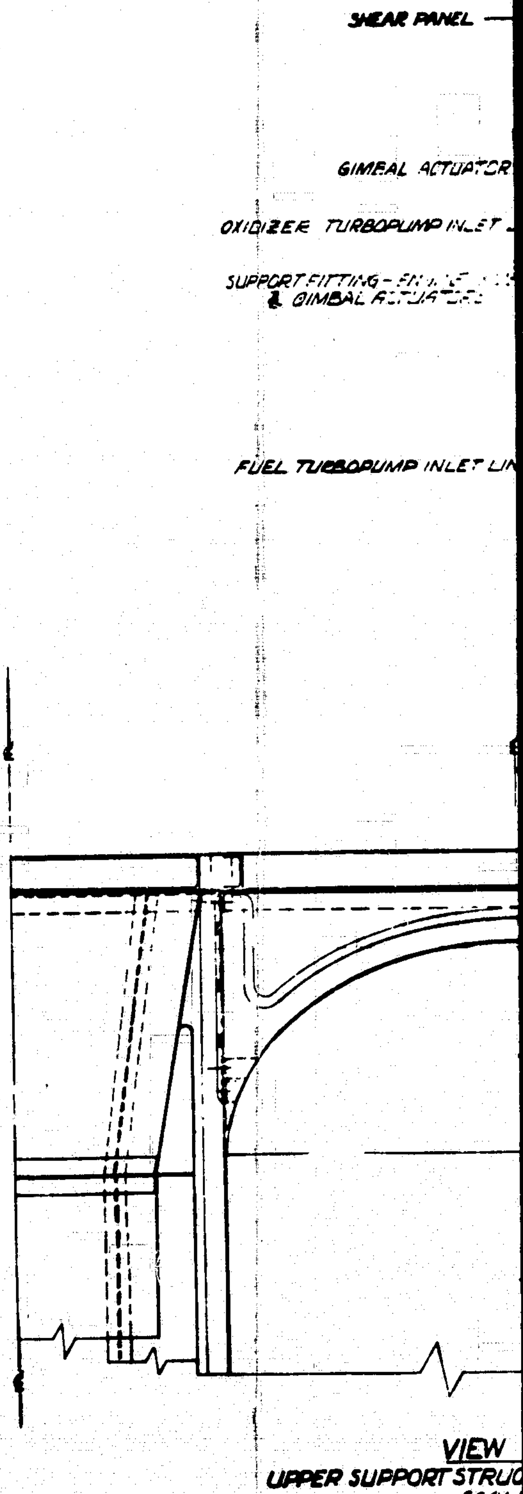


SECTION D-D
LATERAL SUPPORT STRUCTURE - LO_2 TANKS - ENGINE GIMBAL, FUEL LINES, & SHROUD ORIENTATION

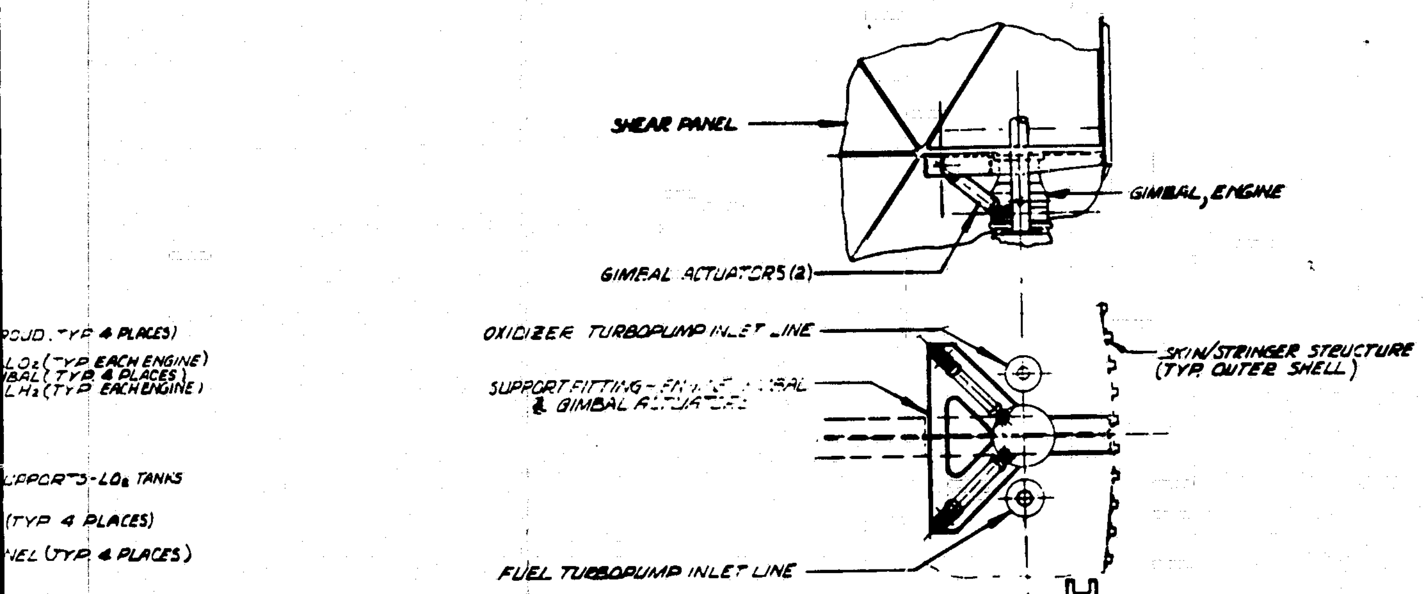


SECTION E-E
INTELLIGENCE MODULE (REF DRAWING NO. 2283-35)

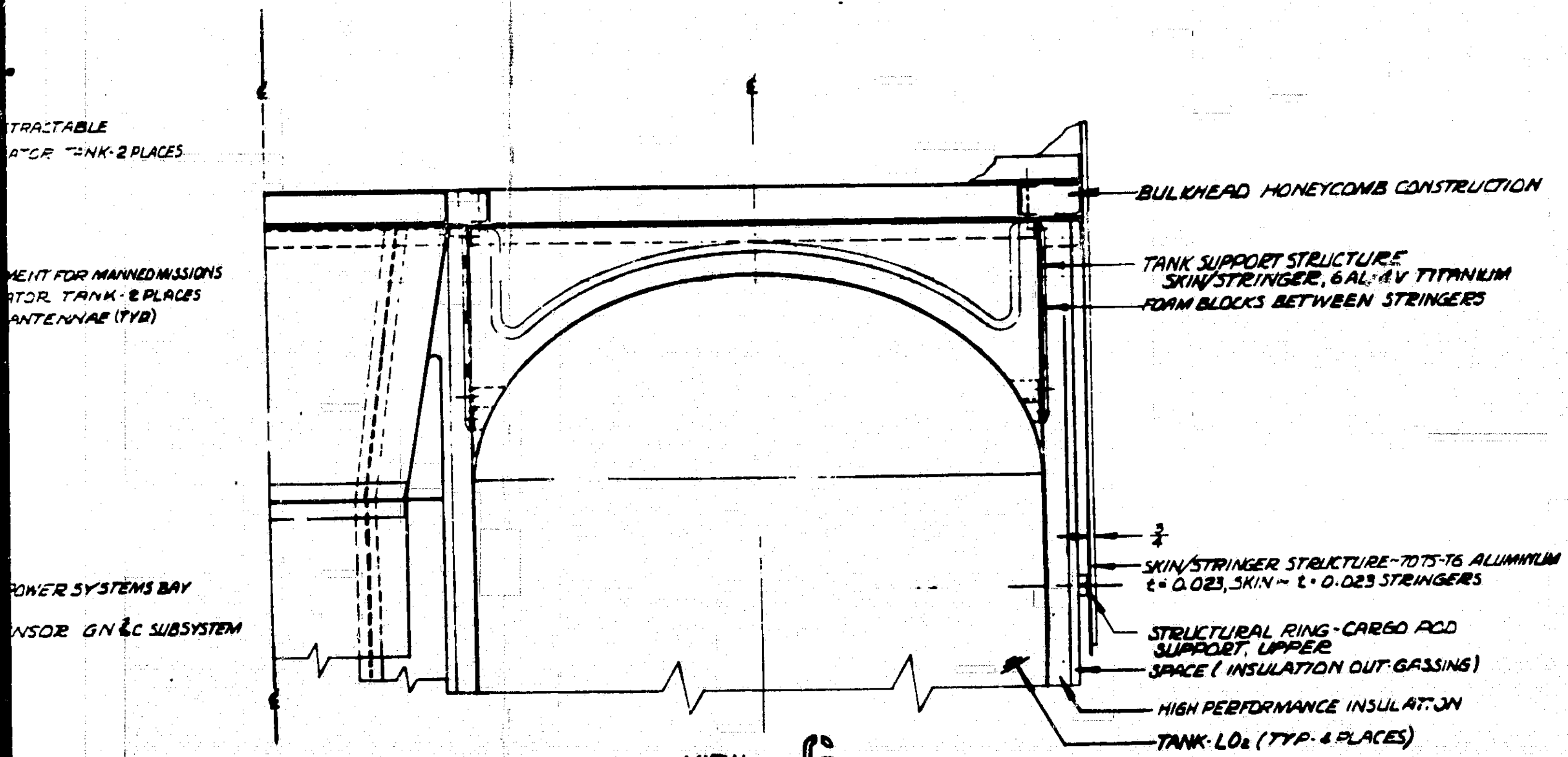


VIEW
UPPER SUPPORT STRUC
SCALE

Figure 3-2. Concept 1,



VIEW H
TYPICAL ENGINE SUPPORT STRUCTURE
& FUEL LINE LOCATIONS
(TYP. 4 PLACES)
SCALE 1/10



VIEW C2
UPPER SUPPORT STRUCTURE & INSULATION - LO2 TANKS
SCALE 1/10

2283-32A
SHEET 3 of 4

Figure 3-2. Concept 1, Space Tug, 80,000 Pounds Propellant Capacity

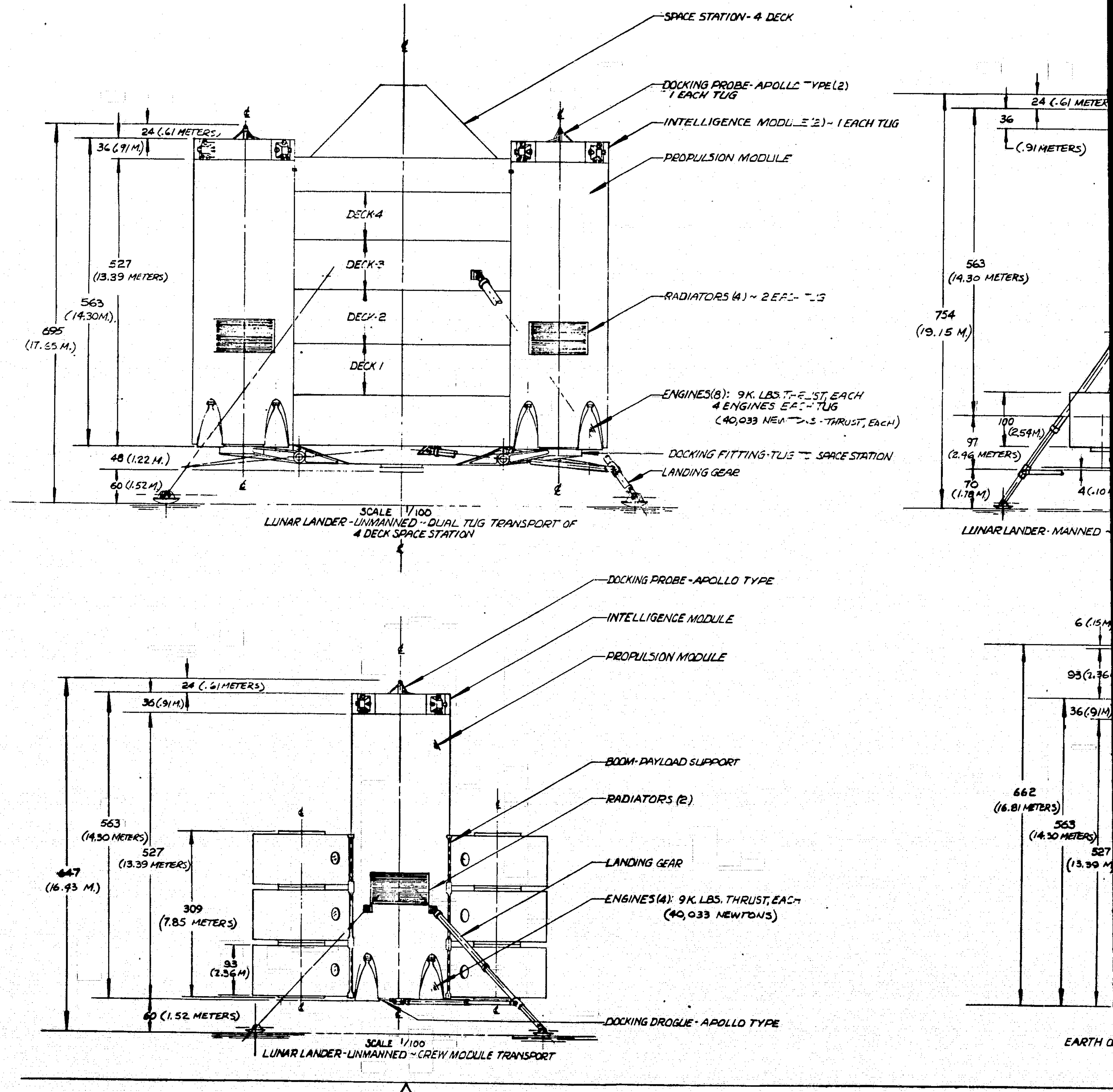


Figure 3-2. Concept 1, Sp

FOLDOUT FRAME

Space Division
North American Rockwell

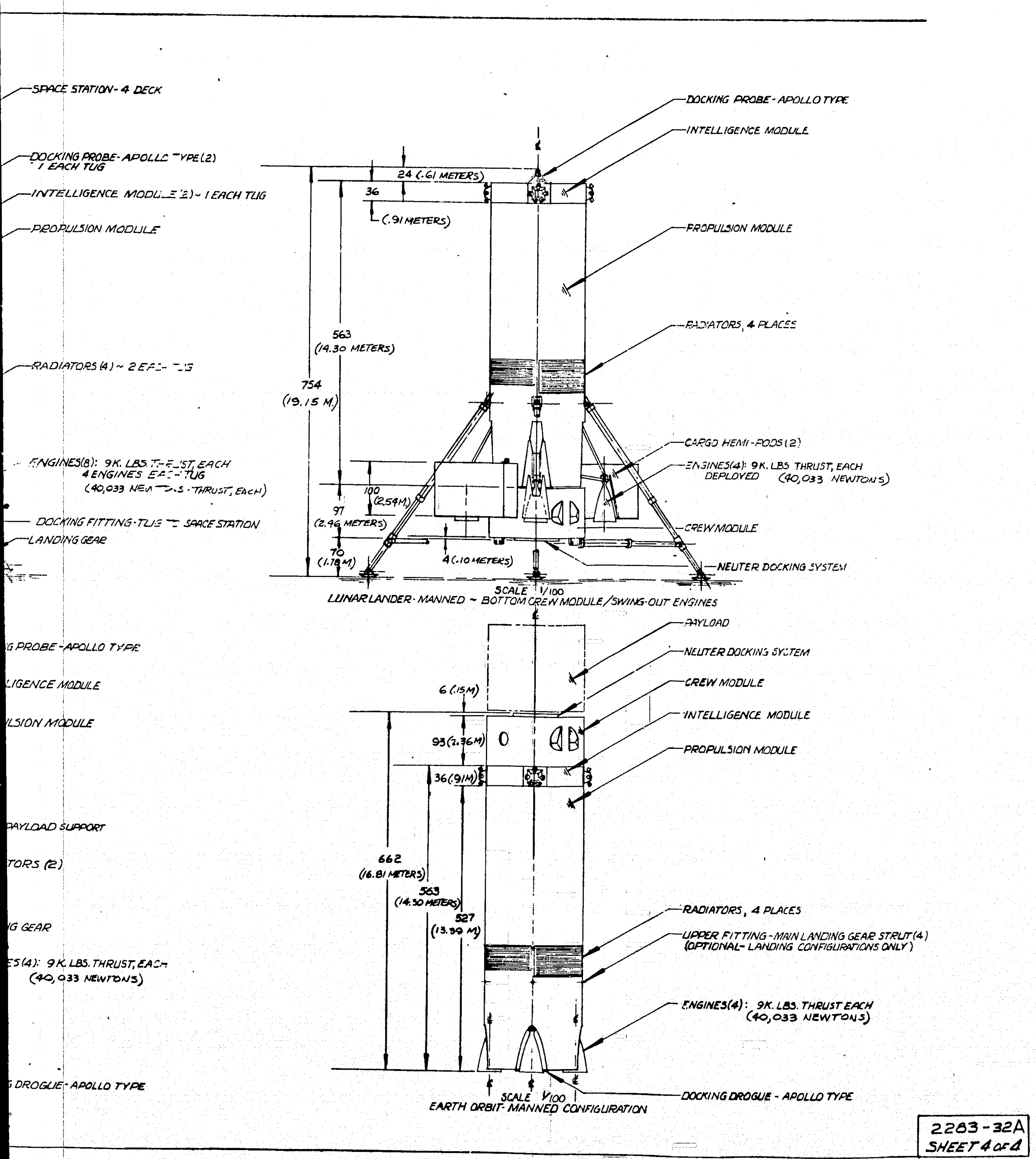


Figure 3-2. Concept 1, Space Tug, 80,000 Pounds Propellant Capacity

The overall vehicle dimensions are 180 inches (4.57 m) in diameter by 563 inches (14.3 m) long. The docking probe on the forward end extends an additional 14 inches (0.35 m) forward. The PM consists of a single LH_2 tank, four LOX tanks, and four main propulsion engines. The single LH_2 tank has an internal volume of 3300 cubic feet (93.45 m^3) with an inside diameter of 164 inches (4.16 m). The four LOX tanks have an internal volume of 275 cubic feet (7.8 m^3) each and are 62 inches (1.57 m) inside diameter. Both the LH_2 and LOX tanks are constructed of aluminum and are supported from titanium skin/stringer skirts. Boron epoxy construction would be used for the skirts if warranted by structural and thermal considerations.

Each of the tanks incorporate the use of a low-conductance radial sway brace at the end of the tank opposite the skirt. The sway braces can be constructed from boron epoxy to satisfy conductance requirements. Each of the five tanks is made up of a straight cylindrical center section with elliptical bulkheads. The elliptical bulkhead ratio is 1.4 to 1.

Each tank and skirt is fully insulated from the space environment by 1.5 inches (0.04 m) of high-performance insulation attached directly to the tanks. An atmospheric control barrier is placed between the insulation and the inside shell of the PM structure.

At the forward end of the vehicle, located on the vehicle centerline, is an Apollo-type active docking probe which is supported by a conical skin/stringer structure. The conical structure is an integral part of the IM; this aids in distributing the docking loads to the outside structure. The LH_2 tank support skirt is attached to a ring which forms a major load-carrying frame. The exterior or outer shell of the PM is a skin/stringer aluminum structure which constitutes the main load-carrying member between major frames.

The other major frame in the PM is the main load distribution frame which is stabilized by four radial beams. This frame is located between the LH_2 and LOX tanks and is used as the main support for the LOX tank skirts. The frame is of aluminum honeycomb shelf-type construction with stiffener webs. The shelf is stabilized and stiffened by the four radial beams. The beams are rib-stiffened type chem-milled aluminum structures. These beams support the passive docking cone at the aft end of the PM, and the main engine mounting structure. The LH_2 tank aft sway braces are attached to the outer rim of the LOX tank support frame. The sway braces for the LOX tanks are attached to the radial beams. The outer skin/stringer structure aft of the LOX tank support frame is again the main load distributing structure and stabilizing member for the radial beams.

The IM is attached to the forward end of the PM. The general configuration of the IM is 15 feet (4.57 m) in diameter and three feet (0.91 m) long. The module has four ACS pods with five engines on each pod. Two of the engines face aft, one forward, and one to each side. The pods are retractable within the IM to meet the EOS cargo bay 15-foot (4.57 m) diameter requirement. The communications and data management, guidance, and navigation, attitude control, environmental control and electrical power equipment is located within the IM. A total of 62 pounds (28 kg) of gaseous usable propellant for ACS and EPS is contained in four accumulator tanks in the IM.

The PM accommodates four main engines. Each engine has a 9,000-pound (40,034 newtons) thrust level at a chamber pressure of 1100 psia (758 newtons/square centimeter). The engine operates at a mixture ratio of 6 to 1 with a specific impulse of 463 seconds (4540 newton-seconds/kg). The nozzle area expansion ratio is 400 to 1. Each engine is mounted to a radial beam within a cavity at the aft end of the PM. The engines gimbal inboard to accommodate the 15-foot (4.57 m) diameter EOS cargo bay clearance requirement. Each engine nominally gimbals five degrees in two orthogonal directions for thrust vector alignment. View H of Figure 3-2 depicts the installation of the engine to the radial beam. Propellant refueling drogues and control valves are located on the aft bulkhead of the PM, as shown in Section C-C. The refueling drogues and valves are completely insulated from the space environment to minimize contamination and boiloff. The controls for each refueling system are separate which enables refueling to be performed sequentially and selectively, thus eliminating problems which might arise when transferring both LH_2 and LOX propellants. The transfer of propellants would only be performed when the tug vehicle is hard docked to the propellant supply vehicle.

The attachment of the payload to the PM is accomplished through use of the probe/drogue docking system in conjunction with compression pads. The adjustable compression pads are located on the forward end of the IM. When docked, the probe/drogue assembly is utilized as a tension tie. The compression pads are extended until they contact the base of the payload and then are adjusted to load the docking probe to a predetermined tension.

Two types of S-band antennas have been utilized for communications. Two 24-inch (0.61 m) diameter parabolic dish antennas are mounted on booms on opposite sides of the vehicle on the forward end of the propulsion module in the annular volume at the top of the LH_2 tank. View J of Figure 3-2 demonstrates the stowage of these dishes to meet the 15-foot (4.57 m) diameter cargo bay restriction. The dish antennas have the capability of rotating 360 degrees about their deployment boom and also hinge 100 degrees

6

on an axis perpendicular to the boom to give more than a hemisphere of coverage for each antenna. Ten omni-type S-band antennas are used on the tug vehicle. Two are mounted on each dish antenna boom, one facing forward and one facing aft. One each is located on each of the RCS pods. The last two are located on the forward and aft ends of the tug. Sun acquisition sensors are also located on each of the RCS pods. Docking equipment is located on the forward and aft bulkheads of the PM in the form of TV cameras and laser rendezvous and docking radar.

The environmental control system is made up of equipment within the IM and radiators on the external surface of the PM eighty square feet (7.43 m^2) of radiator area is required for the unmanned geosynchronous mission; however, sufficient area has been provided for a total of 148 square feet (13.75 m^2) which is the area required to perform manned earth-orbit missions. The radiators are located on the PM skin/stringer structural shell between the LH_2 and LOX tanks.

The tank insulation concept adopted for the space-based tug is shown in Views F and G (Figure 3-2). If a ground based approach were adopted, the major change would be the possible addition of a foam insulation between the tank wall and the HPI (Narsam) insulation on the LH_2 tank, ground purge gas bags and rents, and air repressurization equipment for re-entry.

The basic PM and IM which have been previously discussed, may be used in combinations with other modules and kits to perform a wide variety of missions. Six such missions are depicted in small scale drawings in Figure 3-2. The first of these missions utilizes the basic PM and IM to perform low earth-orbit unmanned transfer of various size payloads. The second approach incorporates the addition of a crew module forward of the IM to execute manned low earth-orbit missions. Each of these concepts makes available 80,000 pounds (36,287 kg) of usable propellant. The unmanned configuration is 563 inches (14.3 m) long and the manned is 668 inches (16.9 m) long.

The remaining four configurations are lunar landers. Each one utilizes a landing gear kit and a space radiator kit which radiates toward space when the tug is on the lunar surface. The first configuration shown is an unmanned lander. Several cargo modules which have been configured as CM shells are shown attached to the side of the tug. The basic vehicle is 563 inches (14.3 m) long. The next lander is manned and is identical to the first with the addition of a crew module (CM) on the forward end. The cargo modules (CAM's) shown are hemipods which, when put together, fit within the EOS cargo bay 15-foot (4.57 m) diameter envelope. This concept is 758 inches (19.3 m) long.

The next configuration is a manned lander with a different PM than the basic module. The concept utilizes a bottom mounted CM and a PM with swing-out engines. This configuration is made up (aft to forward) of a CM, a PM with swing-out engines, and an IM. The radiators have been mounted on the forward end of the IM. Hemipod CAM's have been incorporated near the aft end of the vehicle. The difference in length from the baseline PM is an addition of 30 inches (0.75 m). The vehicle is 668 inches (16.9 meters) long overall.

The last lunar landing configuration features two propulsion stages operating in a parallel mode to land a centrally located four-deck space station core module. The stages are made up of a PM and a forward attachable IM. The landing gear is attached to the space station core module. The vehicle configuration is 563 inches (14.3 m) long.

All of the lunar lander configurations make 80,000 pounds (36,287 kg) of propellant available to the main engines even though a lesser amount is required to complete the landing mode A mission.

3.2.3 CONCEPT 5 (FIGURE 3-3)

Concept 5 is an unmanned two-stage geosynchronous mission-optimized vehicle. Each stage has the capacity of 41.2 K pounds of usable propellant for the main engines, 500 pounds for the attitude control engines, and 150 pounds for the electrical power systems. The propellant tank volumes and sizes are presented here.

The single LH_2 tank has an internal volume of 1.620 cubic feet (45.7m^3) an inside diameter of 164 inches (4.16 m), and an overall inside of length of 117 inches (3.48 m). The liquid oxygen tanks (four per stage) have an internal volume of 137 cubic feet (3.87 m^3), an inside diameter of 60 inches (1.53 m), and an internal length of 90 inches (2.28 m). All of these tanks are made of aluminum and are suspended from titanium skirts with non-thermal conducting sway bracking at the opposite ends. The general tank shape is a straight cylindrical center section with elliptical end heads. The end head ratio is 1.4 to 1. All propellant tanks and suspension skirts are fully insulated from the space environment with high-performance insulation, a vent space, and combination meteoroid barrier, vent shroud, or purging bag is ground-based. This insulation system is known as NARSAM. An optical material for the tank support skirt is glass or boron epoxy which will probably be used for the sway braces.

As with Concept 1, the general configuration of the IM is 15 feet (4.57 m) in diameter by 3 feet (.915 m) long, with the four attitude control pods having five nozzles on each pod - two face aft, one faces up, and one faces to each side.

The ACS pods are retractable into the IM to meet the 15-foot (4.57 m) diameter clearance requirements. The avionics systems located in the IM section are the communications, guidance and navigation, and data management equipment. The attitude control system, including tanks, is located in the IM, along with the electrical power system and the environmental control system (except for the radiators which are located on the outside of the vehicle).

At the forward end of each stage located on the center line is an Apollo-type active docking probe, supported by a conical skin-stringer structure. This structure is an integral part of the IM which aids in distributing the docking loads to the outside structure. This docking cone attaches to the same structural ring as the liquid hydrogen tank skirt, forming one of the major load carrying frames. The exterior of the PM is skin-stringer aluminum type of structure, which is the main axial load carrying member between the two major frames.

The second major frame is the main load distribution frame stabilized by four radial beams, and is used as the main vehicle support structure. This frame is located between the liquid hydrogen and the liquid oxygen tanks and is the main support of the liquid oxygen tanks by means of conical skirts. The liquid hydrogen tank lower radial sway braces are attached to the outer rim of LOX tank support frame. The axial radial beams are rib-stiffened type aluminum structure. They are the main support for the passive docking cone, and the four main thrust engines. The LOX tank lower sway bracing is attached to these radial beams for stabilization. The outer skin stringer structure aft of the second major frame becomes a secondary structure and closeout member for the aft end of the vehicle.

Each propulsion stage has four 9800-pound (43500 N) thrust engines with 1100-psi (757 n/cm^2) chamber pressure, 400 expansion ratio, specific impulse of 463 seconds (4530 n sec/kg) and a 6 to 1 mixture ratio. These engines are mounted in cavities in the aft end of the propulsion stage and gimbal inboard for stowage within the 15-foot (5.57 m) diameter requirement.

The aft passive docking cone is located on the center line of the PM, and is similar to that on the Apollo vehicle.

Located on the aft closeout bulkhead are the refueling drogues and their control valves. The refueling equipment is fully insulated and enclosed to prevent contamination and reduce boiloff losses. The controls for each drogue are separate to enable each drogue to be operated sequentially and prevent any potential problems between the LH_2 and the LOX propellants. Fuel will be transferred only during a hard dock condition, when the Tug is docked to the propellant depot.

On the aft conical structure of the PM, four adjustable interstage stabilizing trusses are attached, and are individually activated by the docking probe. The stabilizing trusses are extended until they contact the aft propulsion stage and load the docking probe on the second stage to a predetermined tension load.

The communication system has two types of antennas. Two 24-inch (.61 m) diameter S-band dish antennas, retractable for storage during launch in the EOS, and ten S-band omni-type antennas located on the ACS pads, the dish antenna booms, and one on each end of the propulsion stage.

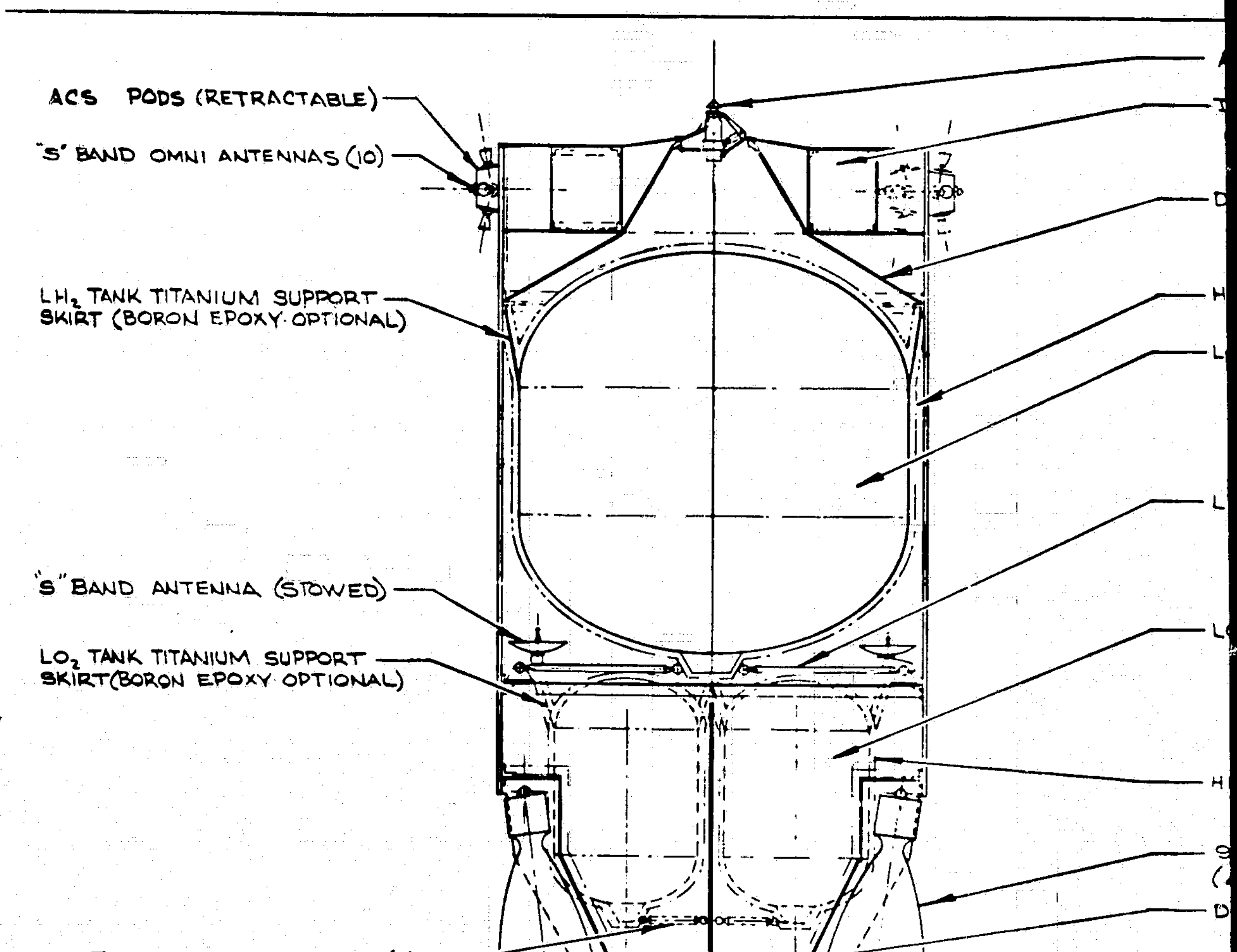
The environmental control system consists of externally located radiators in addition to the internal equipment. Eighty square feet of radiator is required for geosynchronous missions, and 148 square feet (13.75 m^2) required for manned missions. Three radiators are located between the main frames on the cylindrical section, externally on the skin stringer structure.

The basic PM's can be used for various missions as shown on Sheet 4 of Figure 3-3. These missions include manned, and unmanned earth orbiting vehicles, geosynchronous vehicles, and manned and unmanned lunar landing vehicles handling a wide variety of payload modules.

3.2.4 CONCEPT 11 (DRAWING 2283-34)

The concept shown on Figure 3-4 is that of the stage and a half. This concept consists of a small recoverable propulsion module and an expended tank set. The primary mission for which the concept is sized is an unmanned geosynchronous mission to deliver 10,000 pounds (4536 kg). The tank set is sized to deliver 53,000 pounds (24,040 kg) of usable propellant to the main engines. It also contains 500 pounds (227 kg) of usable propellant for the attitude control (ACS) and electrical power (EPS) subsystems. The PM delivers 11,000 pounds (4898 kg) of usable propellant to the main engines and contains 100 pounds (45 kg) for the ACS and EPS subsystems.

As shown, the geosynchronous configuration is 597 inches (49.75 feet) (15.16 meters) long from the forward end of the tank set to the aft end of the PM. The tank set is 377 inches (9.57 meters) long plus 14 inches (0.35 meters) for the docking probe at the forward end. A single LH_2 tank and four LOX tanks are within the tank set. The single liquid hydrogen tank has an internal volume of 2020 cubic feet (57.2 cubic meters) is 164 inches (4.16 meters) inside diameter and 203 inches (5.16 meters) long inside. The four LOX tanks have an internal volume of 105 cubic feet (2.66 cubic meters)



DUAL STAGE GEOSYNCHRONOUS SPACE TUG

(41,210 LBS USABLE PROPELLANT / STAGE)
(18,660 KILOGRAMS)

SECTION A - A

FOLDOUT FRAME 1

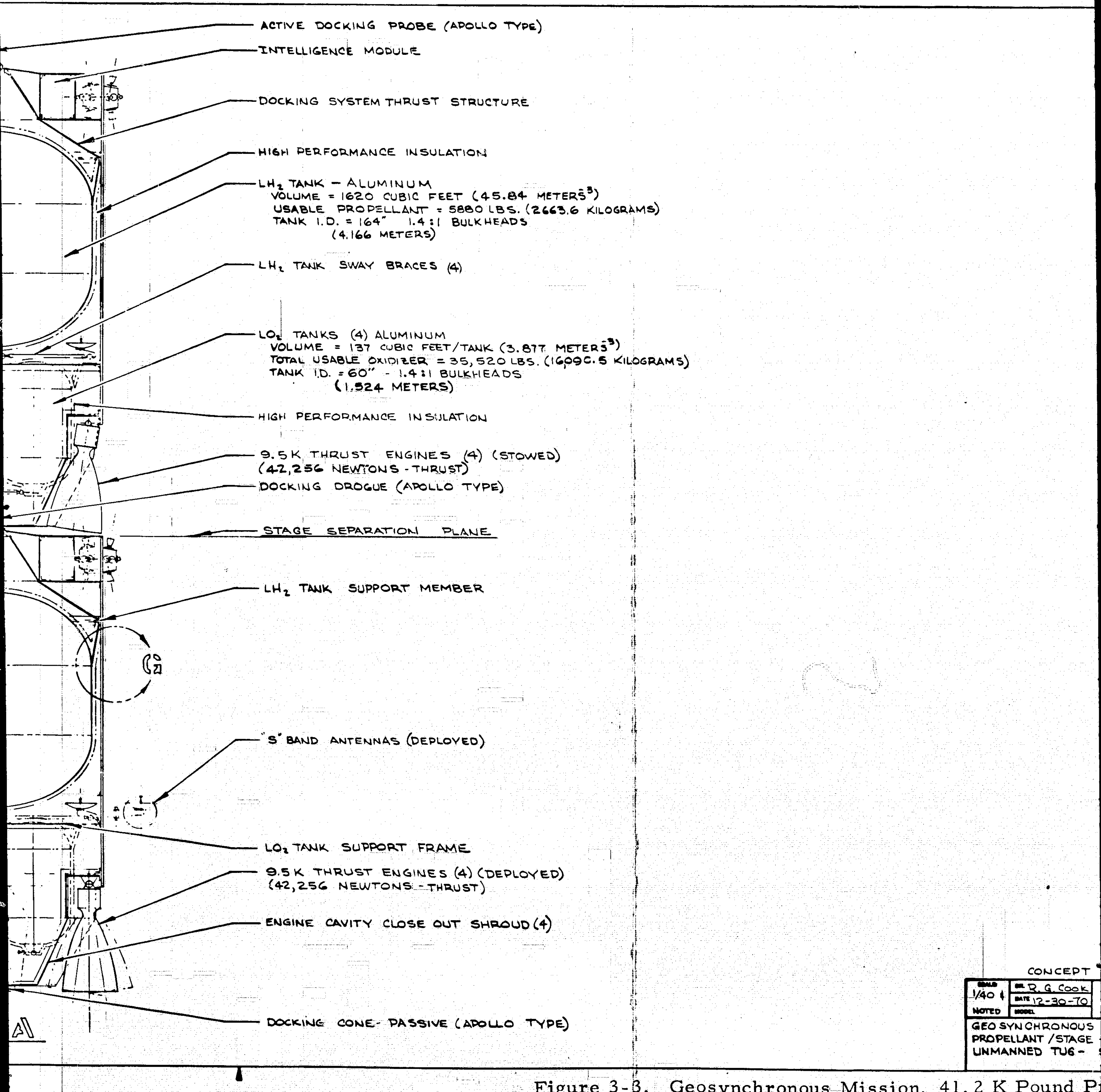


Figure 3-3. Geosynchronous Mission, 41.2 K Pound P
Recoverable Unmanned Tug

FOLDOUT FRAME 2

FOLDOUT FRA

Space Division
North American Rockwell

DBE (APOLLO TYPE)

E

UST STRUCTURE

ISULATION

JM
C FEET (45.84 METERS^3)
NT = 5880 LBS. (2663.6 KILOGRAMS)
.4 : 1 BULKHEADS
RS)

CES (4)

NUM
FEET/TANK (3.877 METERS^3)
ER = 35,520 LBS. (16090.5 KILOGRAMS)
: 1 BULKHEADS
TERS)

I SULATION

NES (4) (STOWED)
THRUST)
POLLO TYPE)

PLANE

MEMBER

EPLOYED)

FRAME

NES (4) (DEPLOYED)
THRUST)

E OUT SHROUD (4)

SIVE (APOLLO TYPE)

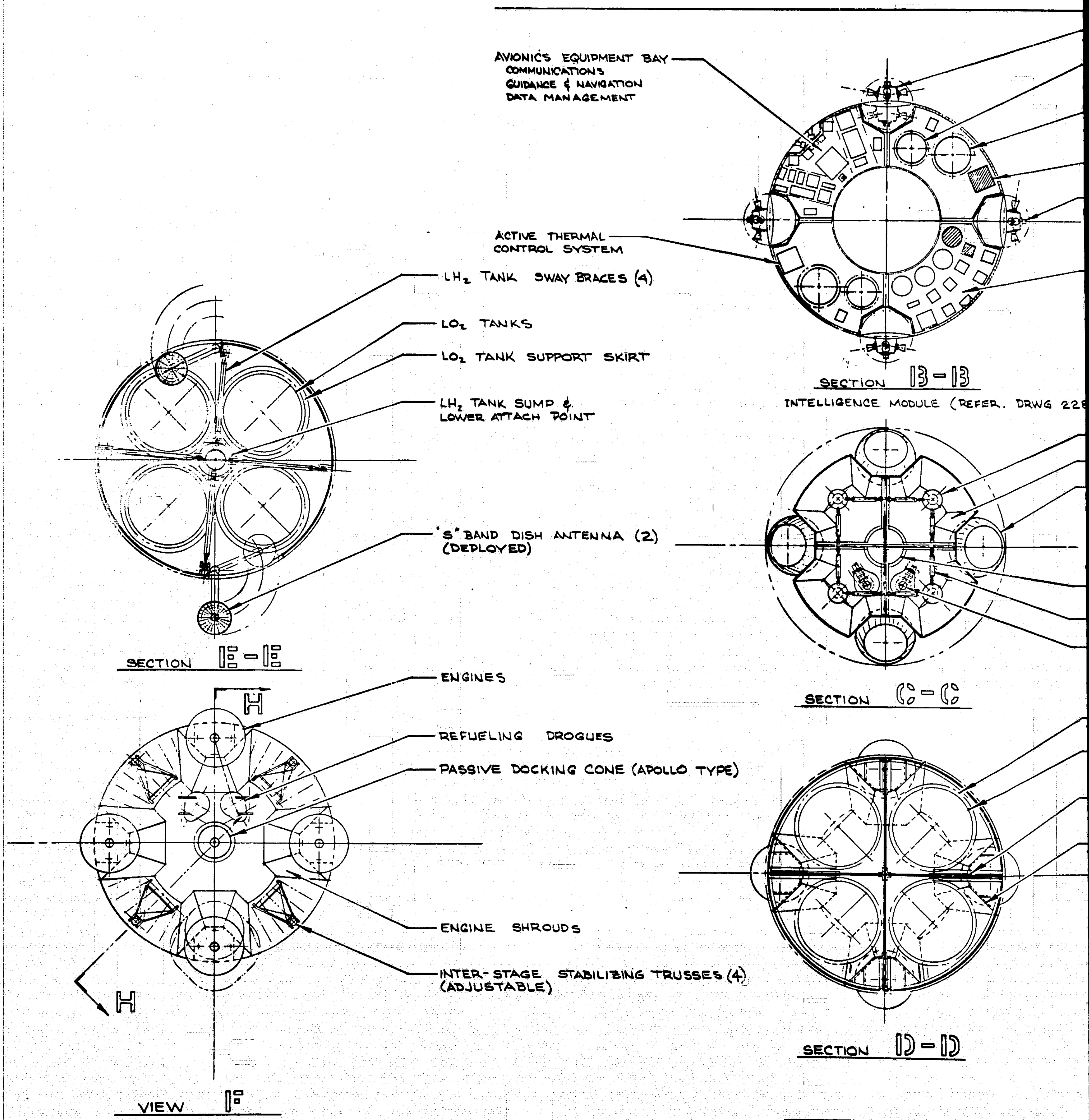
THIS PRINT REDUCED 50%

CONCEPT # 5

SCALE 1/40 NOTED	DR. R. G. COOK DATE 12-30-70 MODEL	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
GEO SYNCHRONOUS MISSION - 41.2 K LBS PROPELLANT/STAGE - TWO STAGE RECOVERABLE UNMANNED TUG - SPACE TUG STUDY -			2283-33B SHEET # 1 OF 4

Figure 3-3. Geosynchronous Mission, 41.2 K Pound Propellant Stage, Two-Stage Recoverable Unmanned Tug

1 OF 4



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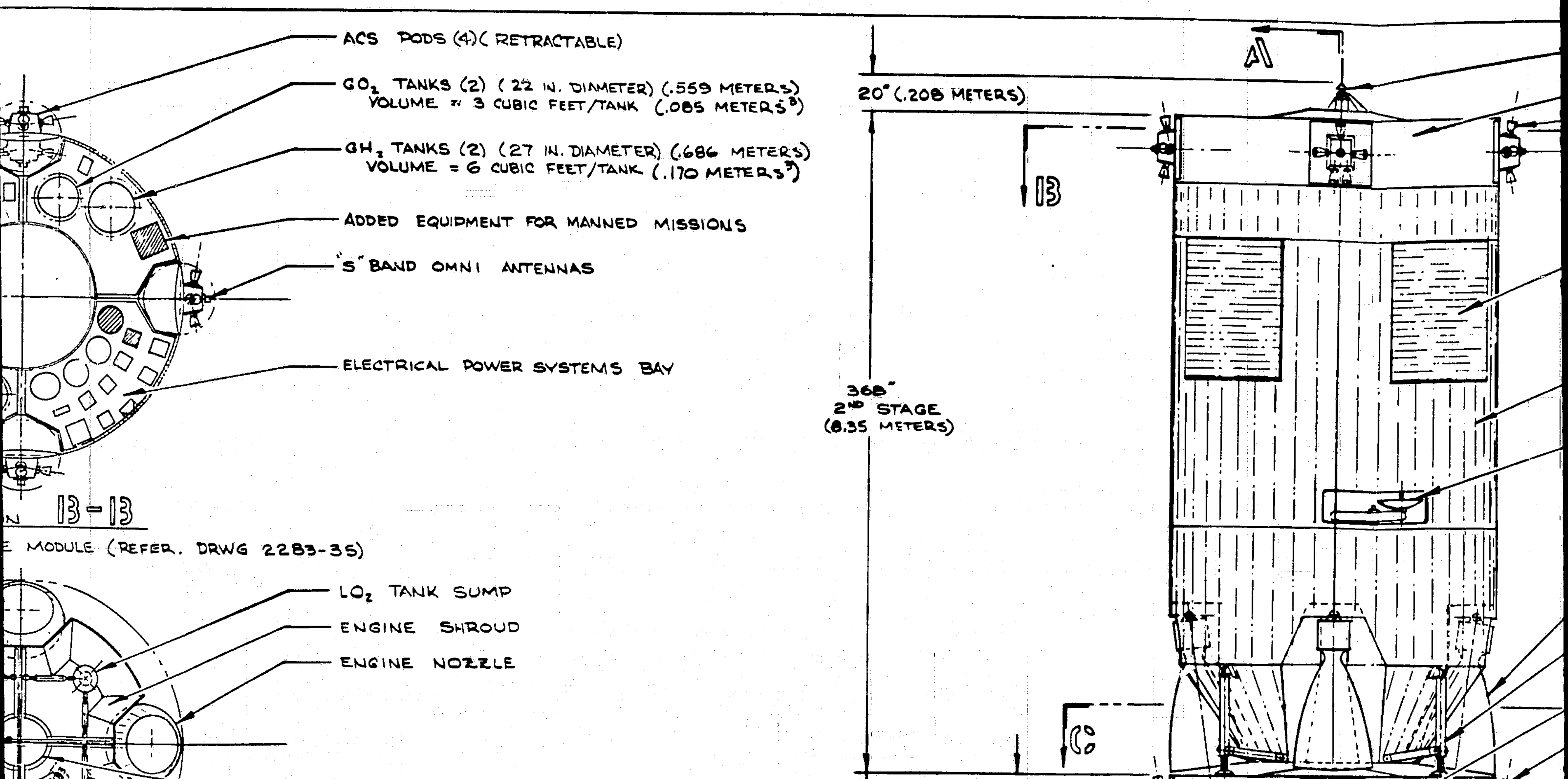


Figure 3-3. Geosynchr
Recoverab

FOLDOUT FRAME 2

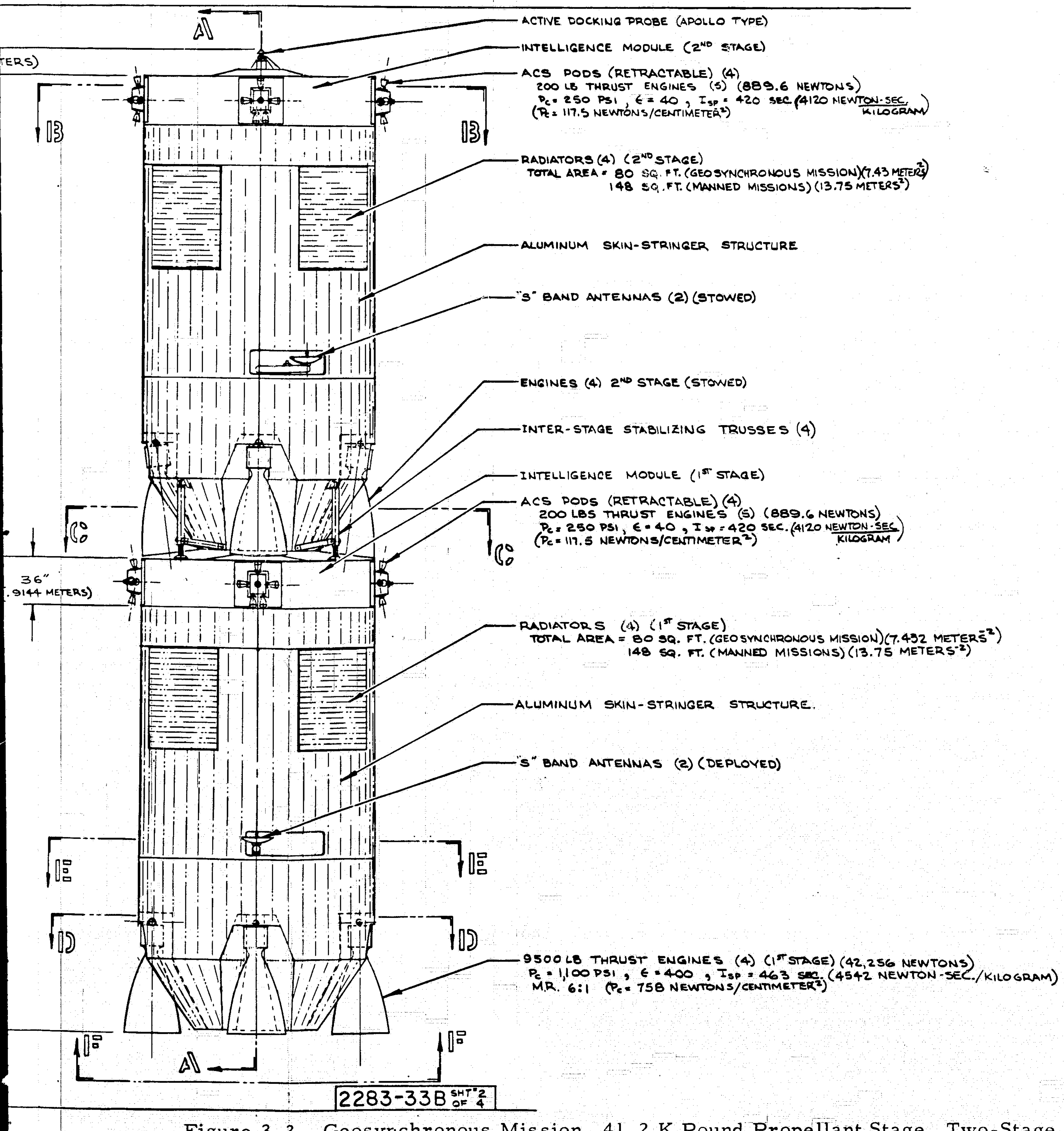
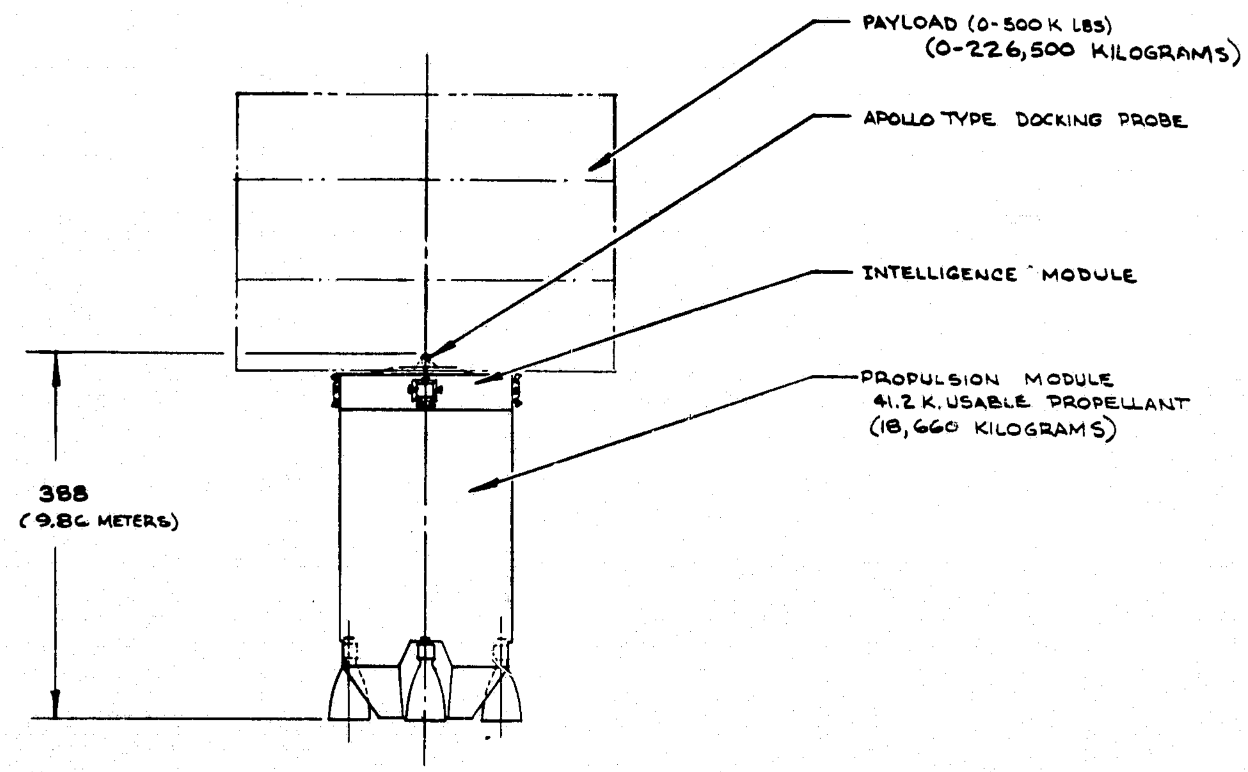
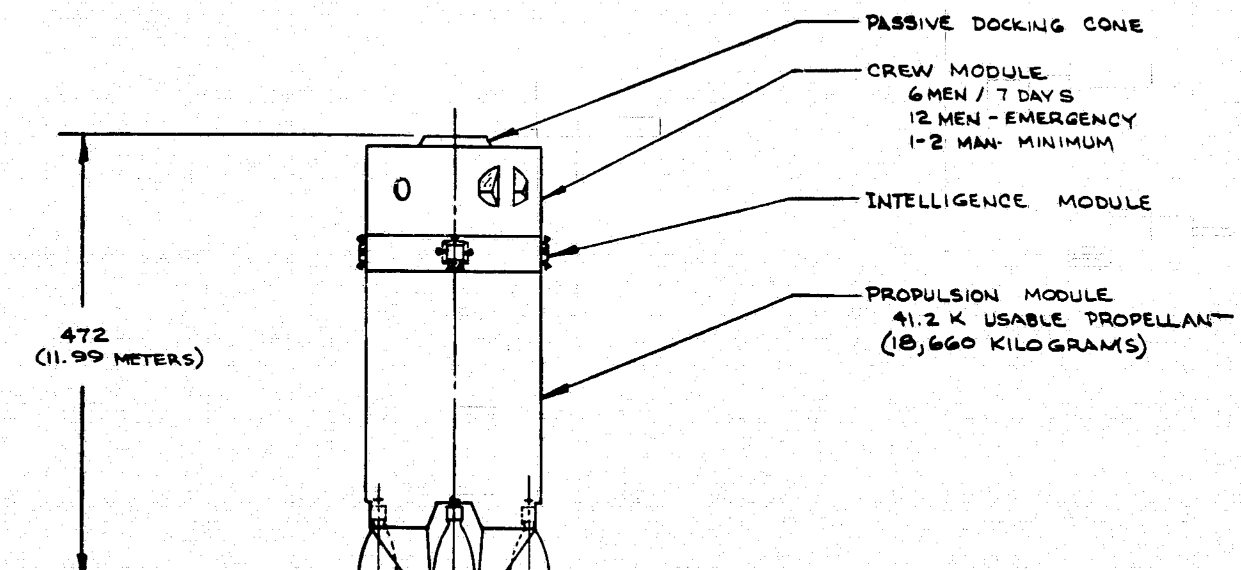


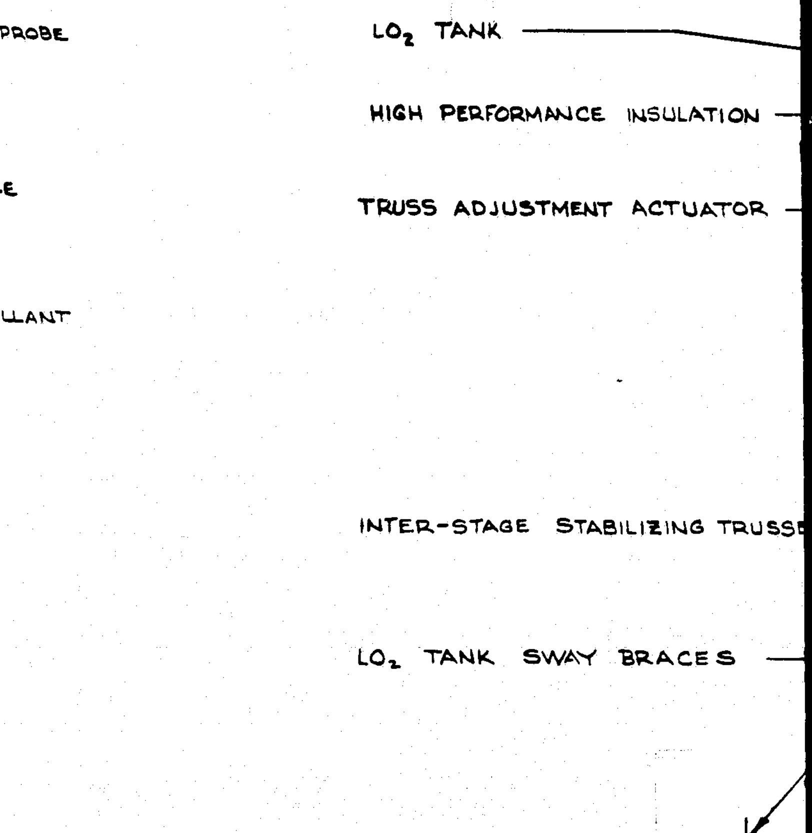
Figure 3-3. Geosynchronous Mission, 41.2 K Pound Propellant Stage, Two-Stage Recoverable Unmanned Tug



UNMANNED EARTH ORBITING TUG



MANNED EARTH ORBITING TUG



MANNED LUNAR L

(BOTTOM MOUNTED C

FOLDOUT.

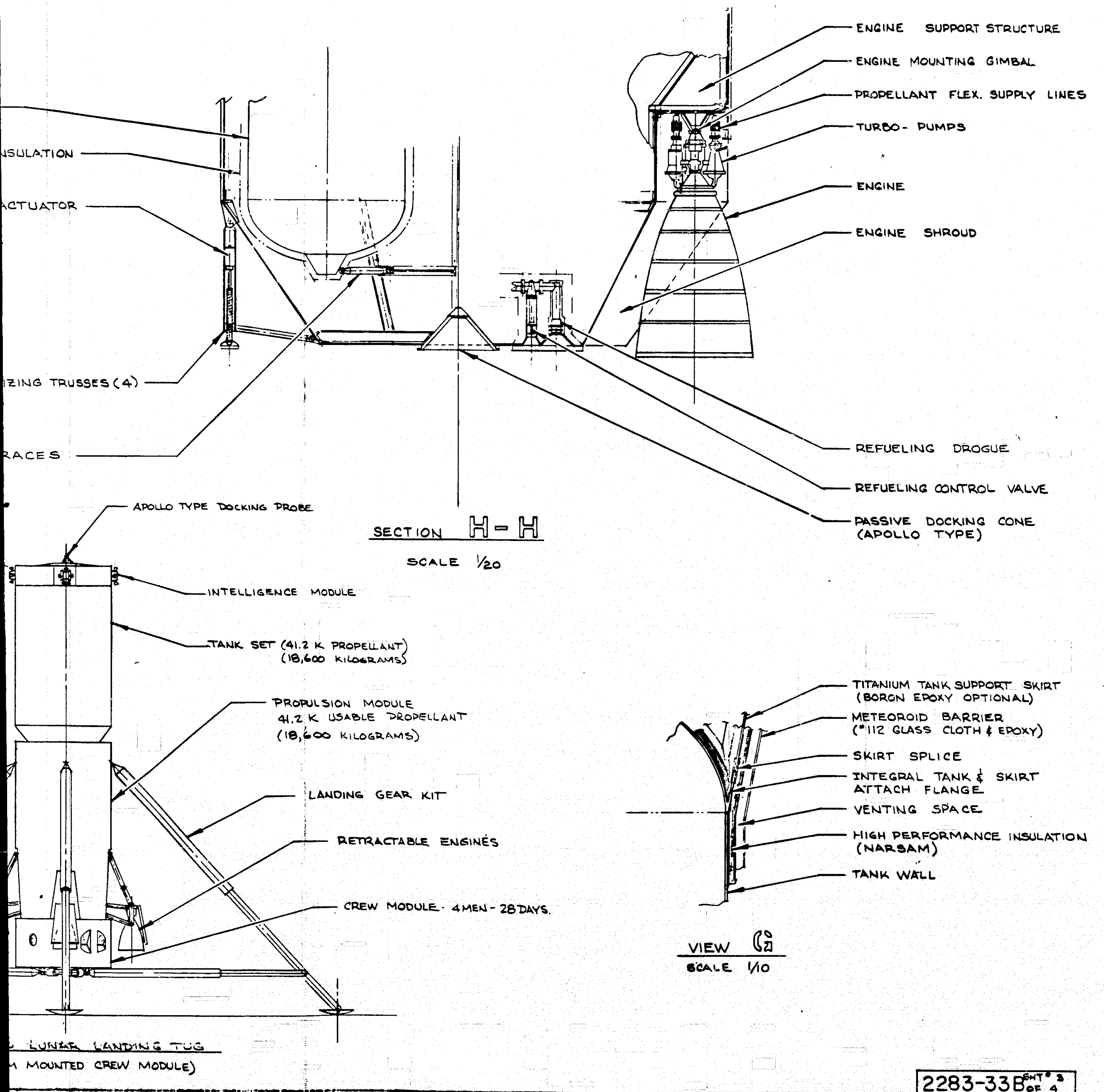
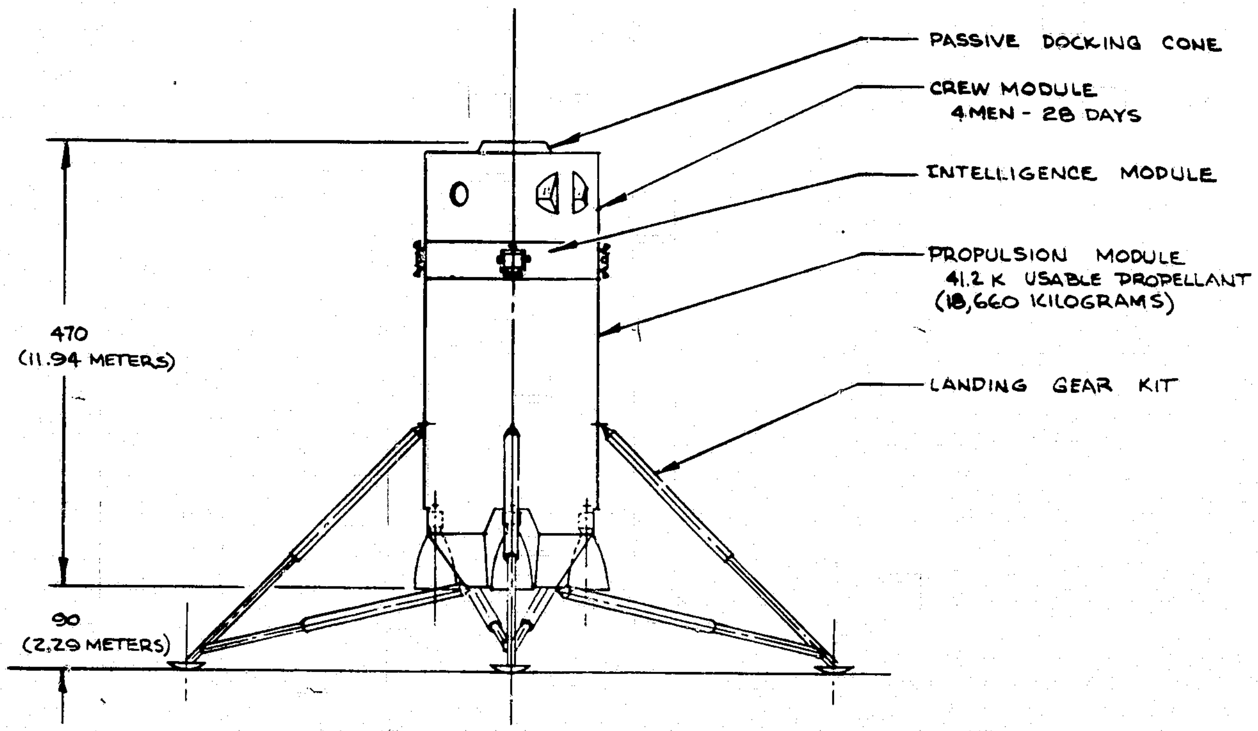
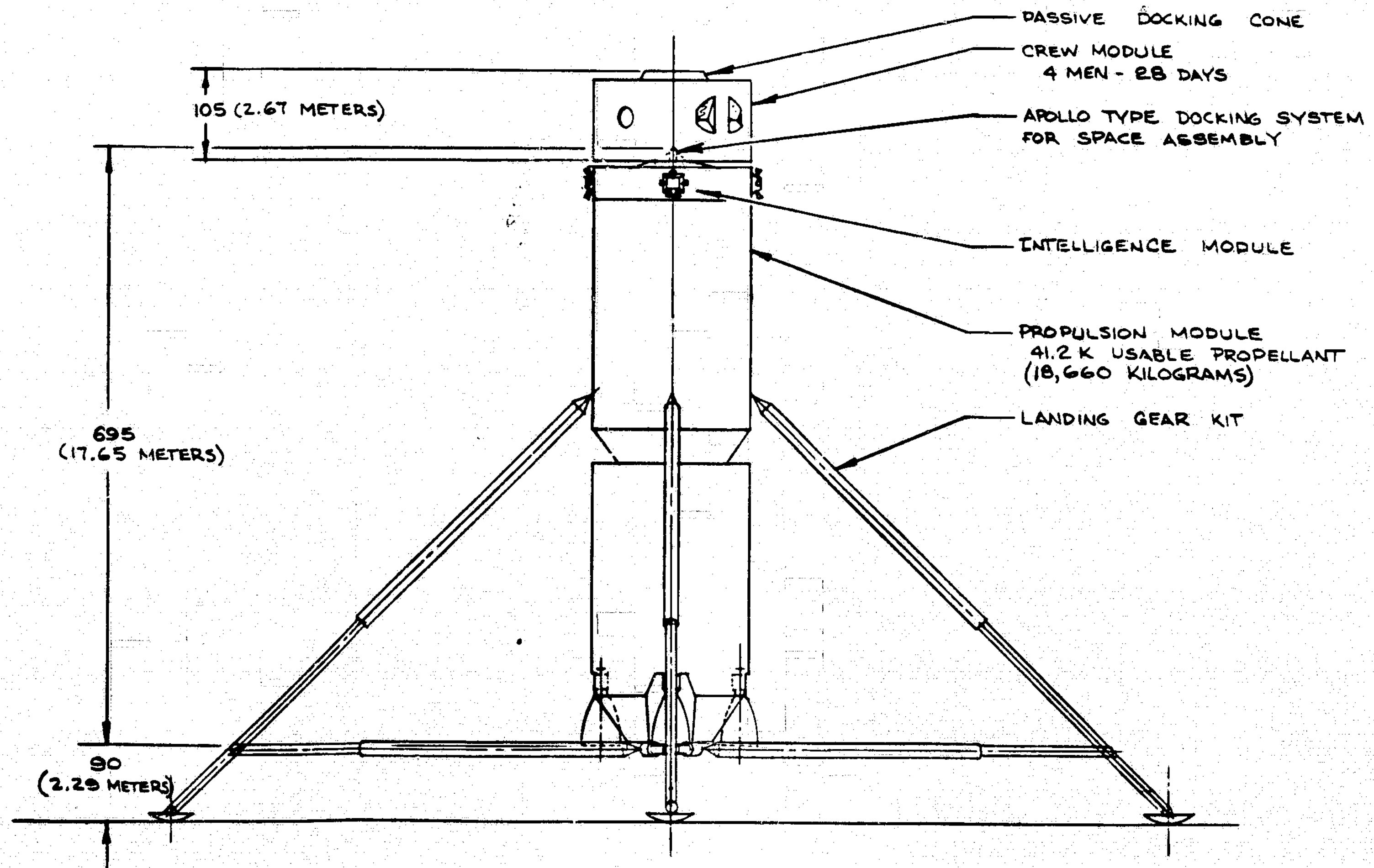


Figure 3-3. Geosynchronous Mission, 41.2 K Pound Propellant Stage, Two-Stage Recoverable Unmanned Tug

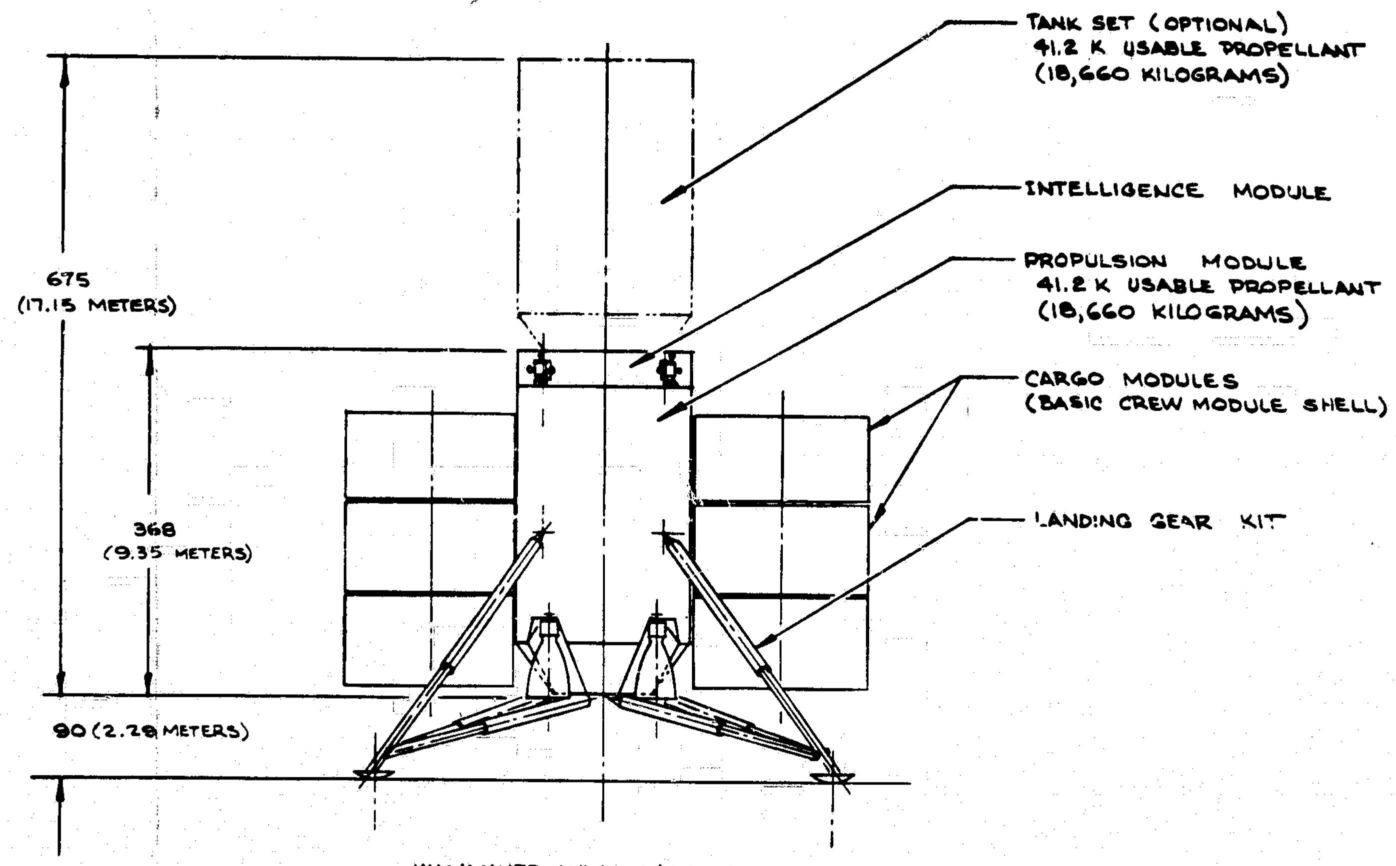


MANNED LUNAR LANDING TUG
(LIMITED LIFT-OFF CAPABILITY)

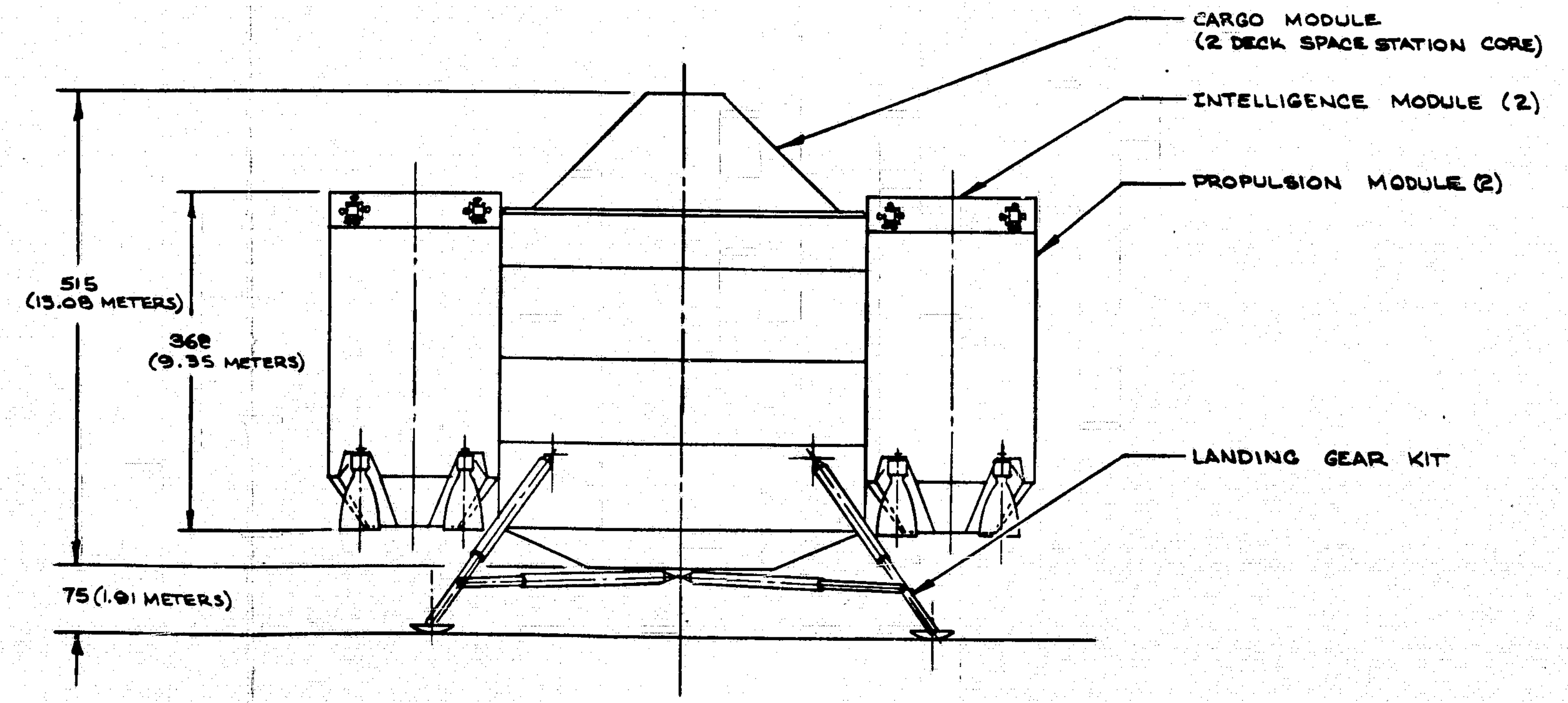


MANNED LUNAR LANDING TUG
(TOP MOUNTED CREW MODULE)

FOLDOUT FRAME



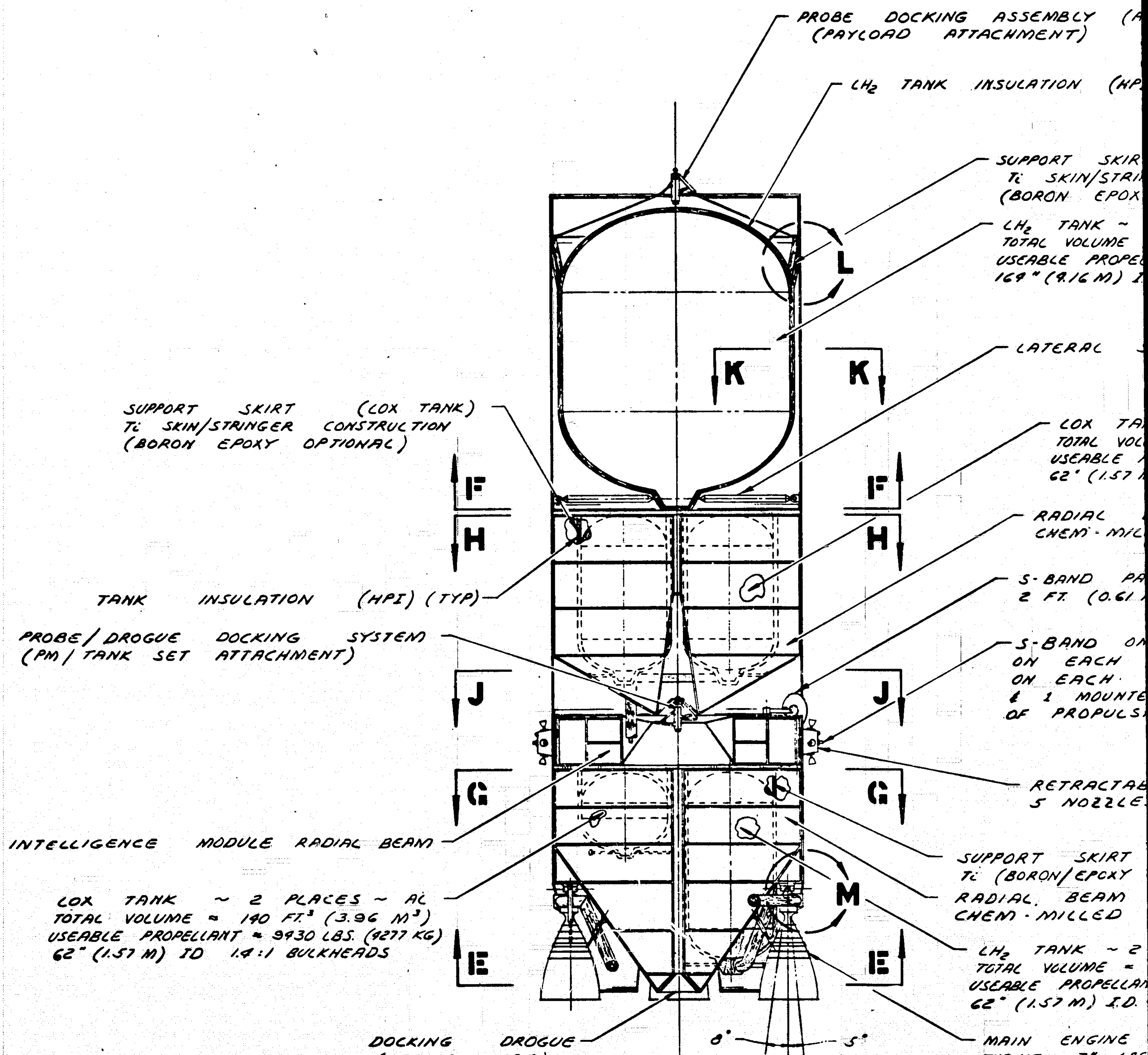
UNMANNED LUNAR LANDING TUG



UNMANNED LUNAR LANDING TUG
DUAL PROPULSION MODULES
(41.2 K USABLE PROPELLANT EACH)
(18,660 KILOGRAMS)

2283-33B SHT. 4 OF 4

Figure 3-3. Geosynchronous Mission, 41.2 K Pound Propellant Stage, Two-Stage Recoverable Unmanned Tug



SECTION A - A

FOLDOUT FRAME 1

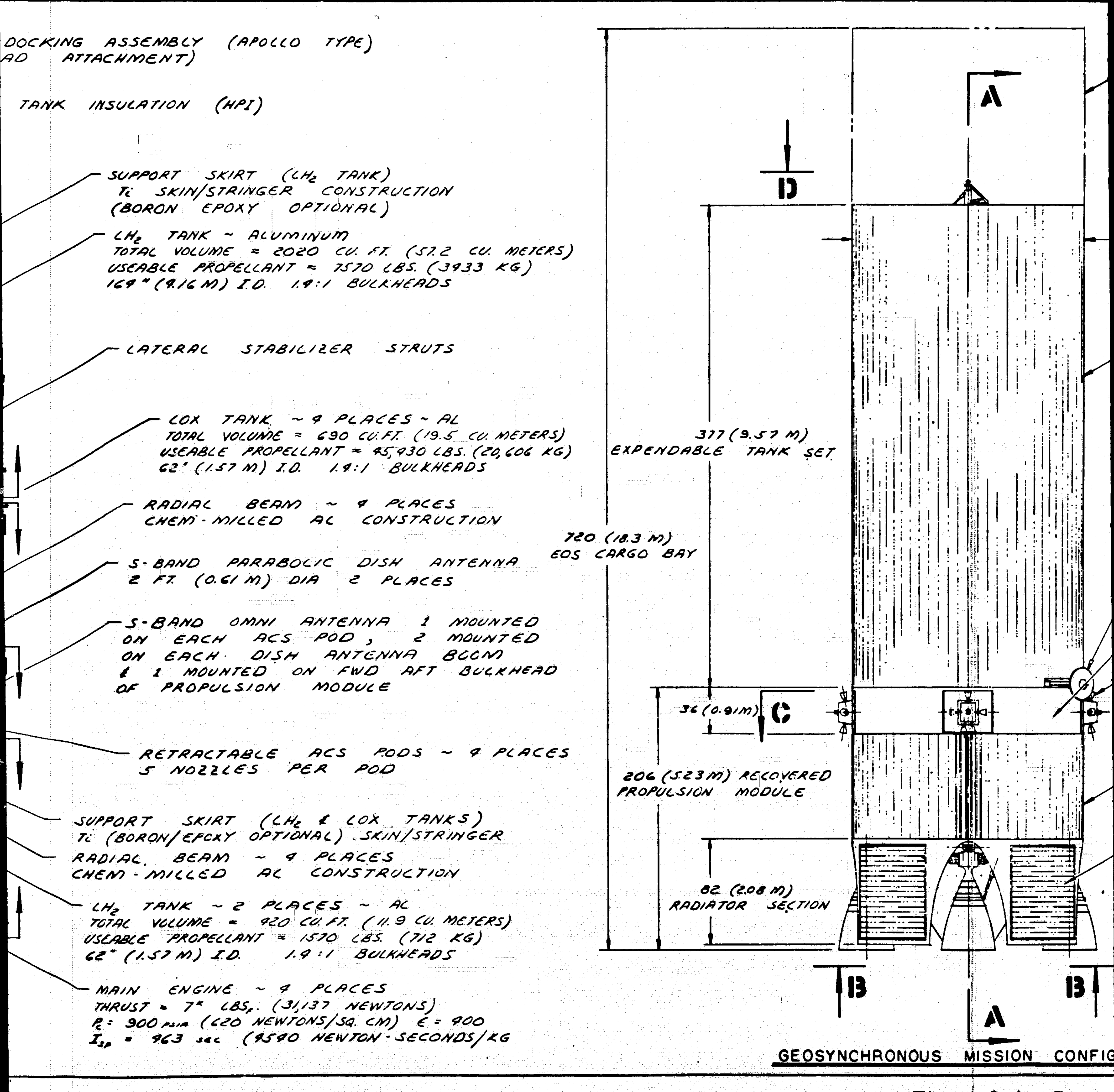


Figure 3-4. Conce
Recov

FOLDOUT FRAME 2

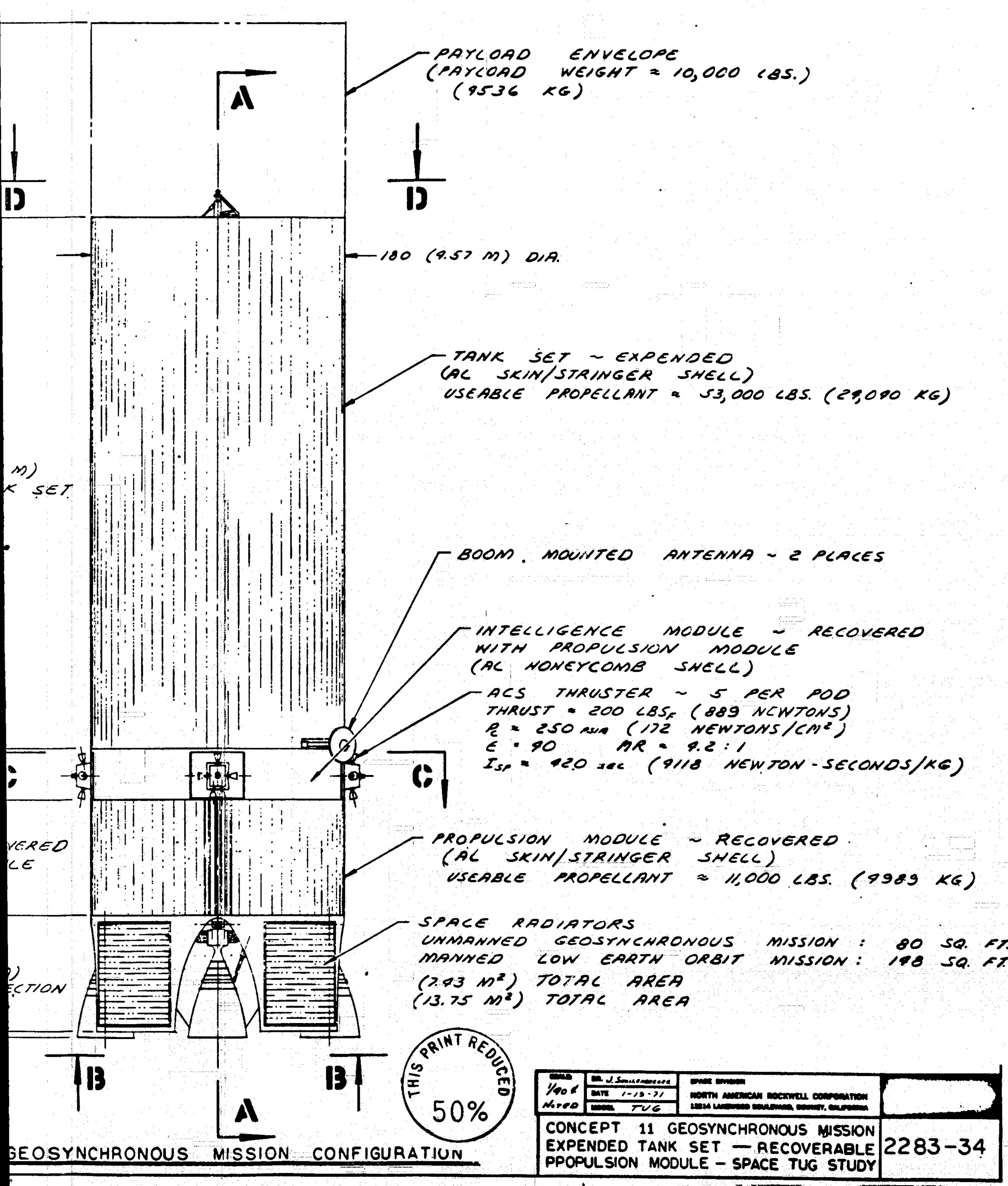
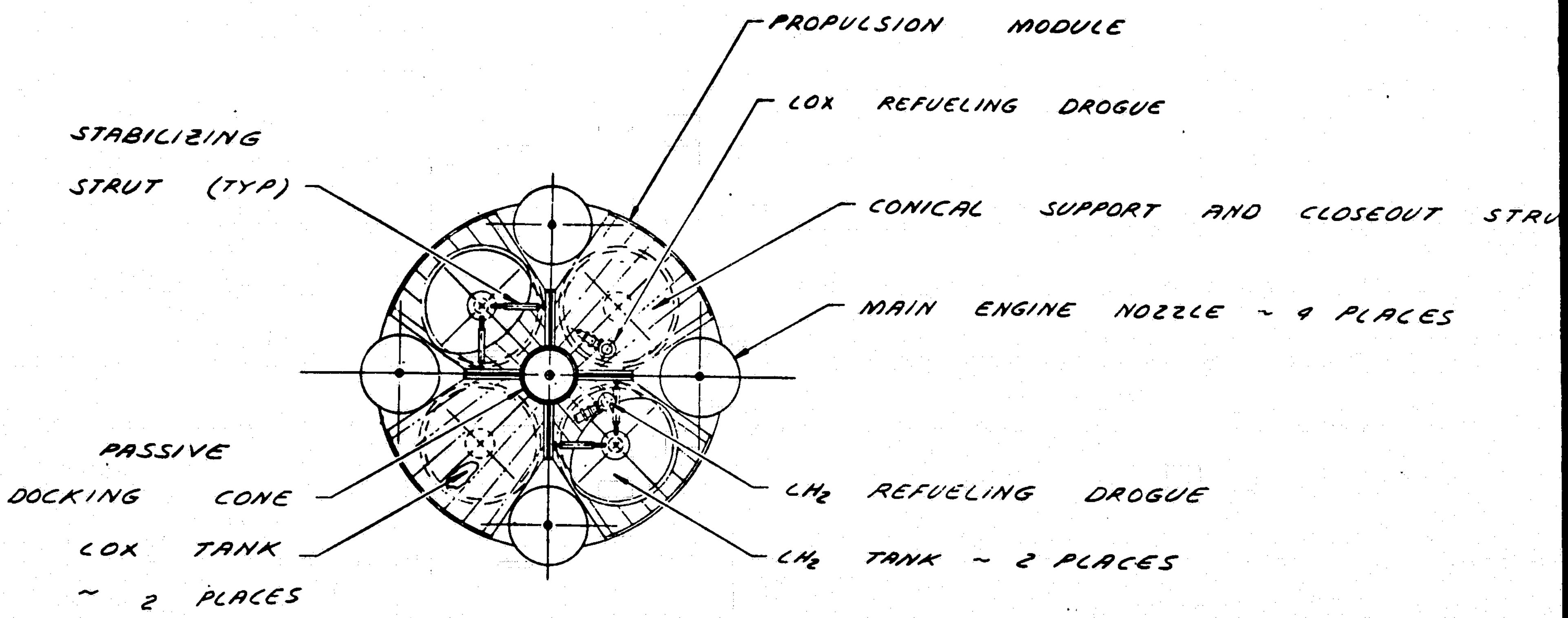
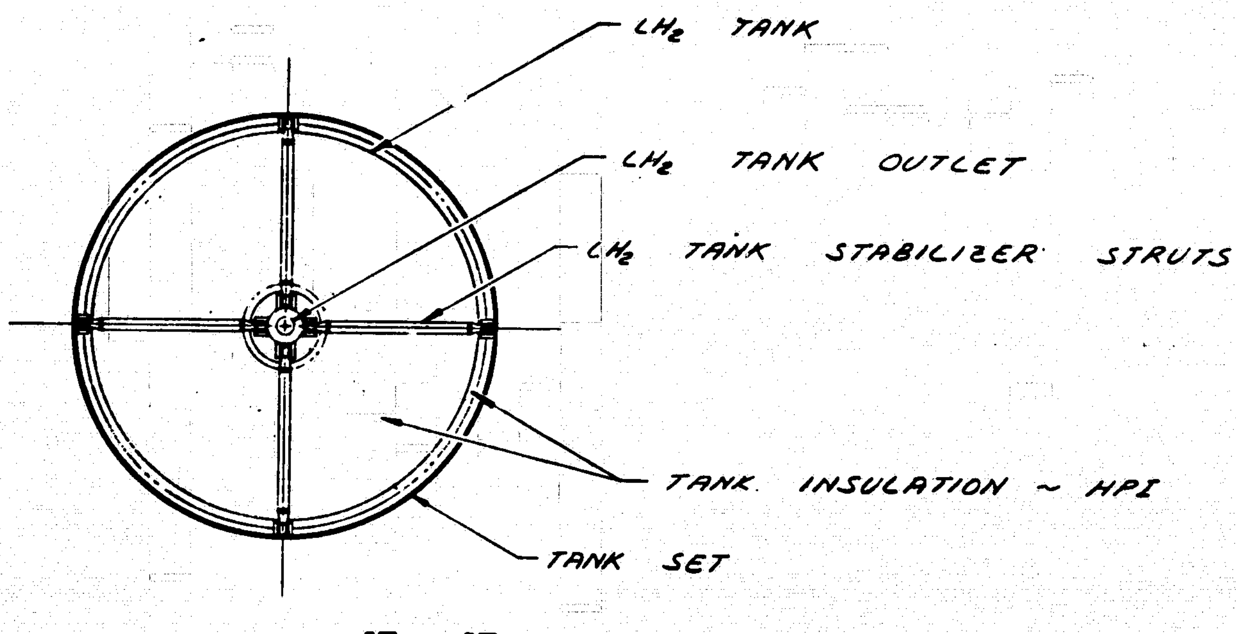


Figure 3-4. Concept 11, Geosynchronous Mission Expended Tank Set, Recoverable Propulsion Module

FOLDOUT FRAME 3 1 OF 5

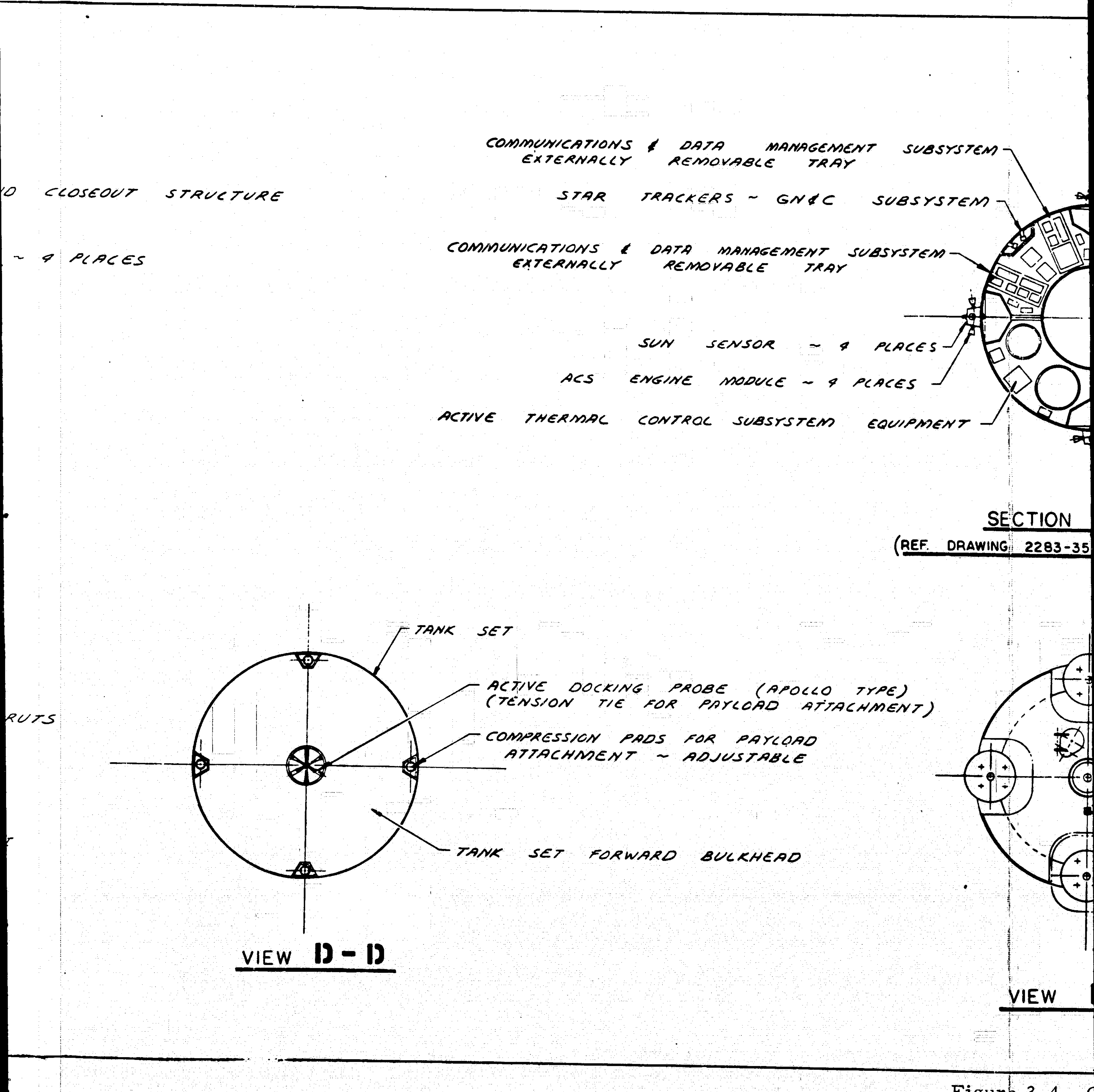


SECTION E - E



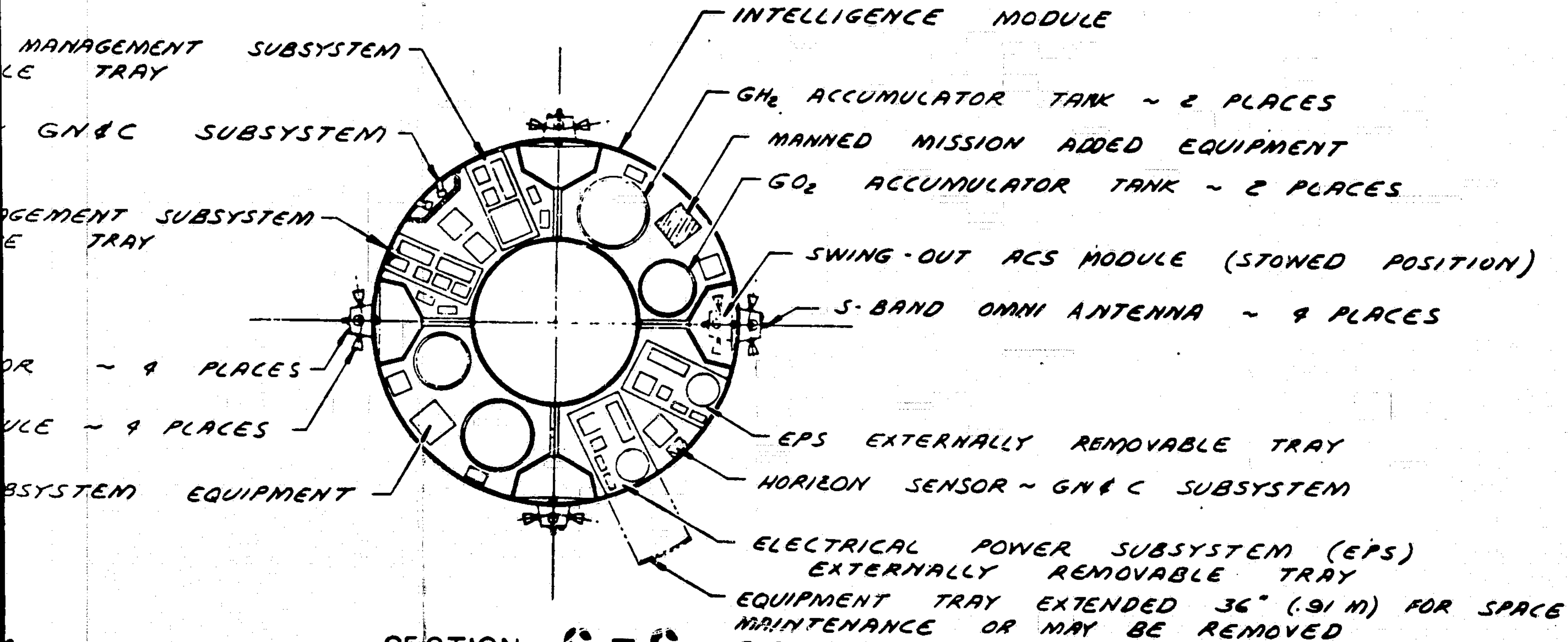
SECTION F - F

FOLDOUT FRAME 1

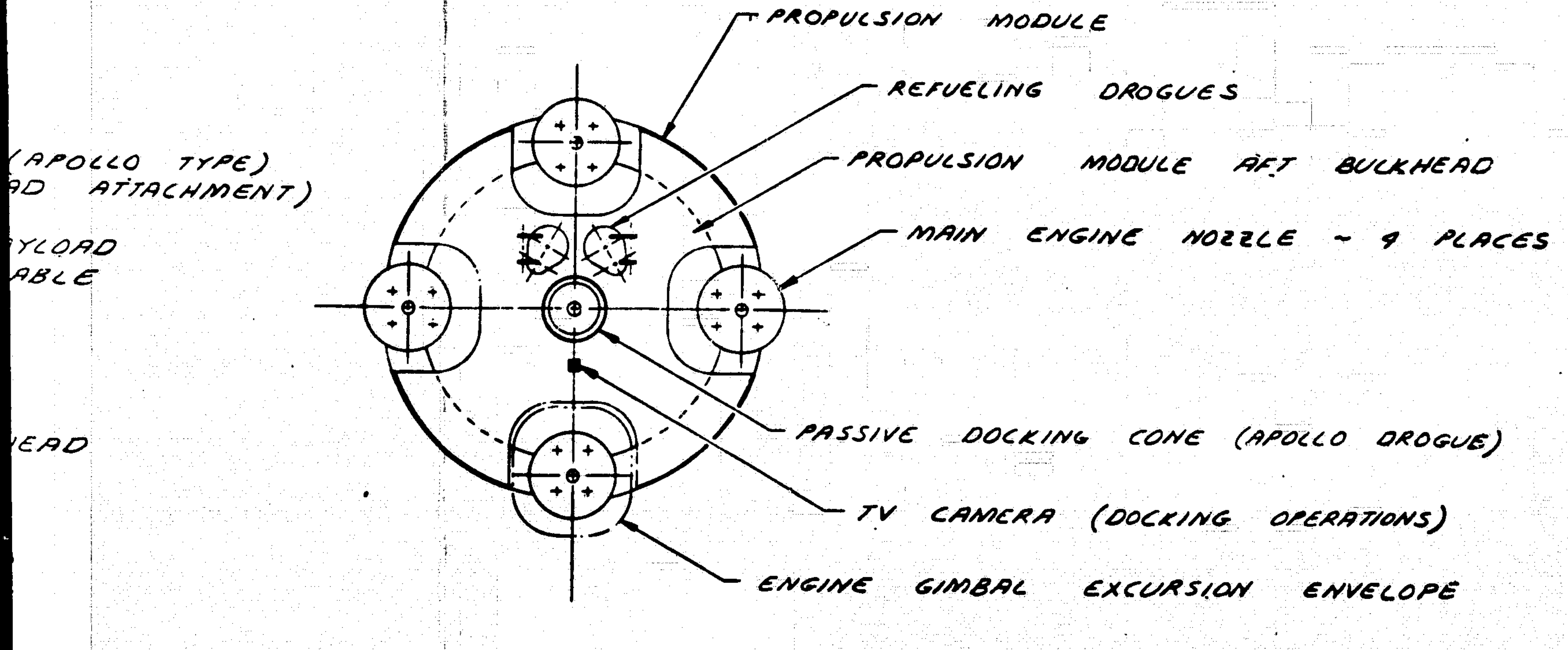


D CLOSEOUT STRUCTURE
~ 4 PLACES
RUTS
COMMUNICATIONS & DATA MANAGEMENT SUBSYSTEM
EXTERNALLY REMOVABLE TRAY
STAR TRACKERS ~ GN&C SUBSYSTEM
COMMUNICATIONS & DATA MANAGEMENT SUBSYSTEM
EXTERNALLY REMOVABLE TRAY
SUN SENSOR ~ 4 PLACES
ACS ENGINE MODULE ~ 4 PLACES
ACTIVE THERMAL CONTROL SUBSYSTEM EQUIPMENT
SECTION
(REF. DRAWING 2283-35
TANK SET
ACTIVE DOCKING PROBE (APOLLO TYPE)
(TENSION TIE FOR PAYLOAD ATTACHMENT)
COMPRESSION PADS FOR PAYLOAD
ATTACHMENT ~ ADJUSTABLE
TANK SET FORWARD BULKHEAD
VIEW D - D
VIEW

Figure 3-4.



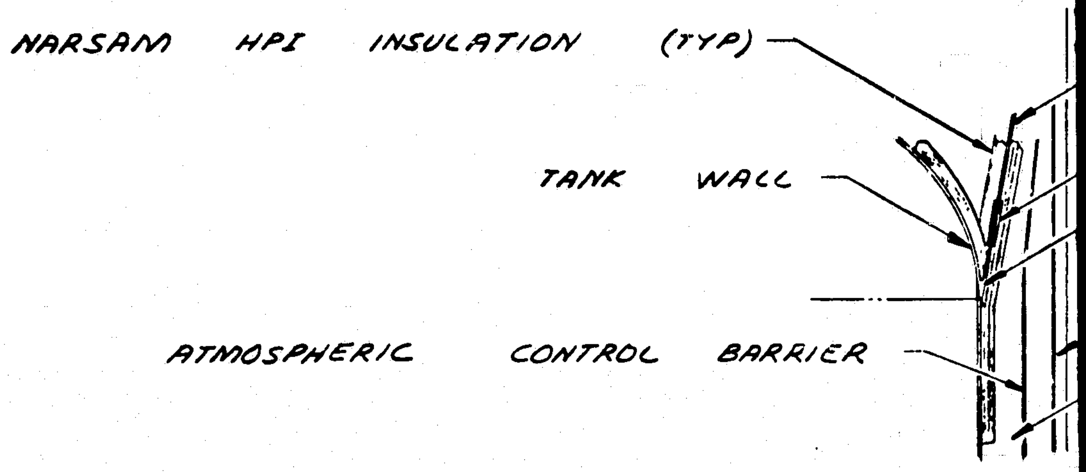
SECTION C - C
(REF. DRAWING 2283-35 FOR IM GENERAL ARRANGEMENT)



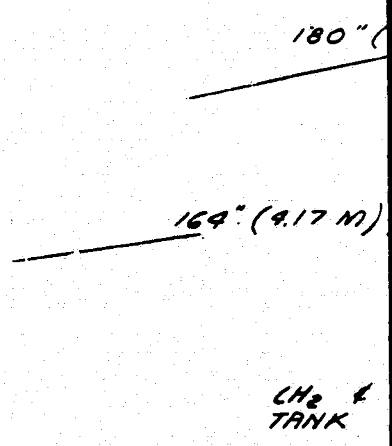
VIEW B - B

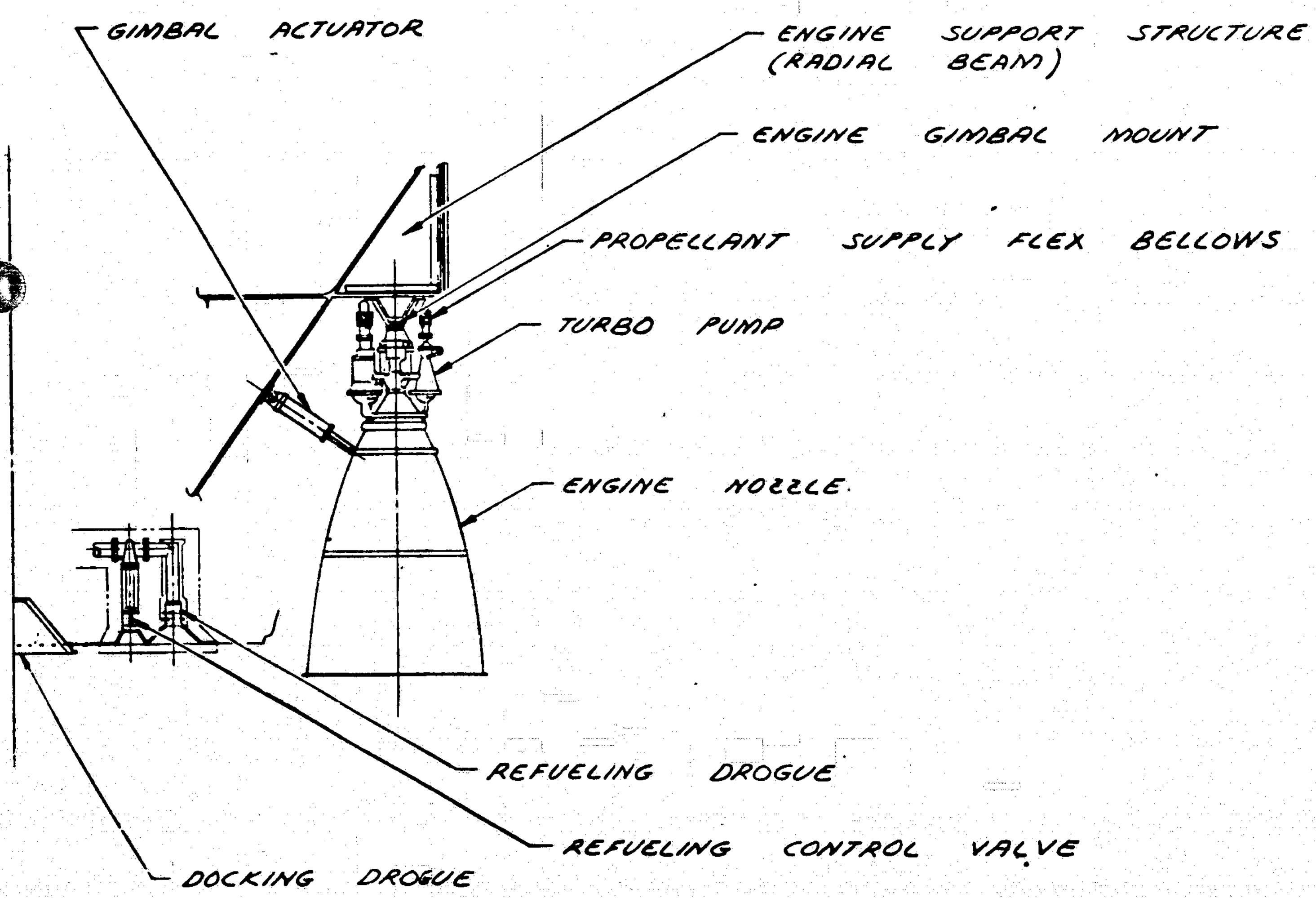
Figure 3-4. Concept 11, Geosynchronous Mission Expended Tank Set, Recoverable Propulsion Module

FOLDOUT 3 2 OF 5



DETAIL L
SCALE 1/10





DETAIL M
SCALE 1/20

FOLDOUT FRAME 1

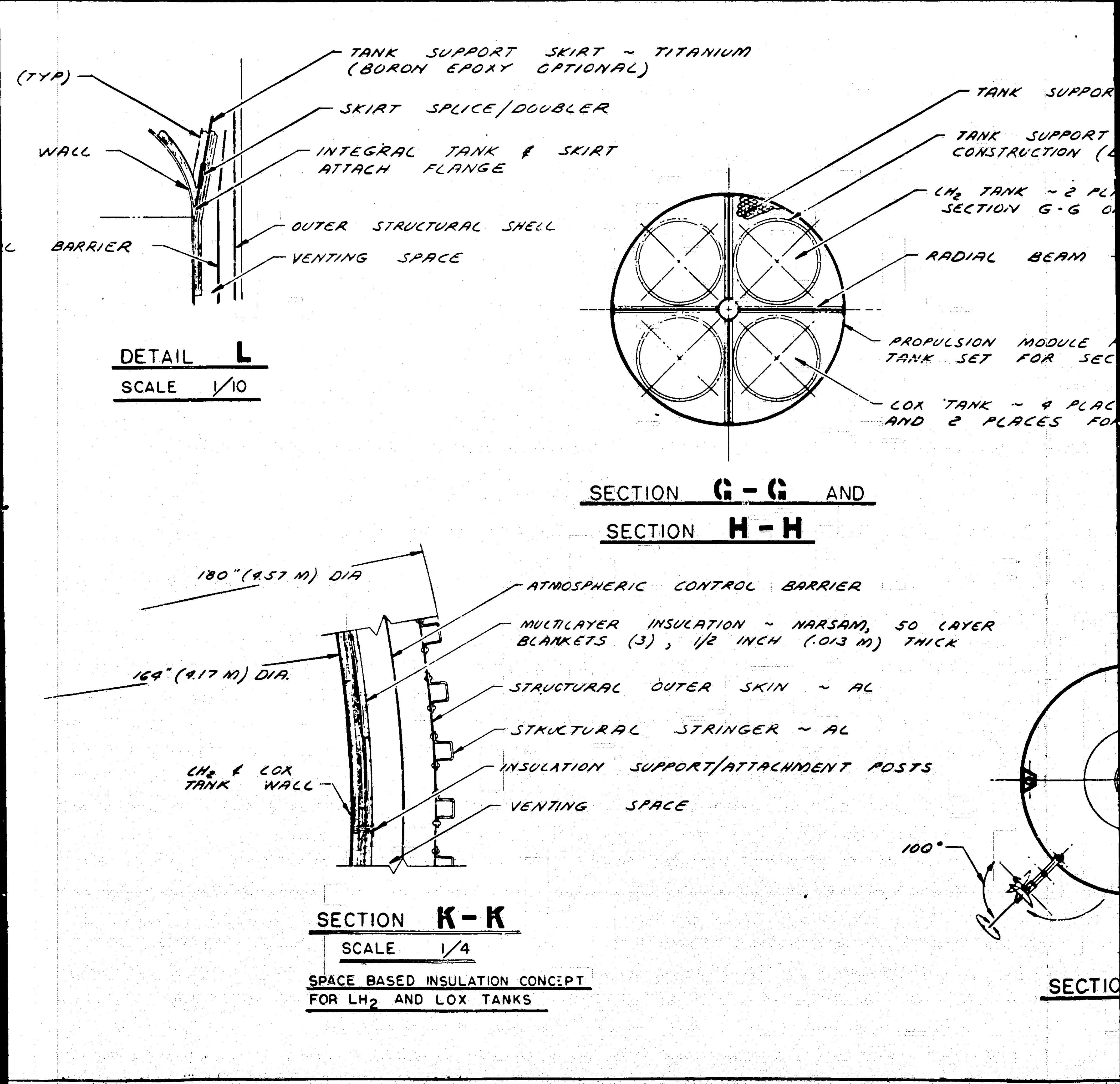


Figure 3-4.

FOLDOUT FRAME 2

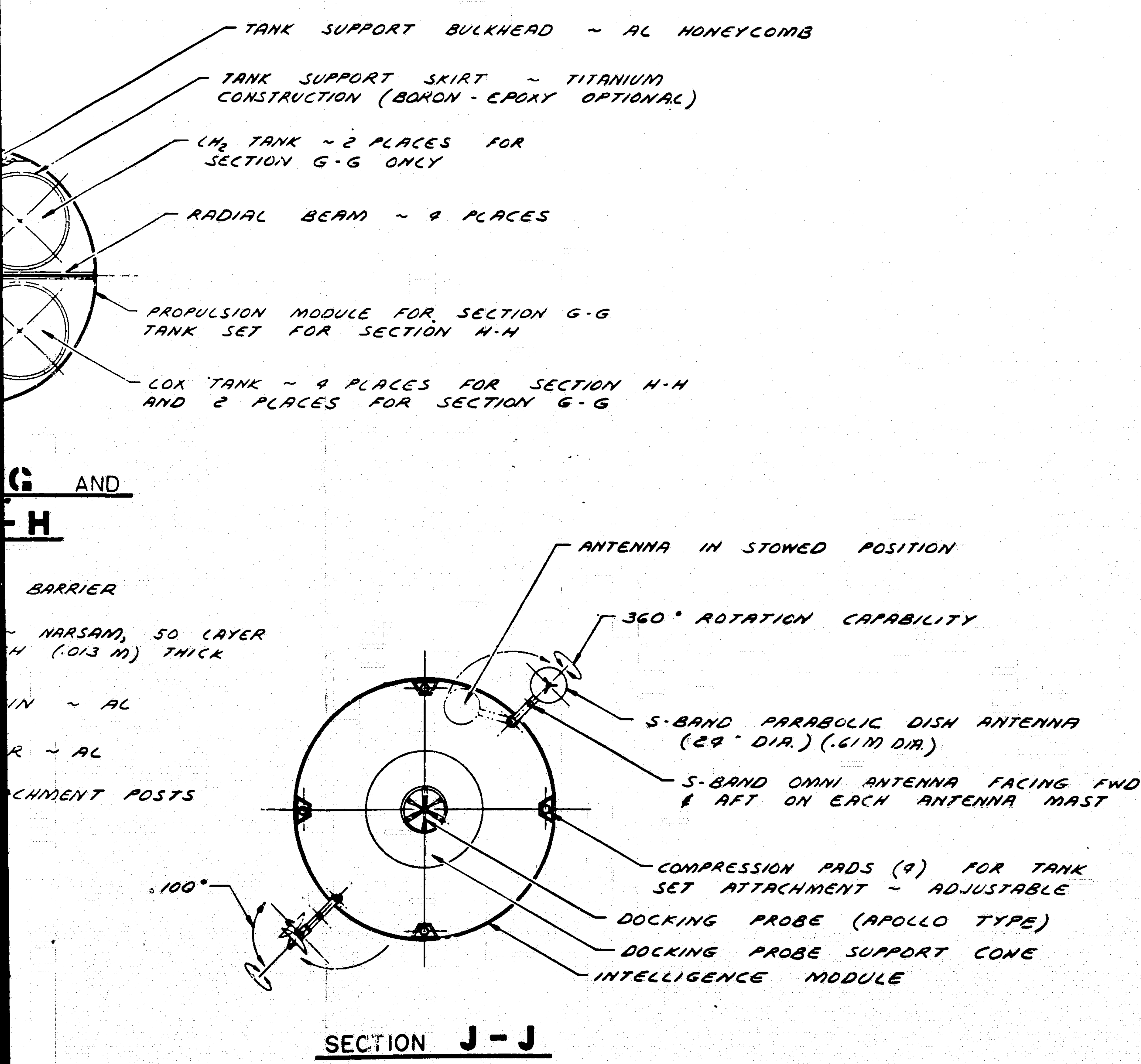
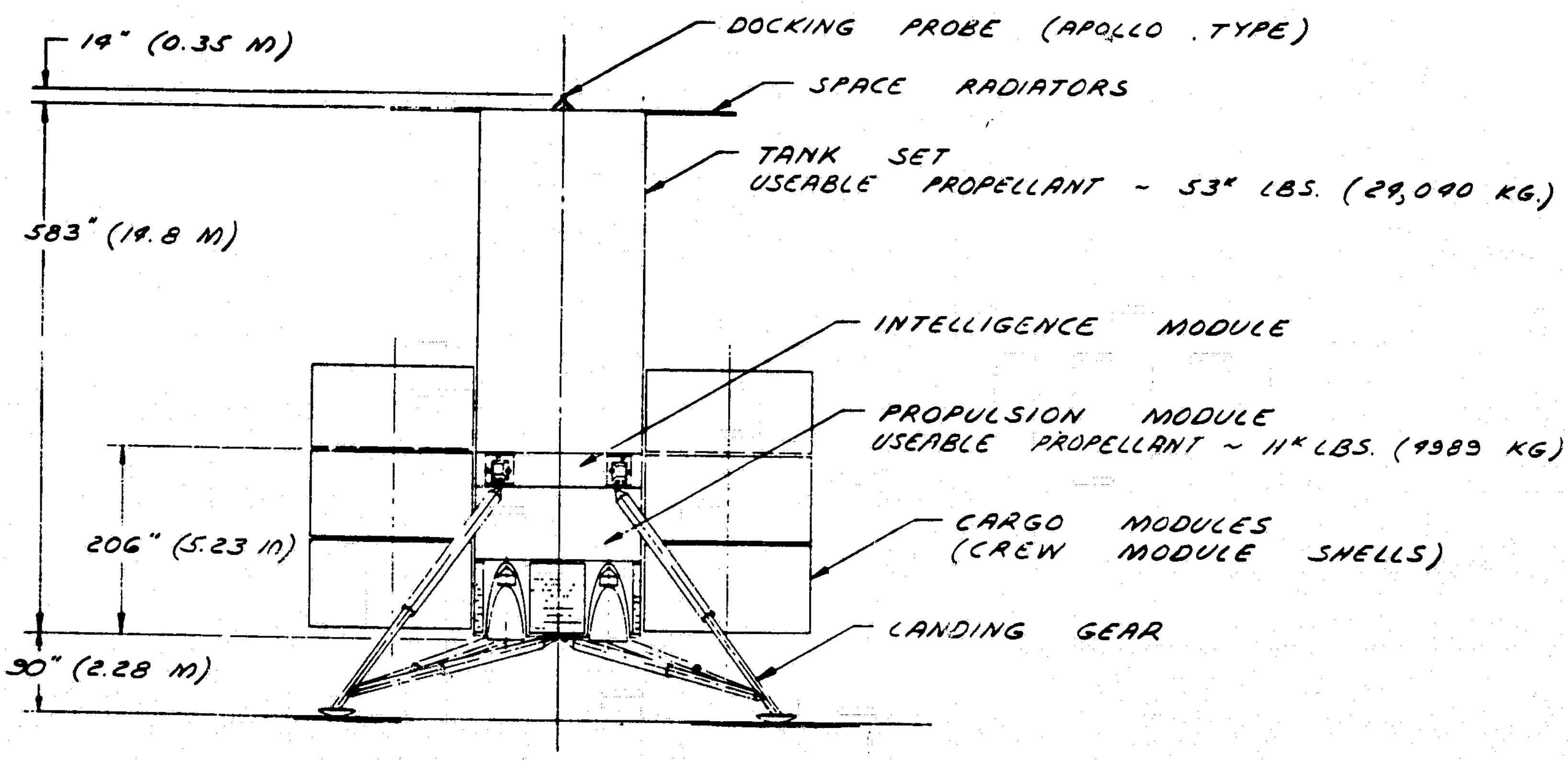
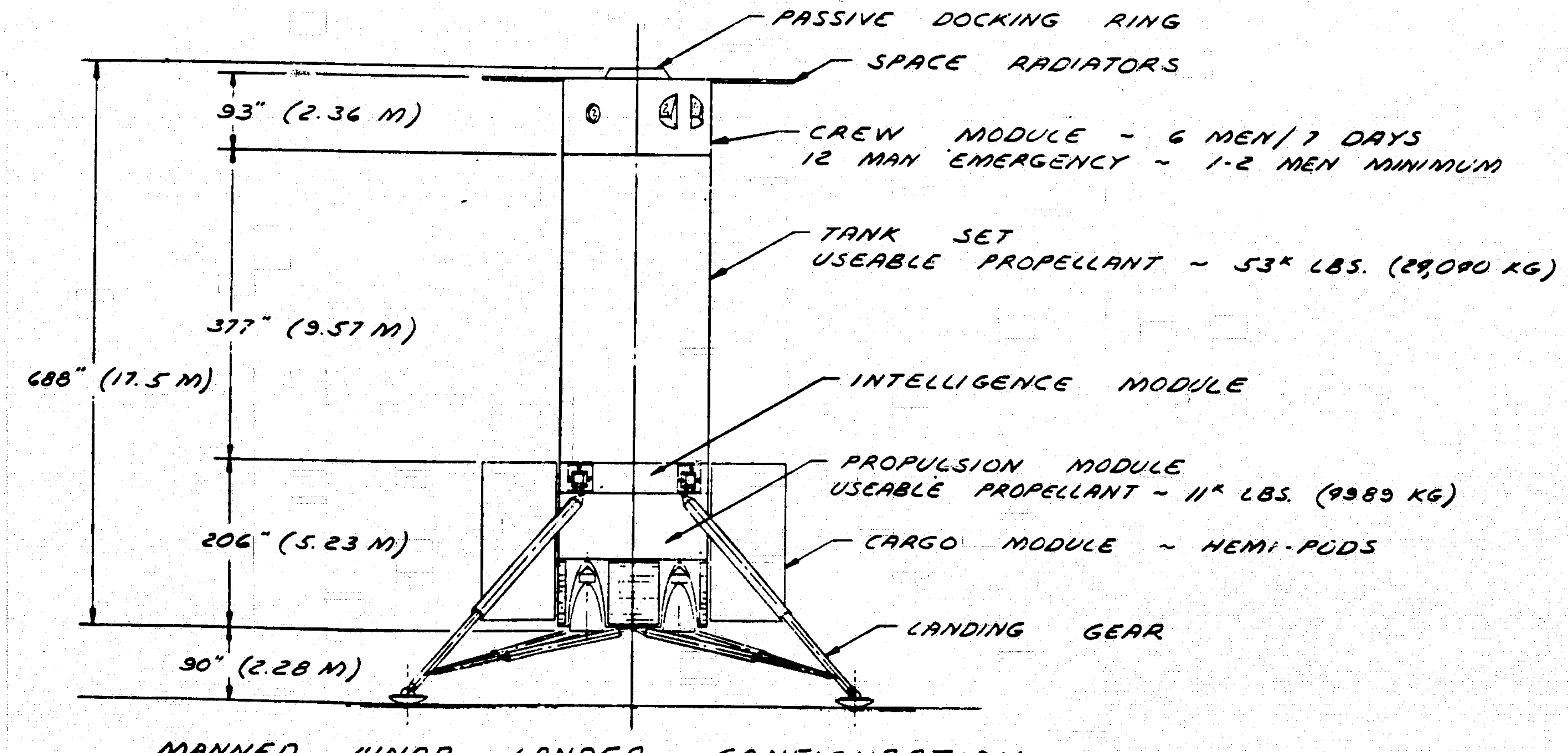


Figure 3-4. Concept 11, Geosynchronous Mission Expended Tank Set, Recoverable Propulsion Module



UNMANNED LUNAR LANDER CONFIGURATION

SCALE: 1/100

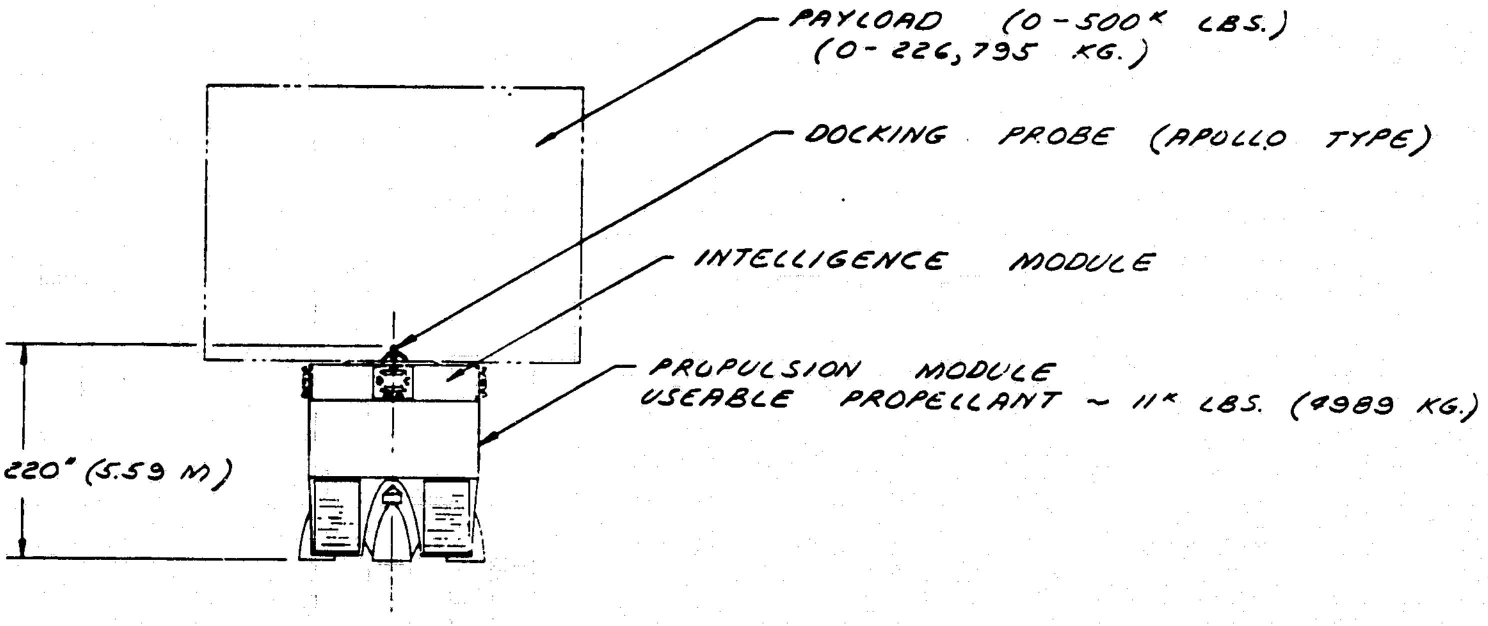


MANNED LUNAR LANDER CONFIGURATION

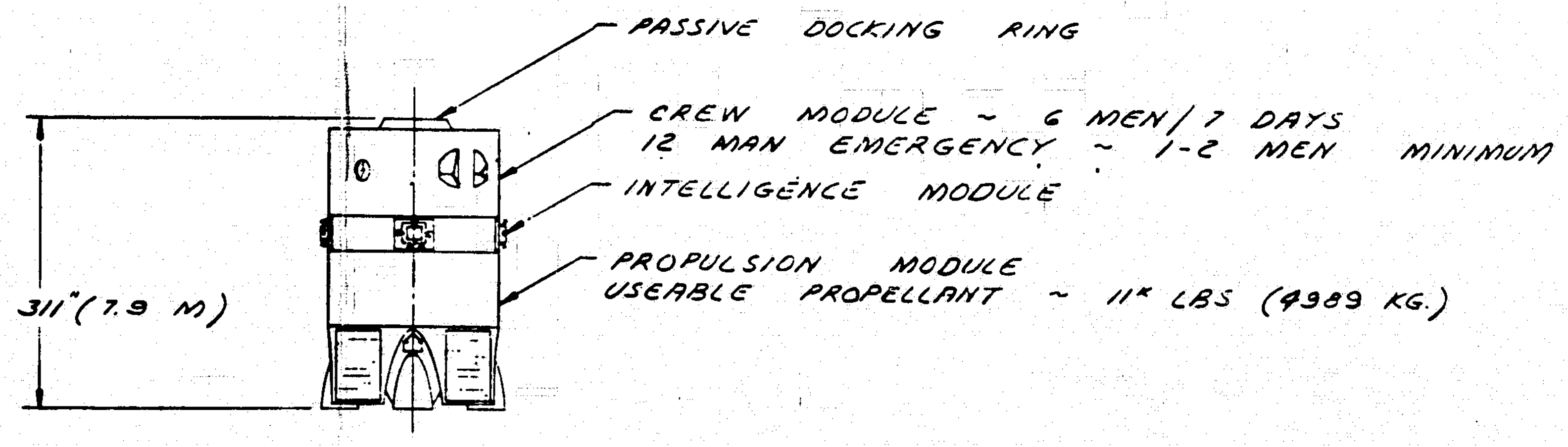
SCALE: 1/100

UNI

MANN

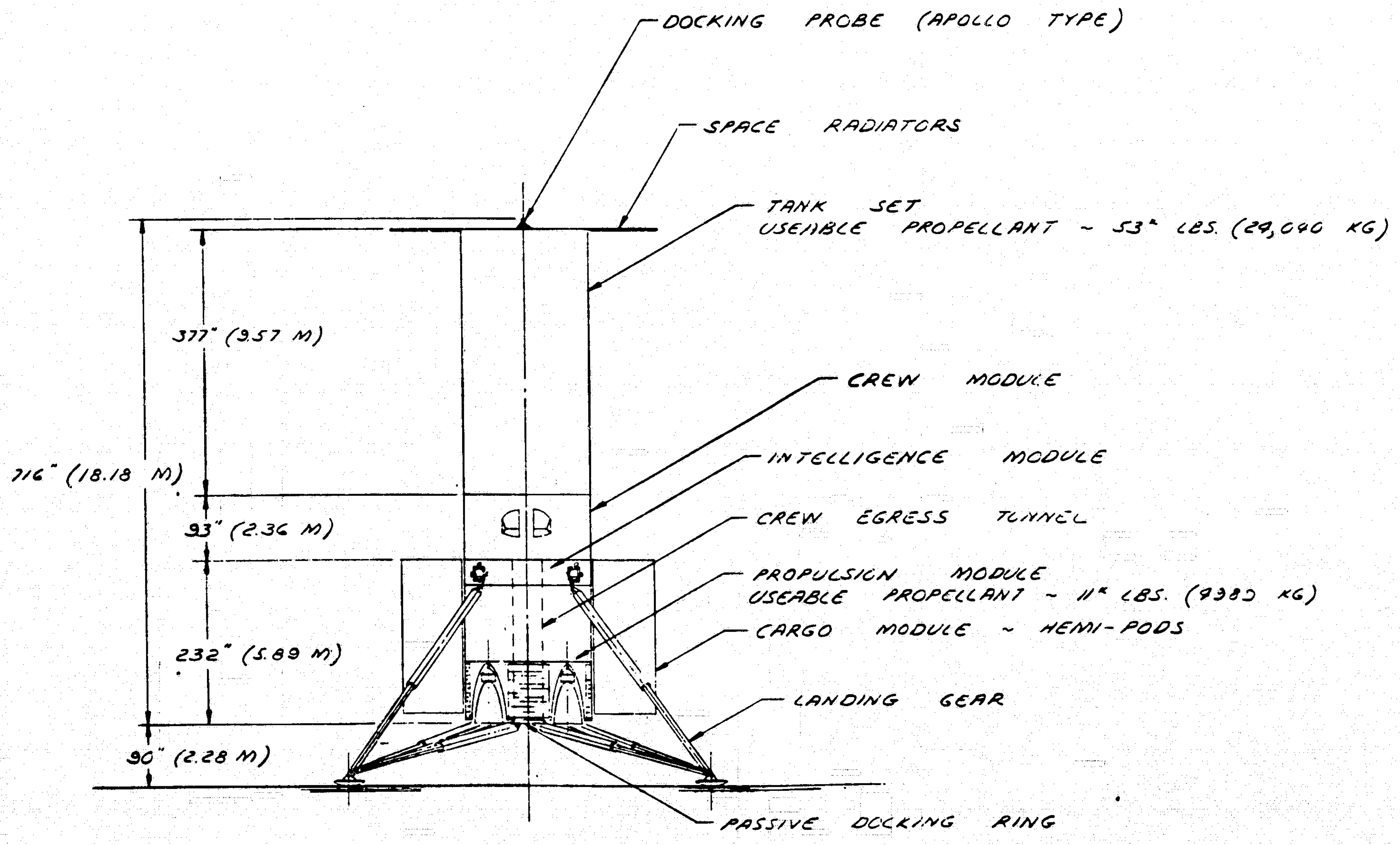


UNMANNED LOW EARTH ORBIT CONFIGURATION
SCALE : 1/100



MANNED LOW EARTH ORBIT CONFIGURATION
SCALE: 1/100

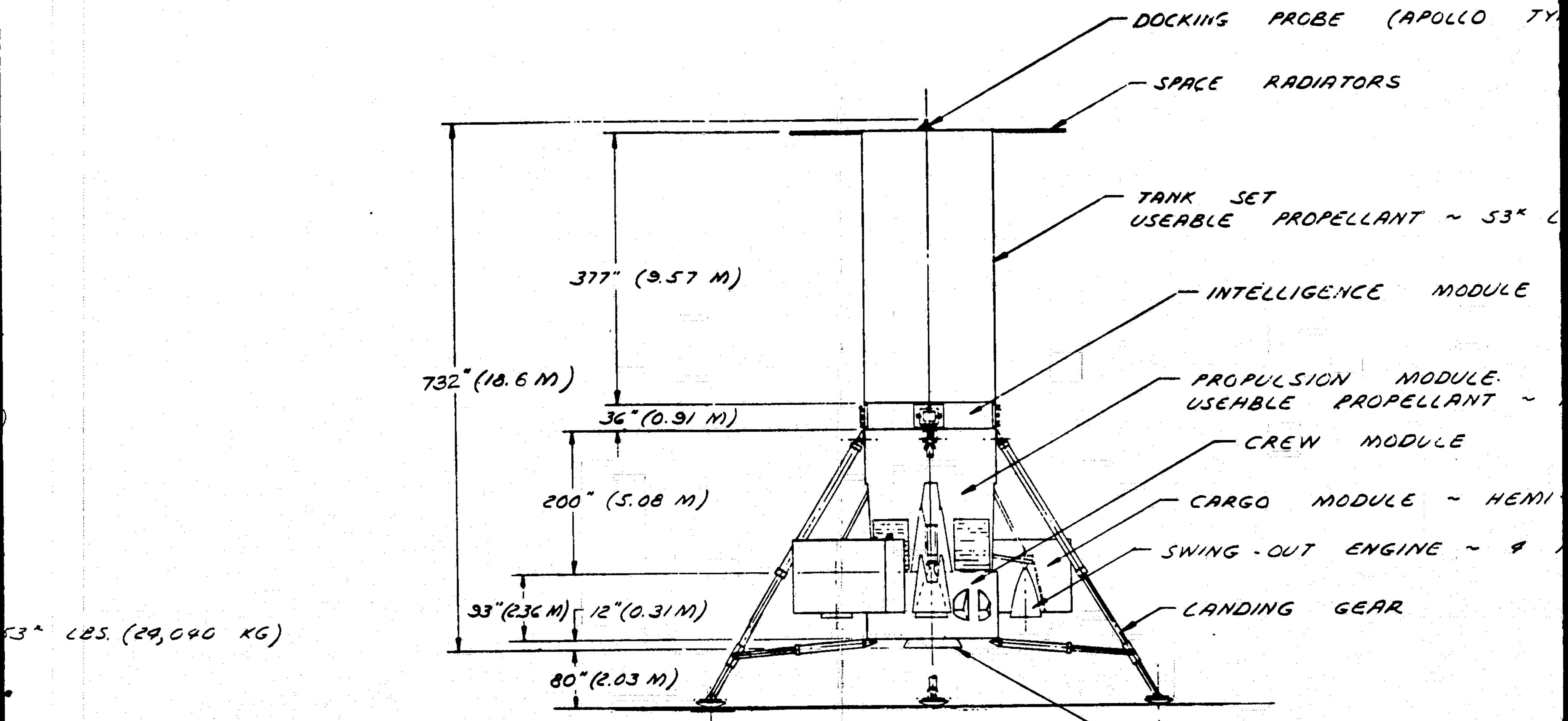
Figure 3-4. Concept 11, Geosynchronous Mission Expended Tank Set, Recoverable Propulsion Module



MANNED LUNAR LANDER CONFIGURATION

CENTER CM W/TUNNEL SCALE: 1/100

FOLDOUT FRAME 1



MANNED LUNAR LANDER CONFIGURATION
SWING-OUT ENGINE CONCEPT SCALE: 1/100

Figure 3-4. Concept 11, Geosynchronous Mission Expen
Recoverable Propulsion Module

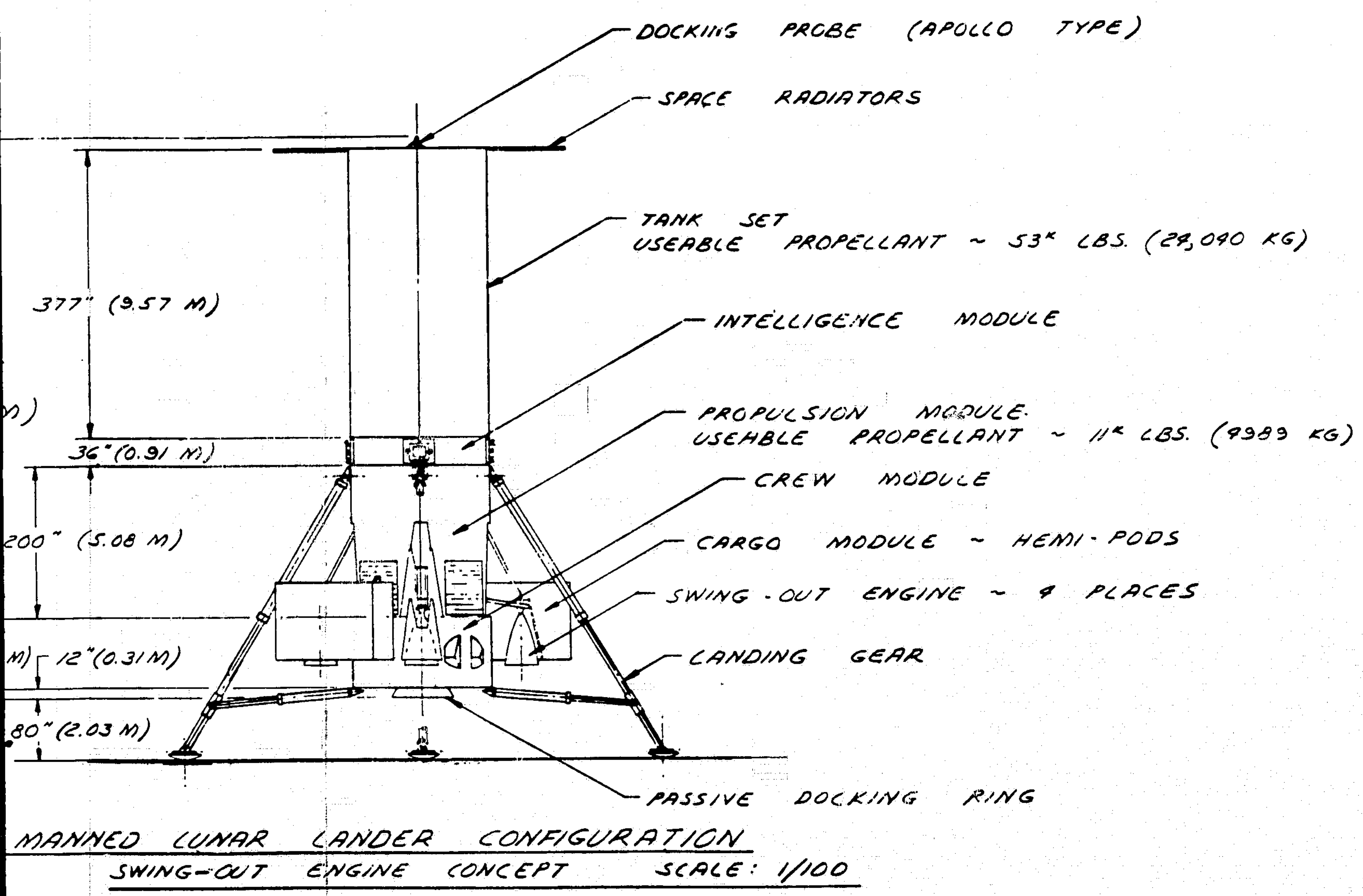


Figure 3-4. Concept 11, Geosynchronous Mission Expended Tank Set, Recoverable Propulsion Module

each, are 62 inches (1.57 meters) inside diameter and are 113 inches (2.87 meters) long inside. Both the LH_2 and LOX tanks are constructed of aluminum and are supported from titanium skin-stringer skirts. Glass- or boron-epoxy construction may be used for the skirts if warranted from structural and thermal considerations. Each tank incorporates the use of a non-thermal conducting sway brace at the end of the tank opposite the skirt. The sway braces will be constructed from boron-epoxy to meet the non-thermal conducting requirement. Each of the five tanks consists of a straight cylindrical center section with elliptical bulkheads. The elliptical bulkhead ratio is 1.4 to 1.

Each tank and skirt is fully insulated from the space environment by 1.5 inches (0.04 meters) of high-performance insulation attached directly to the tanks. An atmospheric control barrier is placed between the insulation and the inside shell of the tank set structure.

At the forward end of the tank set, located on the vehicle centerline, is an Apollo-type active docking probe supported by a conical skin-stringer structure. The conical structure is an integral part of the tank set which aids in distributing the docking loads to the outside structure. This cone attaches to the same structural ring as the liquid hydrogen tank support skirt and forms a major load-carrying frame. The exterior or outer shell of the tank set is a skin-stringer aluminum structure which constitutes the main load-carrying member between major frames.

The other major frame in the tank set is the main load distribution frame. It is stabilized by the four radial beams. This frame is located between the liquid hydrogen and liquid oxygen tanks and is used as the main support for the liquid oxygen tank skirts. The frame is of aluminum honeycomb shelf-type construction with stiffener webs. The shelf is stabilized and stiffened by the four radial beams. The beams are rib-stiffened chem-milled aluminum structures. These beams support the passive docking cone at the aft end of tank set and also support the compression members for attachment of the tank set to the propulsion module. The liquid hydrogen tank aft sway braces are attached to the outer rim of the liquid oxygen tank support frame. The sway braces for the liquid oxygen tanks are attached to the radial beams. The outer skin-stringer structure aft of the liquid oxygen tank support frame is again the main load distributing structure and stabilizing member for the radial beams.

The recoverable PM is comprised of the main engines and two each liquid hydrogen and liquid oxygen tanks, plus the IM. The PM/IM combination is 206 inches (5.23 meters) long plus 14 inches (0.35 meters) for the docking probe at the forward end. The hydrogen tanks have an internal volume of 210 cubic feet (5.9 cubic meters) each, are 62 inches (1.57 meters)

inside diameter and are 135 inches (3.43 meters) long inside. The LOX tanks have an internal volume of 70 cubic feet (1.9 cubic meters) each, are also 62 inches (1.57 meters) inside diameter and are 55 inches (1.4 meters) long inside. Both the LH_2 and LOX tanks are constructed from aluminum and supported from titanium skirts. The concept for supporting these tanks is identical to that employed by the LOX tanks in the tanks set. This is evidenced by the commonality of sections "G" and "H" on the drawing. The tanks are stabilized by sway braces constructed of boron epoxy. Each tank consists of a straight cylindrical center section with elliptical bulkheads of 1.4 to 1 ratio. Commonality of the bulkheads for all the 62-inch (1.57 meter) diameter tanks will reduce the tooling and cost of each tank.

Each tank is insulated from the space environment by high performance insulation attached directly to the tank and an atmospheric control barrier between the insulation and the PM outer shell.

The PM is structurally similar to the LOX tank portion of the tank set. The PM tanks are suspended from a major load distributing frame at the forward end. This frame is stabilized by four radial beams. The frame is of aluminum honeycomb shelf-type construction with stiffener webs. The rib-stiffened chem-milled aluminum beams stiffen and stabilize the frame. The beams also provide mounts for the four main engines and support the aft passive docking cone. The sway braces for the LH_2 and LOX tanks also are attached to the radial beams. The outer skin-stringer shell structure becomes the main load distributing member and also stabilizes the radial beams. The IM is attached to the forward end of the PM.

The general configuration of the IM is 15 feet (4.57 meters) in diameter and 3 feet (0.91 meter) long. The module has four ACS pods with five engines on each pod. Two of the engines face aft, one forward and one to each side. (See Volume 5 of this report for subsystem sizing and redundancy philosophy.) The pods are retractable within the IM to meet the EOS cargo bay 15-foot (4.57 meter) diameter requirement. The communications and data management, guidance and navigation, attitude control, environmental control, and electrical power equipment all is located within the intelligence module. A total of 62 pounds (28 kg) of gaseous usable propellant for ACS and EPS is contained within the IM in four accumulator tanks.

At the forward end of the IM located on the vehicle center line is an Apollo-type active docking probe supported by a conical skin-stringer structure. This structural cone is an integral part of the IM and aids in distributing the docking loads to the outer structural shell.

The PM accommodates four main engines. Each engine has a 7000-pound (31,137 newtons) thrust level at a chamber pressure of 900 psia (620 newtons per square centimeter). The engine operates at a mixture ratio of 6 to 1 with a specific impulse of 463 seconds (4540 newton-seconds per kg). The nozzle area expansion ratio is 400:1. Each engine is mounted to a radial beam within a cavity at the aft end of the propulsion module. The engines gimbal inboard to accommodate the 15 foot (4.57 m) diameter EOS cargo bay clearance requirement. Each engine nominally gimbals 5 degrees in two orthogonal directions for thrust vector alignment. Detail M of Figure 3-4 depicts the installation of the engine to the radial beam. Propellant refueling drogues and control valves are located on the aft bulkhead of the PM as shown in View B-B, section E-E and detail M of the figure. The refueling drogues and valves are completely insulated from the space environment to minimize contamination and boiloff. The controls for each refueling system are separate. This enables refueling to be performed sequentially and selectively thus eliminating potential problems which might arise when simultaneously transferring both LH_2 and LOX propellants. The transfer of propellants would be performed only when the tug vehicle is hard docked to the propellant supply vehicle.

The tank set is attached to the PM through use of the probe/drogue docking system in conjunction with compression pads. The adjustable compression pads are located on the forward end of the PM as shown in section J-J of the figure. When docked, the probe/drogue assembly is used as a tension tie. The compression pads are extended until they contact the base of the tank set and then are adjusted to load the docking probe to a predetermined tension load. This same arrangement is provided at the forward end of the tank set for payload attachment.

Two types of S-band antennas have been used for communications. Two 24-inch (0.61 meter) diameter parabolic dish antennas are mounted on booms on opposite sides of the vehicle on the forward end of the IM. Section J-J demonstrates the stowage of these dishes to meet the 15-foot (4.57 meter) diameter cargo bay restriction. The dish antennas can rotate 360 degrees about their deployment boom and also hinge 100 degrees on an axis perpendicular to the boom to give more than a hemisphere of coverage for each antenna. Ten omni-type S-band antennas are used throughout the tug vehicle. Two of them are mounted on each dish antenna boom, one facing forward and one facing aft. One omni antenna is located on each of the ACS pads. The last two omni antennas are located on the forward and aft ends of the tug vehicle. Sun acquisition sensors also are located on each of the ACS pads. Docking equipment is located on the forward and aft bulkheads of the PM in the form of TV cameras and laser rendezvous and docking radar.

The environmental control system consists of equipment within the IM and radiators on the external surface of the PM. A total of 80 square feet (7.43 square meters) of radiator area is required for the unmanned geosynchronous mission. However sufficient area has been provided for a total of 148 square feet (13.75 square meters), the requirement for manned earth orbit missions. The radiators are located on the PM aft skirt section of the skin-stringer shell. This section of the shell has been configured as a separate skirt to facilitate quick change of the radiator panels from unmanned to manned configurations.

The tank insulation concept adopted for the space-based Tug is shown in section K-K and detail L. If a ground-based approach were adopted, the major change would be the addition of a foam insulation between the tank wall and the HPI (Narsam) insulation. A discussion of cryogenic thermal control is contained in Appendix F.

The basic PM, IM and tank set which have been previously discussed, may be used in various combinations with other modules and kits to perform a wide variety of missions. Six such missions are depicted in small scale drawings at the end of Figure 3-4. The first of these missions utilizes the basic PM and IM to perform low earth-orbit unmanned transfer of various size payloads. The second approach incorporates the addition of a CM forward of the IM to execute manned low earth orbit missions. Each of these concepts makes available 11,000 pounds (4989 kg) of usable propellant. The unmanned configuration is 200 inches (5.59 meters) long and the manned is 311 inches (7.9 meters) long.

The remaining four configurations are lunar landers. Each lunar lander utilizes a landing gear kit and a space radiator kit which radiates toward space when the tug is on the lunar surface. The first configuration shown is an unmanned lander which utilizes a PM, IM, and tank set. In this concept the tank set is recovered. Several cargo modules which have been configured as CM shells are shown attached to the side of the tug. The basic vehicle is 583 inches (14.8 meters) long. The next lander is manned and is identical to the first with the addition of a CM on the forward end. The cargo modules shown are hemipods which when put together fit within the EOS cargo bay 15-foot (4.57 meters) diameter envelope. This concept is 688 inches (17.5 meters) long.

The last two configurations are manned landers with different PM's than the basic module. The first concept utilizes a bottom-mounted CM and a PM with swing-out engines. This configuration is made up from aft to forward of a CM, a PM with swing-out engines, an IM and a tank set. All pieces of the configuration are recoverable. The radiators have been mounted on the forward end of the tank set. Hemipod cargo modules have been

incorporated near the aft end of the vehicle. The basic difference in length between the baseline PM is an addition of 30 inches (0.76 meter). The vehicle overall is 732 inches (18.59 meters) long.

The last lunar lander configuration features a centrally located CM and a PM with a crew egress tunnel with it. The basic configuration of the PM has been retained with smaller diameter LH_2 and LOX tanks to allow room for the 42-inch (1.07-meter) diameter crew egress tunnel in the center of the PM. The CM is located forward of the IM. Hemipod cargo modules have been used on this concept. The PM is 26 inches (0.66 meter) longer than the basic module which is the increase in length of the tanks. The vehicle is 716 inches (18.18 meters) long. As in the previous configuration, the space radiators are located on the forward end of the tank set.

All of the lunar lander configurations are off loaded to make 53,000 pounds (24,040 kg) of propellant available in the tank set and 11,000 pounds (4989 kg) of propellant available in the PM for the Mode A landing mission.

3.2.5 IM CONCEPT DEVELOPMENT

The separable IM which is used on all tug vehicles is defined on Figure 2-22. The IM is configured as a rectangular cross sectional area toroid with inner diameter of 80 inches (2.03 meters) and outer diameter of 180 inches (4.57 meters). The ACS modules extend beyond the 180-inch (4.57 meters) diameter but are retractable in order to be stowed within the 180-inch (4.57 meters) diameter envelope of the EOS cargo bay. The IM is 36 inches (0.91 meters) high. All of the subsystems equipment required to operate the tug vehicle is accommodated within the IM.

The plan view of the IM shown in section A-A, indicates how the area has been divided into four basic pie-shaped sectors. This helps to keep subsystems separated and thus easily accessible for service and replacement of equipment. Quadrant I contains the communications and data management, guidance and navigation and rendezvous and docking equipment. All of the attitude control, thermal control and environmental control and life support equipment is located in quadrants II and IV. The remaining quadrant, III, contains electrical power equipment plus a small amount of guidance and navigation and rendezvous and docking equipment.

The swing-out ACS modules are located at the intersections of quadrants and retract into cavities at these intersections. The communications and data management and electrical power equipment are located on

externally removable trays which make space maintenance of this equipment feasible with a minimum of perturbation to the remainder of the IM. The trays may be moved on rails to extend 36 inches (0.91 meter) outside the outer mold line for access to the equipment. Also the trays may be completely removed from the IM for replacement. The trays are made of an aluminum honeycomb outer panel and an aluminum honeycomb central shelf to which the equipment is mounted as well as the removal rail guides. Quadrants I and III each contain two trays.

Quadrants II and IV have aluminum honeycomb outer panels and an aluminum honeycomb central shelf. The ACS tanks and other equipment within these quadrants is mounted on the shelf. Sections of the outer panels are removable for access to the equipment, but there are no removable or sliding trays in these quadrants. The central cylinder of the IM is closed out by an aluminum skin. The forward and aft bulkheads also are closed out with aluminum skins attached to circumferential frames. Radial beams of rib stiffened chem-milled aluminum construction form the ACS engine module cavities and stabilize the main circumferential load distributing frames. The ACS engine modules are mounted to aluminum honeycomb swing-out panels.

A guidance and navigation base is provided in Quadrant I. The two-axis gimbaled star trackers are mounted to this base. Large cavities are provided in the structure, as shown in View C-C of Figure 2-22 to accommodate the gimbal excursion envelope of these trackers. Additional guidance and navigation equipment is also mounted to this base as shown in Section B-B of the figure. This equipment is mounted to the inboard side of the base just aft of the quadrants control shelf. Opposite the G & N base, in quadrant III, another cavity is provided for mounting of the horizon sensor (earth edge tracker).

Quadrants II and IV are nearly identical in equipment arrangement with a small exception. Quadrant II contains the ACS engine diver electronics and quadrant IV contains the main propulsion engine gimbal electronics, as well as the active thermal control equipment. Each quadrant accommodates one each GH_2 and GO_2 accumulator tank. The GH_2 tanks are 27 inches (0.69 meters) inside diameter and contain 6 pounds (2.72 kg) of useable propellant. The GO_2 tank inside diameter is 22 inches (0.56 meters) and it contains 25 pounds (11.34 kg) of useable propellant. The tanks are mounted directly to the central shelf through insulating blocks. Each tank is fully insulated from the surrounding environment with HPI insulation blanket attached to each tank. Tank controls and regulators are located adjacent to each tank.

Each ACS module is made up of an engine mount which is attached to a swing-out outer panel. The internal arrangement of the engine mount can be seen in Sections D-D and E-E. Each mount accommodates five engines. Two engines face aft, one faces toward and two face sideways, one to each side.

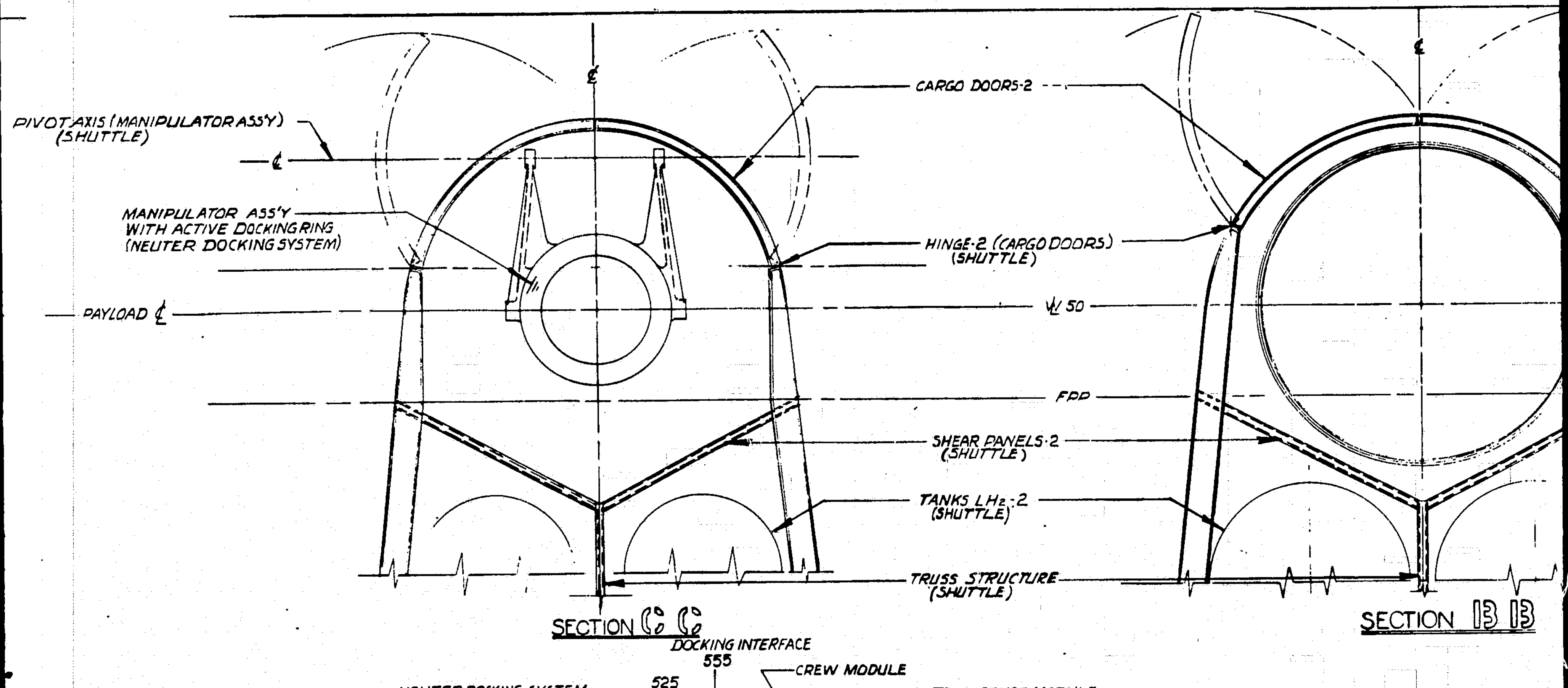
Propellant is supplied to the engines through a manifold within the mount. The outer panel pivots 180 degrees for launch in the EOS cargo bay in order to honor the 180-inch (4.57 meters) diameter envelope. An aluminum skin closeout is provided on the book side of the outer panel to protect the propellant lines and engine mount when the modules are retracted. The engines have a thrust level of 200 pounds (889 newtons) at a chamber pressure of 250 psia (172.3 newtons/square centimeter). The mixture ratio is 4.2 to 1 and the area expansion ratio is 40:1. The engine delivers a specific impulse of 420 seconds (4118.8 newton-seconds/kg). An S-band omni antenna and a sun acquisition sensor are mounted on the outboard flat area on each ACS engine mount.

The IM shown has been configured for an unmanned geosynchronous mission. Area within the IM has been provided for additional equipment required for a manned mission. This allocated area is shaded in the drawing. In quadrant II an additional active thermal control unit is required for manned missions to provide extra cooling as well as life support operations. In addition, another electrical power module is required as shown in Section B-B of the Figure 2-22. Consequently the configuration shown is capable of handling the entire spectrum of unmanned and manned tug missions.

3.2.6 EOS CARGO BAY INTEGRATION STUDY (FIGURES 3-5 AND 3-6)

An important operational aspect of the overall tug program is the method employed to transport the basic vehicle to earth orbit. For purposes of this study it was assumed most likely that the tug vehicle would be transported in the EOS orbiter cargo bay. Fluctuations and changes within the EOS program concerning the exact method of supporting and deploying a payload as well as in the dimensions of the payload envelope precluded any definition of the method of support and deployment of a tug vehicle early in the tug study. At a later date methods of support and deployment as well as payload envelope definition were available and two integration study drawings were prepared.

The first drawing (Figure 3-5) was prepared at an early phase of the study and one of the tug vehicle concepts was selected to demonstrate the method of attachment and deployment. The vehicle was a 60,000 pound (27,215 kg) single stage tug with a crew module. This configuration is based on Concept 2 and the configuration shown is that for a manned low earth orbit mission. The cargo bay was assumed to employ a pivoting



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SHEET 2 OF 2

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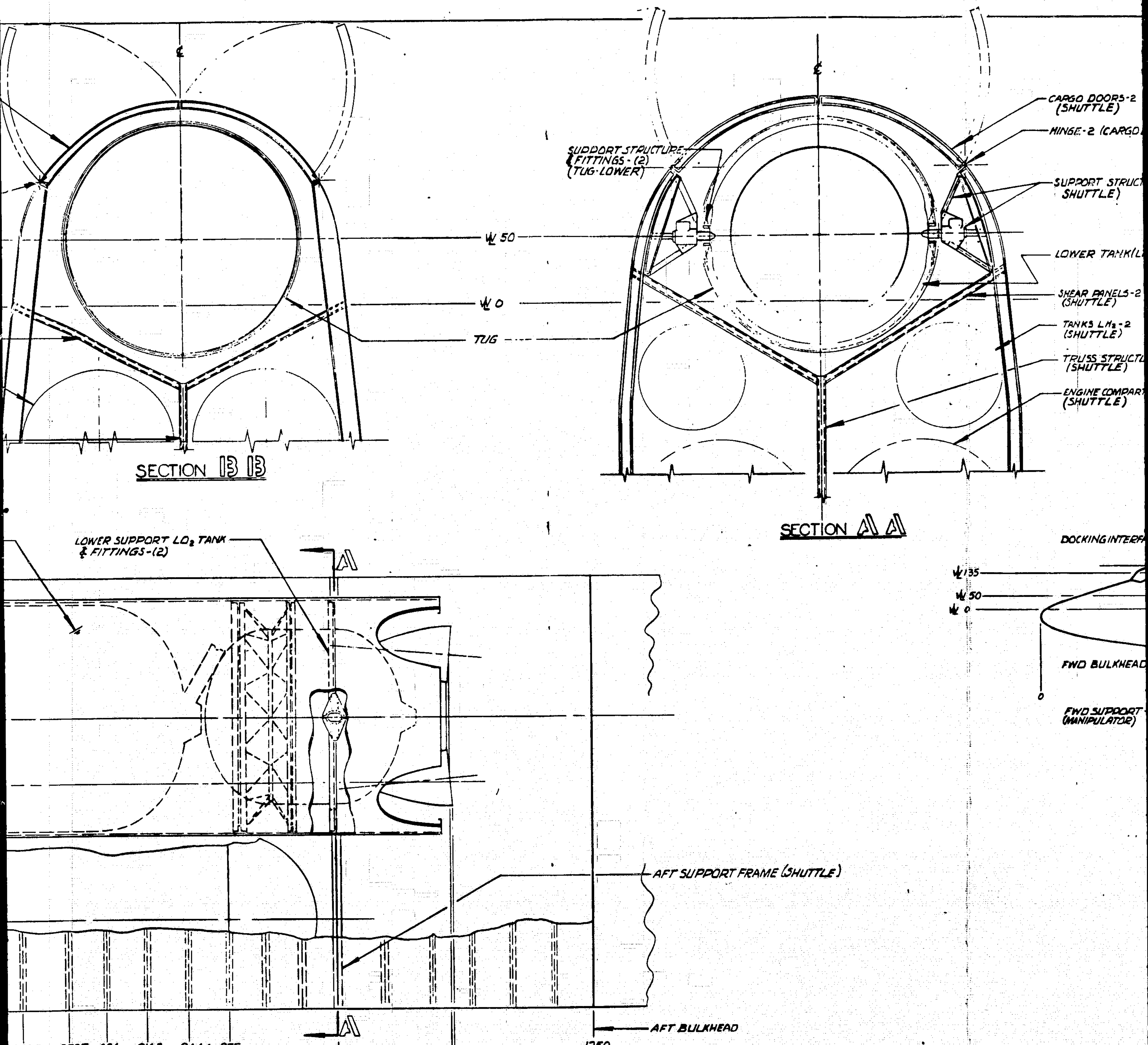


Figure 3-5. Tug/EOS C

FOLDOUT FRAME 2

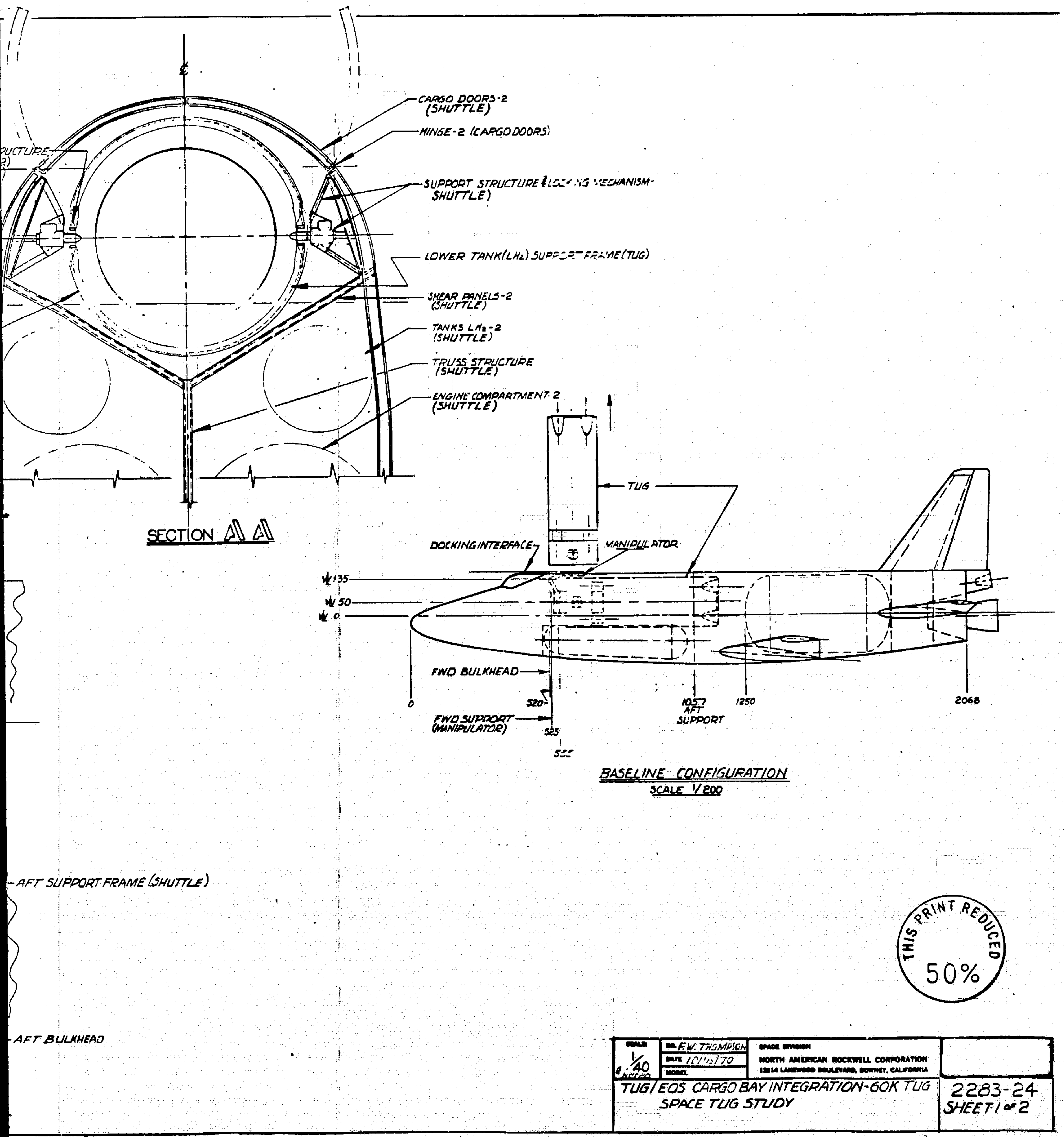
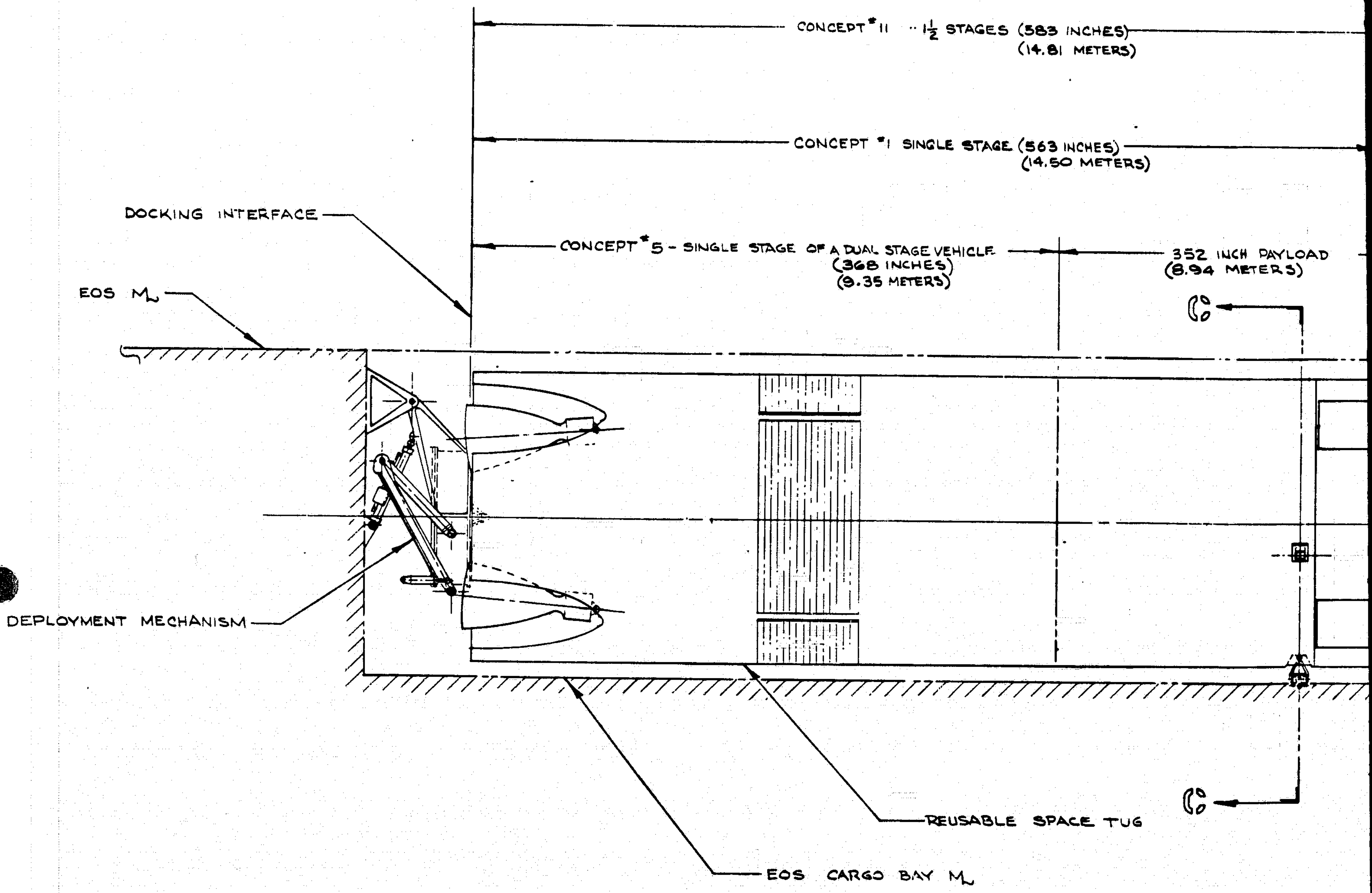


Figure 3-5. Tug/EOS Cargo Bay Integration, 60,000 Pound Tug

2

FOLDOUT FRAME 3



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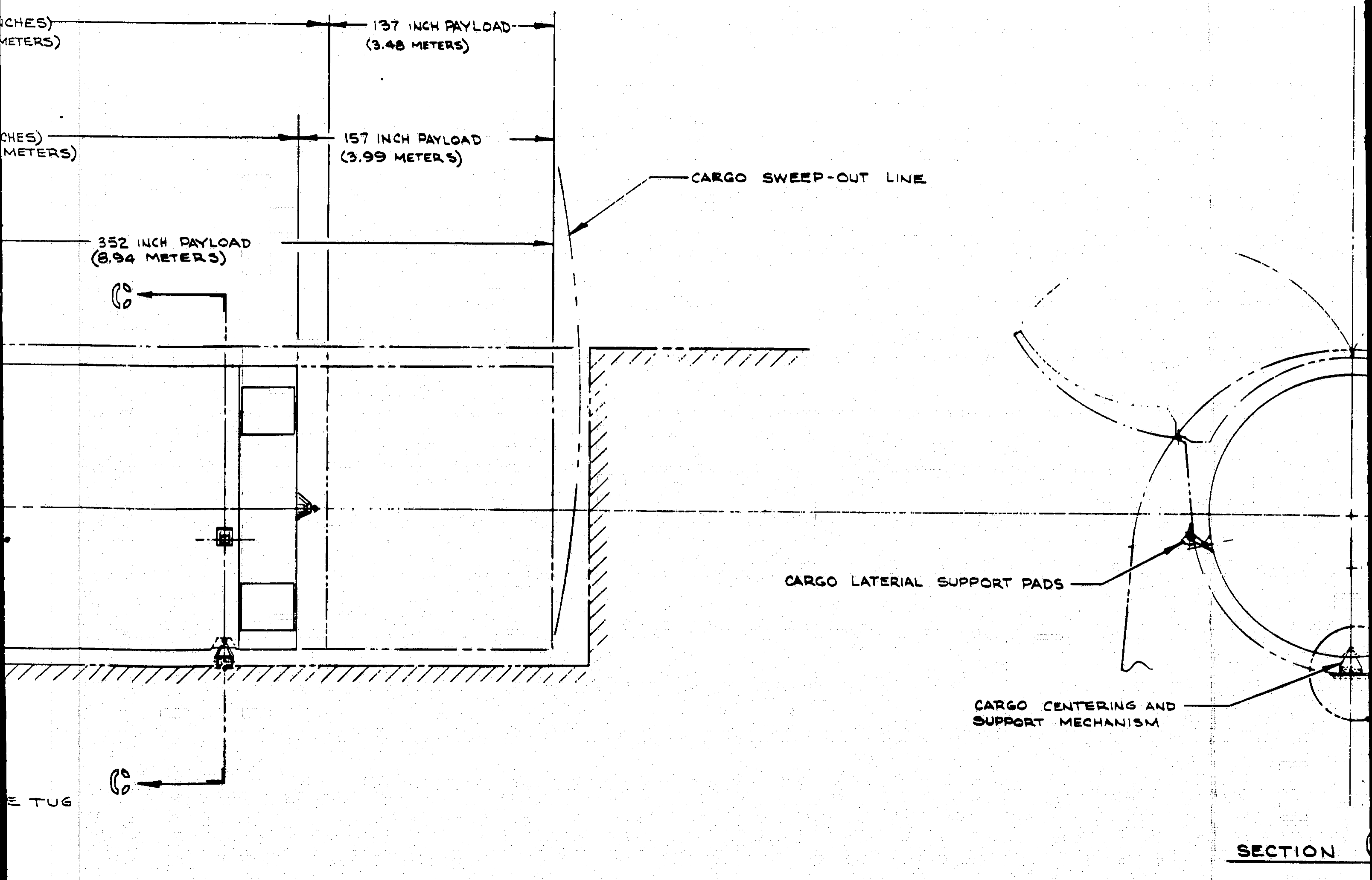

Figure 3-6. Reusa

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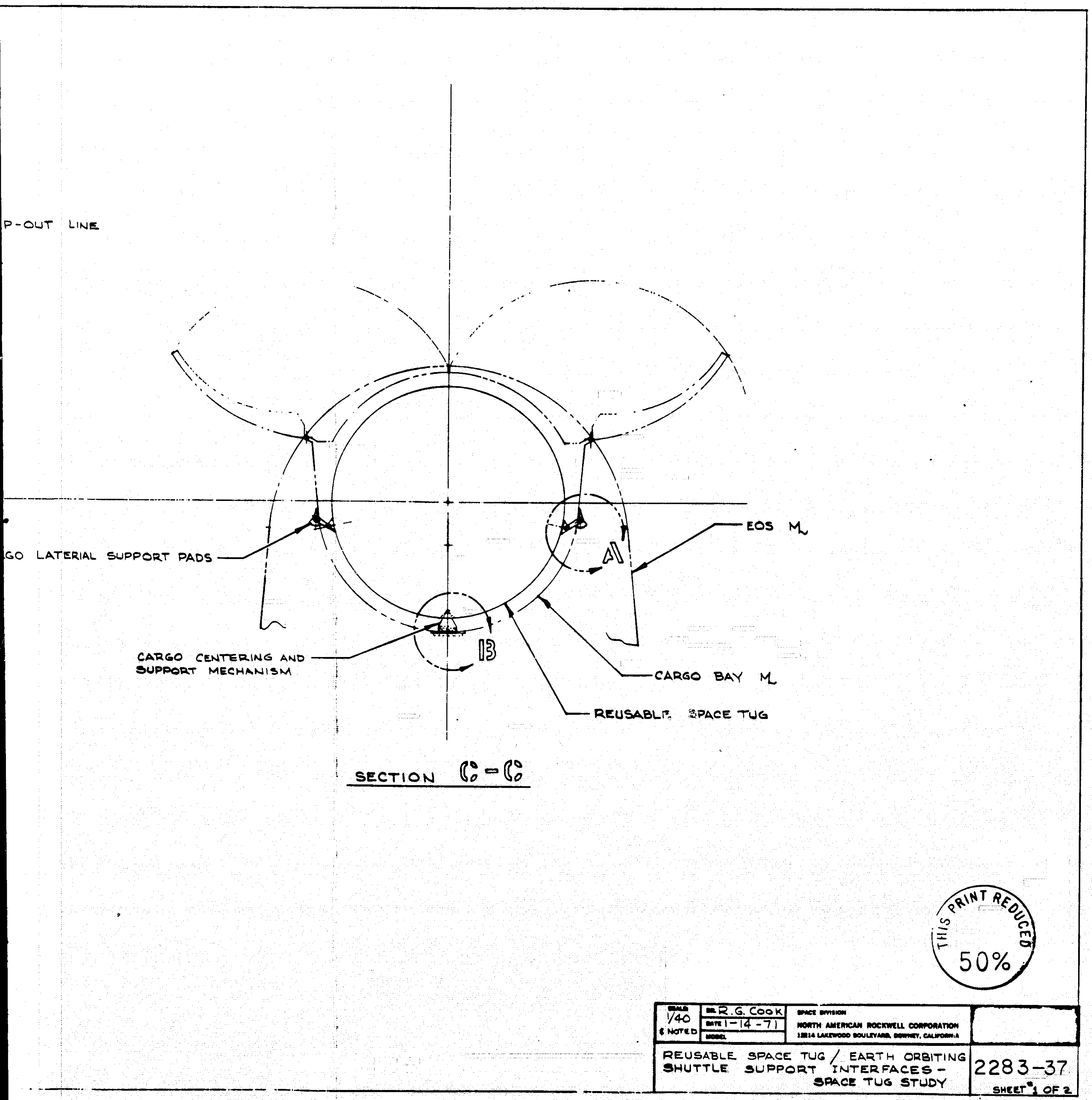
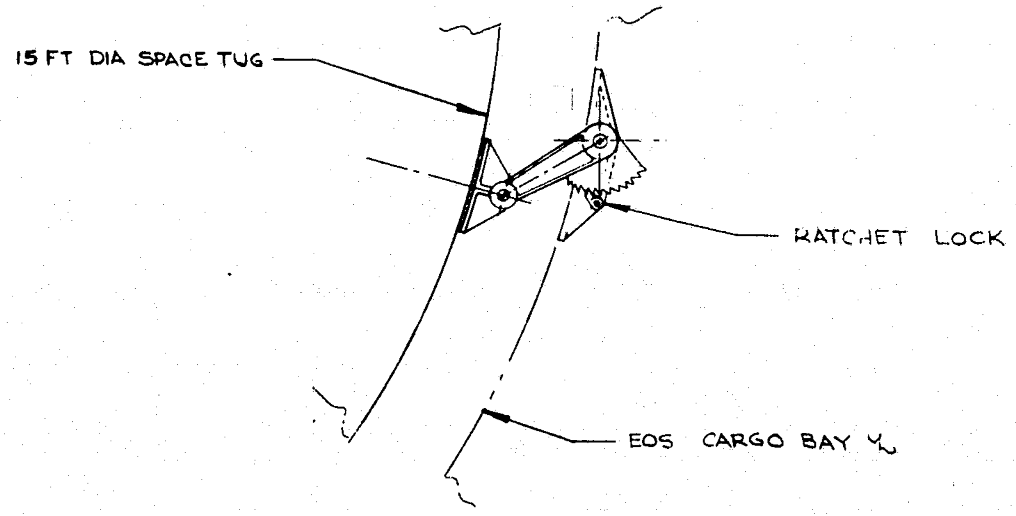
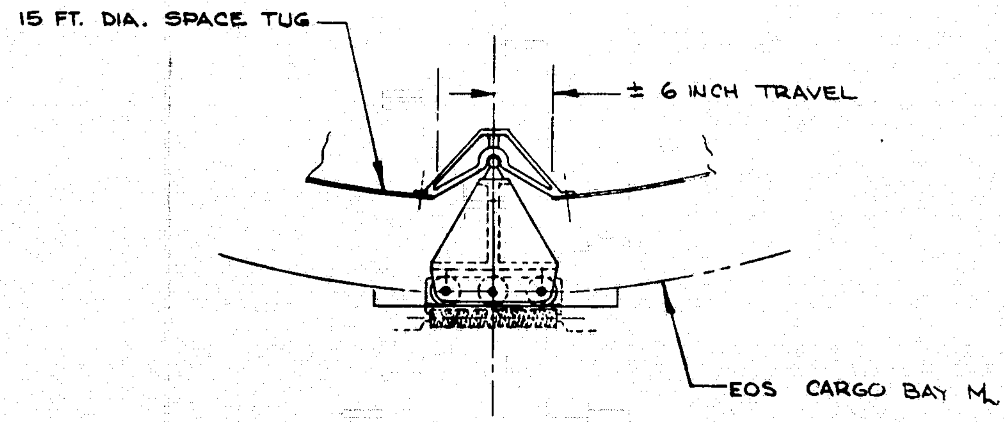


Figure 3-6. Reusable Space Tug, Earth Orbiting Shuttle Support Interfaces

2



DETAIL A
SCALE : 1/10
SHOWS LATERAL CARGO SUPPORT PAD



DETAIL B
SCALE : 1/10
SHOWS SPRING CENTERED ROLLER MOUNTED TUG SUPPORT

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Figure 3-6. Reusable Space Tug, Earth Orbiting Shuttle Support Interfaces

manipulator assembly which attached directly to the payload (tug) and supported it vertically in the cargo bay. All of the longitudinal launch loads are reacted by this manipulator assembly. Lateral and torsional loads are reacted by pin mechanisms located on the sides of the cargo bay at an aft support frame. The pins fit into receptacles in the tug. The pins are slotted longitudinally to allow for axial movement of the payload, and are retracted once in orbit for payload deployment.

In this configuration the tug is supported in a normally upright position when on the launch pad. The crew module is at the forward end of the vehicle and attaches through a neuter docking system to the manipulator assembly. This arrangement allows the tug crew to enter the tug crew module from the orbiter crew compartment through the manipulator assembly and neuter docking system. The vehicle arrangement in conjunction with the support and deployment system make an attractive concept with many desirable features (upright position, crew access, etc.). The most unattractive feature is the requirement for the crew module and intelligence module to support the propulsion module during launch or for the payloads to support the vehicle with no CM.

The tug vehicle configuration and EOS cargo bay configuration utilized on this drawing were preliminary and were refined during the later phase of the study. The second drawing (2283-37) was prepared as part of the design refinement and made use of more recent data concerning the manipulator assembly, lateral support concept, and payload envelope dimensions. The drawing depicts the single stage Concept 1 refined configuration for an unmanned geosynchronous mission and gives dimensions also for the Concept 5 and 11 refine configuration vehicles. The major difference between this concept and that shown on the previous drawing is the orientation of the tug vehicle within the cargo bay and the method of providing lateral restraint.

The earth-orbiting shuttle (EOS) supports the reusable tug during launch, space flight, and reentry. The EOS is equipped with a deployment system consisting of a radius arm and hinge, actuating mechanism, locking links and braces, and an umbilical between the EOS and the tug for servicing and checkout of the tug while it is being transported in the EOS.

The tug is secured to the EOS at the deployment system interface by means of an active docking probe and a matching passive drogue. The probe is part of the EOS deployment radius arm and the drogue is part of the tug. This mechanism is capable of supporting the tug axially during ±3 g loading caused by maximum acceleration, during deployment, and releasing of the tug in space, and the docking, retrieving, and stowage of the tug in the EOS cargo bay for space operations.

Other supports are required at the other end of the propulsion module to locate and support the tug in the cargo bay. A centering roller support system is used to capture the payload end of the propulsion module and lock it to the EOS cargo bay. This unit is capable of ±6 inches (.157 m) lateral motion, but is spring loaded to return to center. On the same EOS station plane as the centering and lock system are two adjustable cargo lateral support pads which are ratchet operated to remove excessive play. The cargo bay can house and deploy a cargo module 15 feet (4.57 m) in diameter and 60 feet (18.3 m) in length without interference to the EOS structure.

Each of the three baseline vehicle concepts developed during the design refinement phase are shown within the cargo bay payload envelope and the remaining length available for tug payload is also indicated. For the Concept 1 vehicle 157 inches (4 meters) remains for payload. The Concept 11 vehicle has 137 inches (3.5 meters) available and the Concept 5 vehicle can accommodate 352 inches (8.9 meters) of payload. In the case of the Concept 5 vehicle (two stages where both are recovered from an unmanned geosynchronous mission) only one stage can be accommodated within the EOS cargo bay payload envelope. Consequently, two EOS launches and in-space assembly of the two stages is required for the geosynchronous mission.

The EOS cargo deployment system described above provides for positive support of the tug within the bay and for deployment outside the bay for launch release into space. It also provides a hard docking port for tug upon return to EOS from a space mission, and for subsequent securing into the cargo bay for earth return. It should be noted that this EOS scheme is but one of several studied so far (but is the one which has received the most study at NR). Another concept (described earlier in the Design Approach section of this report) does not employ a hard docking port at all but instead utilizes manipulator arms or cherry-pickers. With this later concept, the tug would be supported for all loads during launch to orbit by means of retaining provisions along the length of the vehicle. Deployment would be by release of the retainers, extraction of tug by means of the cherry pickers, and release to space. Upon return to EOS from a mission, the tug would maneuver into close proximity with the EOS cargo bay ("soft docking") and the cherry pickers would be used to grasp the tug and insert it into the cargo bay for retention. Thus, a "soft docking" concept is used with the cherry picker scheme.

With the earlier hard docking deployment arm scheme, a docking port is necessary on the base of the tug. This requires a clear centerline on the base of the tug, which tends to preclude a single-engine configuration unless a special concentric docking adapter outside of the engine diameter were to be used. Since the latter appears undesirable from a total space program

compatibility standpoint, the baseline tug designs provided the centerline docking with four outboard engines (two also would be permissible). With the later cherry picker cargo deployment scheme, the centerline docking provision is not needed so that another constraint on tug engine arrangement is removed. It should be noted that the EOS hard docking scheme requires the tug to be mounted inverted within the bay on the launch pad, complicating somewhat the fill and venting provisions if the tug were to be ground fueled (ground-based).

3.3 BASIC CONCEPTS WEIGHTS

From the structural and equipment weight analyses presented in Section 2.2.7, the final iteration of propellant loading was computed for the three base configurations (Concepts 1, 5, and 11). All propellant capacities were derived for the equatorial geosynchronous missions. The base weight analyses were performed on a spacecraft designed to start from a 100 n mi (161 kilometer) orbit, deliver a 10,000-pound (4,536-kilogram) payload and return the spacecraft to a 100 n mi (161 kilometer) earth orbit. Summary weight statements of the three concepts for the geosynchronous missions are presented in Tables 3-1 through 3-3. The IM used is the unmanned design, the PM design used is the unmanned or space manned design.

Four summary weight statements for the lunar landing missions using components of the geosynchronous missions are presented in Tables 3-4 through 3-7. Several configurations of lunar landing can be obtained for each concept. Two concepts for Concept 1 are illustrated. The first consists of the addition of a CM on top of the IM/PM and cargo modules at the base. This results in a high center of gravity and therefore a rather large and heavy landing gear. The second configuration moves the crew module to the aft end of the IM/PM but necessitates the addition of swing-out-and-down PM engines. A net weight reduction of approximately 27 percent is noted for lowering the crew module. Concept 5 uses the geosynchronous mission PM's by adding the swing-out-and-down engines to one and removing the engines from the other (thus forming a tank set, TS). A gross weight approximately 5 percent larger than the equivalent design using Concept 1 components is noted. Concept 11 uses both the PM and TS sized for the geosynchronous mission. The configuration in which the CM is placed between the PM and TS was selected for weight illustration. The resultant spacecraft weight is approximately 1.6 percent lighter than Concept 1. The metric unit equivalents of these weight summaries are presented in Tables 3-8 through 3-14.

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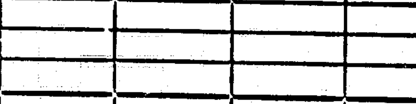
Table 3-1. Concept 1 - Geosynchronous Summary Weight Statement - English Units

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	520	3,015						
3.0	Induced envir prot	140	690						
4.0	Launch recov and docking	200	75						
5.0	Main propulsion	-	1,599						
6.0	Orient control sep and ull	358							
7.0	Prime power source	406							
8.0	Power conv and distr	100							
9.0	Guidance and navigation	276	17						
10.0	Instrumentation	101							
11.0	Communication	935							
12.0	Environmental control	184	29						
13.0	Growth allowance	160	270						
14.0	Personnel provisions								
15.0	Crew sta control and pan								
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	3,380	5,696	-					9,075
17.0	Personnel								
18.0	Cargo			10,000					10,000
19.0	Ordnance								
20.0	Ballast								
21.0	Resid prop. and serv items	15	975						990
	SUBTOTALS (INERT WEIGHT)	3,395	6,670	10,000					20,065
22.0	Res prop. and serv items								
23.0	Inflight losses		250						250
24.0	Thrust decay propellant		40						40
25.0	Full thrust propellant		77,975						77,975
26.0	Thrust prop. buildup		80						80
27.0	Pre-ignition losses								
	Aux propellant	45	595						640
	TOTALS (GROSS WEIGHT) (LB)	3,440	85,610	10,000					99,050
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:
CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004
ITEM OR MODULE
A IM
B PM
C PL
D
E
F
SPACECRAFT
M MANNED LAUNCH
U UNMANNED LAUNCH

NOTES & SKETCHES:

- Geosync equatorial mission
- Start from and return to 100 n mi
- Four engines, total thrust = 36,200 lb
- One LH_2 tank and 4 LOX tanks
- Apollo-type docking system



Concept No. 1 - Ge

FORM 3942-Z NEW 2-70

Table 3-2. Concept 5 - Geosynchronous Summary Weight Statement - English Units

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	520	2,070						
3.0	Induced envir prot	140	535						
4.0	Launch recov and docking	200	75						
5.0	Main propulsion	-	1,564						
6.0	Orient control sep and ull	358							
7.0	Prime power source	406							
8.0	Power conv and distr	100	17						
9.0	Guidance and navigation	276							
10.0	Instrumentation	101							
11.0	Communication	935							
12.0	Environmental control	184	29						
13.0	Growth allowance	160	215						
14.0	Personnel provisions								
15.0	Crew sta control and pan								
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	3,380	4,505	3,380	4,505	-			15,770
17.0	Personnel								
18.0	Cargo					10,000			10,000
19.0	Ordnance								
20.0	Ballast								
21.0	Resid prop. and serv items	15	870	15	870				1,770
	SUBTOTALS (INERT WEIGHT)	3,395	5,375	3,395	5,375	10,000			27,540
22.0	Res prop. and serv items								
23.0	Inflight losses		125		125				250
24.0	Thrust decay propellant		20		20				40
25.0	Full thrust propellant		41,210		41,210				82,420
26.0	Thrust prop. buildup		40		40				80
27.0	Pre-ignition losses								
	Aux propellant	45	105	45	495				690
	TOTALS (GROSS WEIGHT) (LB)	3,440	46,875	3,440	47,265	10,000			111,020
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE	
A	IM - No. 1
B	PM - No. 1
C	IM - No. 2
D	PM - No. 2
E	PL
F	

SPACECRAFT	
M	MANNED LAUNCH
U	UNMANNED LAUNCH

NOTES & SKETCHES:

- Geosync equatorial mission
- Start from and return to 100 n mi
- Four engines, total thrust = 38,200 lb/stage
- One LH_2 tank and 4 LOX tanks/stage
- Apollo type docking system

Concept No. 5 - Ge

FORM 3942-2 NEW 2-70

Table 3-3. Concept 11 - Geosynchronous Summary Weight Statement - English Units

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	520	915	2,380					
3.0	Induced envir prot	140	140	580					
4.0	Launch recov and docking	200	75	75					
5.0	Main propulsion	-	1,484	274					
6.0	Orient control sep and ull	358							
7.0	Prime power source	406		29					
8.0	Power conv and distr	100							
9.0	Guidance and navigation	276	17	17					
10.0	Instrumentation	101							
11.0	Communication	935							
12.0	Environmental control	184							
13.0	Growth allowance	160	130	165					
14.0	Personnel provisions								
15.0	Crew sta control and pan								
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	3,380	2,790	3,520	-				9,690
17.0	Personnel								
18.0	Cargo				10,000				10,000
19.0	Ordnance								
20.0	Ballast	15	220	715					950
21.0	Resid prop. and serv items	3,395	3,010	4,235	10,000				20,640
	SUBTOTALS (INERT WEIGHT)								
22.0	Res prop. and serv items								
23.0	Inflight losses		125	125					250
24.0	Thrust decay propellant		20	20					40
25.0	Full thrust propellant		10,980	52,375					63,355
26.0	Thrust prop. buildup		40	40					80
27.0	Pre-ignition losses								
	Aux propellant	45	100	495					640
	TOTALS (GROSS WEIGHT) (LB)	3,440	14,275	57,290	10,000				85,005
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

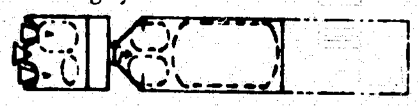
- A IM
- B PM
- C TS
- D PL
- E
- F

SPACECRAFT

- M MANNED LAUNCH
- U UNMANNED LAUNCH

NOTES & SKETCHES:

- Geosync equatorial mission
- Start from and return to 100 n mi
- Four engines on PM, total thrust = 29,300 lb
- Two LOX and 2 LH_2 tanks on PM
- Four LOX and 1 LH_2 tank on T/S
- Apollo docking system



Concept No. 11- Ge

FORM 3942-Z NEW 2-70

Table 3-4. Concept 1 - Lunar Landing A, Summary Weight Statement - English Units

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	520	3,015	2,535					
3.0	Induced envir prot	140	690	340					
4.0	Launch recov and docking	-	75	480	5,200				
5.0	Main propulsion	-	1,599						
6.0	Orient control sep and ull	358							
7.0	Prime power source	861							
8.0	Power conv and distr	100		50					
9.0	Guidance and navigation	410	17	210					
10.0	Instrumentation	-							
11.0	Communication	935		300					
12.0	Environmental control	762	69	190					
13.0	Growth allowance	204	270	605					
14.0	Personnel provisions			1,835					
15.0	Crew sta control and pan			155					
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	4,290	5,735	6,700	5,200	-			21,925
17.0	Personnel Crew			800					800
18.0	Cargo Food, etc.			1,620		10,000			11,620
19.0	Ordnance N_2 and Tk			20					20
20.0	Ballast EVA			360					360
21.0	Resid prop. and serv items	15	975						990
	SUBTOTALS (INERT WEIGHT)	4,305	6,710	9,500	5,200	10,000			35,715
22.0	Res prop. and serv items								
23.0	Inflight losses		250						250
24.0	Thrust decay propellant		40						40
25.0	Full thrust propellant		61,680						61,680
26.0	Thrust prop. buildup		80						80
27.0	Pre-ignition losses								
	Aux propellant	45	3,880	-					3,925
	TOTALS (GROSS WEIGHT) (LB)	4,350	72,640	9,500	5,200	10,000			101,690
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

- A IM
- B PM
- C CM
- D LLG
- E PL
- F

SPACECRAFT

- M MANNED LAUNCH
- U UNMANNED LAUNCH

NOTES & SKETCHES:

Lunar landing mission - NASA Mode A
Four men/3 + 28 + 14 days
PM from concept 1-G
CM top mounted
Landing CG ≈ 26.8 feet from ground

Concept No. 1 - LLAe

FORM 3942-Z NEW 2-70

Table 3-5. Concept 1 - Lunar Landing B, Summary Weight Statement - English Units

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	520	3,115	2,535					
3.0	Induced envir prot	140	690	340					
4.0	Launch recov and docking	-	75	480	3,200				
5.0	Main propulsion	-	2,449						
6.0	Orient control sep and ull	358							
7.0	Prime power source 861	861							
8.0	Power conv and distr	100		50					
9.0	Guidance and navigation	410	17	210					
10.0	Instrumentation								
11.0	Communication	935		300					
12.0	Environmental control	762	69	190					
13.0	Growth allowance	204	320	605					
14.0	Personnel provisions			1,835					
15.0	Crew sta control and pan			55					
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	4,290	6,735	6,700	3,200	-			20,925
17.0	Personnel Crew			800					800
18.0	Cargo Food, etc.			1,620		10,000			11,620
19.0	Ordnance N_2 and Tk			20					20
20.0	Ballast EVA			360					360
21.0	Resid prop. and serv items	15	975						990
	SUBTOTALS (INERT WEIGHT)	4,305	7,710	9,500	3,200	10,000			34,715
22.0	Res prop. and serv items								
23.0	Inflight losses		250						250
24.0	Thrust decay propellant		40						40
25.0	Full thrust propellant		60,110						60,110
26.0	Thrust prop. buildup		80						80
27.0	Pre-ignition losses								
	Aux propellant	45	3,880	-					3,925
	TOTALS (GROSS WEIGHT) (LB)	4,350	72,070	9,500	3,200	10,000			99,120
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

A	IM
B	PM
C	CM
D	LLG
E	PL
F	

SPACECRAFT

M	MANNED LAUNCH
U	UNMANNED LAUNCH

NOTES & SKETCHES:

- Lunar landing mission - NASA Mode A
- Four men/3 + 28 + 14 days
- PM from concept 3-G with swing-out and down engines
- CM bottom mounted
- Landing CG ≈ 19.5 feet from ground

Concept No. 1-LLBe

FORM 3942-Z NEW 2-70

Table 3-6. Concept 5 - Lunar Landing Summary Weight Statement - English Units

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	520	2,170	2,070	2,535				
3.0	Induced envir prot	140	535	535	340				
4.0	Launch recov and docking	-	75	75	480	3,200			
5.0	Main propulsion	-	2,414	364					
6.0	Orient control sep and ull	358							
7.0	Prime power source	861							
8.0	Power conv and distr	100			50				
9.0	Guidance and navigation	410	17	17	210				
10.0	Instrumentation								
11.0	Communication	935			300				
12.0	Environmental control	762	69	69	190				
13.0	Growth allowance	204	265	155	605				
14.0	Personnel provisions				1,835				
15.0	Crew sta control and pan				155				
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	4,290	5,545	3,285	6,700	3,200	-		23,020
17.0	Personnel Crew				800				800
18.0	Cargo Food, etc.				1,620		10,000		11,620
19.0	Ordnance N_2 and Tk				20				20
20.0	Ballast EVA				360				360
21.0	Resid prop. and serv items	15	870	870					1,755
	SUBTOTALS (INERT WEIGHT)	4,305	6,415	4,155	9,500	3,200	10,000		37,585
22.0	Res prop. and serv items								
23.0	Inflight losses		25	225					250
24.0	Thrust decay propellant		20	20					40
25.0	Full thrust propellant		41,500	22,030					63,530
26.0	Thrust prop. buildup		40	40					80
27.0	Pre-ignition losses								
	Aux propellant	45	400	3,480	-				3,925
	TOTALS (GROSS WEIGHT) (LB)	4,350	48,400	29,950	9,500	3,200	10,000		105,400
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

A	IM
B	PM
C	TS
D	CM
E	LLG
F	PL

SPACECRAFT

M	MANNED LAUNCH
U	UNMANNED LAUNCH

NOTES & SKETCHES:

- Lunar landing mission - NASA Mode A
- Four men/3 + 28 + 14 days
- PM from concept 5-G with swing-out and down engines
- TS from concept 5-G less engines
- CM bottom mounted
- Landing cg ≈ 19.5 feet from ground

Concept No. 5-LLe

FORM 3942-Z NEW 2-70

Table 3-7. Concept 11 - Lunar Landing Summary Weight Statement - English Units

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	520	915	2,380	2,535				
3.0	Induced envir prot	140	140	580	340				
4.0	Launch recov and docking	-	75	75	480	3,200			
5.0	Main propulsion	-	1,484	274					
6.0	Orient control sep and ull	358							
7.0	Prime power source	861	-						
8.0	Power conv and distr	100		46	50				
9.0	Guidance and navigation	410	17	-	210				
10.0	Instrumentation	-							
11.0	Communication	935			300				
12.0	Environmental control	762	69	-	190				
13.0	Growth allowance	204	130	165	605				
14.0	Personnel provisions				1,835				
15.0	Crew sta control and pan				155				
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	4,290	2,830	3,520	6,700	3,200	-		20,540
17.0	Personnel Crew				800				800
18.0	Cargo Food, etc.				1,620		10,000		11,620
19.0	Ordnance N_2 and Tk				20				20
20.0	Ballast EVA				360				360
21.0	Resid prop. and serv items	15	220	715					950
	SUBTOTALS (INERT WEIGHT)	4,305	3,050	4,235	9,500	3,200	10,000		34,290
22.0	Res prop. and serv items								
23.0	Inflight losses		25	225					250
24.0	Thrust decay propellant		20	20					40
25.0	Full thrust propellant		10,780	48,150					58,930
26.0	Thrust prop. buildup		40	40					80
27.0	Pre-ignition losses								
	Aux propellant	45	400	3,480	-				3,925
	TOTALS (GROSS WEIGHT) (LB)	4,350	14,315	56,150	9,500	3,200	10,000		97,515
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:
CODE, SYSTEM REF. MIL-M-38310A OR SP-6004
ITEM OR MODULE
- A IM
- B PM
- C TS
- D CM
- E LLG
- F PL

SPACECRAFT
- M MANNED LAUNCH
- U UNMANNED LAUNCH

NOTES & SKETCHES:

Lunar landing mission - NASA Mode A
Four men/3 + 28 + 14 days
PM and TS from concept 11-G
CM mounted between PM and TS
Landing cg ≈ 19.5 feet from ground

Concept 11-LLe

FORM 3942-Z NEW 2-70

Table 3-8. Concept 1 - Geosynchronous Summary Weight Statement - Metric Units

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	236	1,368						
3.0	Induced envir prot	64	313						
4.0	Launch recov and docking	91	34						
5.0	Main propulsion		725						
6.0	Orient control sep and ull	162							
7.0	Prime power source	184							
8.0	Power conv and distr	45							
9.0	Guidance and navigation	125	8						
10.0	Instrumentation	46							
11.0	Communication	424							
12.0	Environmental control	83	13						
13.0	Growth allowance	73	123						
14.0	Personnel provisions								
15.0	Crew sta control and pan								
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	1,533	2,584	-					4,117
17.0	Personnel								
18.0	Cargo			4,536					4,536
19.0	Ordnance								
20.0	Ballast								
21.0	Resid prop. and serv items	7	442						449
	SUBTOTALS (INERT WEIGHT)	1,540	3,026	4,536					9,102
22.0	Res prop. and serv items								
23.0	Inflight losses		113						113
24.0	Thrust decay propellant		18						18
25.0	Full thrust propellant		35,369						35,369
26.0	Thrust prop. buildup		36						36
27.0	Pre-ignition losses								
	Aux propellant	20	270						290
	TOTALS (GROSS WEIGHT)(KG)	1,560	38,832	4,536					44,928
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:

CODE, SYSTEM, REF, MIL-M-38310A OR SP-6004

ITEM OR MODULE

- A IM
- B PM
- C PL
- D
- E
- F

SPACECRAFT

- M MANNED LAUNCH
- U UNMANNED LAUNCH

NOTES & SKETCHES:

Concept 1-Gm

FORM 3942-Z NEW 2-70

Table 3-9. Concept 5 - Geosynchronous Summary Weight Statement - Metric Units

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	236	939						
3.0	Induced envir prot	64	243						
4.0	Launch recov and docking	91	34						
5.0	Main propulsion		709						
6.0	Orient control sep and ull	162							
7.0	Prime power source	184							
8.0	Power conv and distr	45							
9.0	Guidance and navigation	125	8						
10.0	Instrumentation	46							
11.0	Communication	424							
12.0	Environmental control	83	13						
13.0	Growth allowance	73	98						
14.0	Personnel provisions								
15.0	Crew sta control and pan								
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	1,533	2,044	1,533	2,044	-			7,154
17.0	Personnel								
18.0	Cargo					4,536			4,536
19.0	Ordnance								
20.0	Ballast								
21.0	Resid prop. and serv items	7	395	7	395	-			804
	SUBTOTALS (INERT WEIGHT)	1,540	2,439	1,540	2,439	4,536			12,494
22.0	Res prop. and serv items								
23.0	Inflight losses		57		57				114
24.0	Thrust decay propellant		9		9				18
25.0	Full thrust propellant		18,693		18,693				37,386
26.0	Thrust prop. buildup		18		18				36
27.0	Pre-ignition losses	20	48	20	225				313
	Aux propellant								
	TOTALS (GROSS WEIGHT)(KG)	1,560	21,264	1,560	21,441	4,536			50,361
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:	
CODE, SYSTEM REF, MIL-M-38310A OR SP-6004	
ITEM OR MODULE	
A	IM - NO. 1
B	PM - NO. 1
C	IM - NO. 2
D	PM - NO. 2
E	PL
F	
SPACECRAFT	
M	MANNED LAUNCH
U	UNMANNED LAUNCH

NOTES & SKETCHES:

Concept 5-Gm

FORM 3942-Z NEW 2-70

Table 3-10. Concept 11 - Geosynchronous Summary Weight Statement - Metric Units

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	236	415	1,080					
3.0	Induced envir prot	64	64	263					
4.0	Launch recov and docking	91	34	34					
5.0	Main propulsion		672	124					
6.0	Orient control sep and ull	162							
7.0	Prime power source	184							
8.0	Power conv and distr	45		21					
9.0	Guidance and navigation	125	8						
10.0	Instrumentation	46							
11.0	Communication	424							
12.0	Environmental control	83	13						
13.0	Growth allowance	73	59	75					
14.0	Personnel provisions								
15.0	Crew sta control and pan								
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	1,533	1,265	1,597	-				4,395
17.0	Personnel								
18.0	Cargo				4,536				4,536
19.0	Ordnance								
20.0	Ballast								
21.0	Resid prop. and serv items	7	100	324					431
	SUBTOTALS (INERT WEIGHT)	1,540	1,365	1,921	4,536				9,362
22.0	Res prop. and serv items								
23.0	Inflight losses		57	57					114
24.0	Thrust decay propellant		9	9					18
25.0	Full thrust propellant		4,981	23,757					28,738
26.0	Thrust prop. buildup		18	18					36
27.0	Pre-ignition losses	20	45	225					290
	Aux propellant								
	TOTALS (GROSS WEIGHT) (KG)	1,560	6,475	25,987	4,536				38,558
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:	
CODE, SYSTEM, REF, MIL-M-38310A OR SP-6004	
ITEM OR MODULE	
A	IM
B	PM
C	TS
D	PL
E	
F	
SPACECRAFT	
M	MANNED LAUNCH
U	UNMANNED LAUNCH

NOTES & SKETCHES:

Concept 11-Gm

FORM 3942-Z NEW 2-70

Table 3-11. Concept 1 - Lunar Landing A, Summary Weight Statement - Metric Units

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	236	1,368	1,150					
3.0	Induced envir prot	64	313	154					
4.0	Launch recov and docking		34	218	2,359				
5.0	Main propulsion		725						
6.0	Orient control sep and ull	162							
7.0	Prime power source	391							
8.0	Power conv and distr	45		23					
9.0	Guidance and navigation	186	8	95					
10.0	Instrumentation								
11.0	Communication	424		136					
12.0	Environmental control	346	31	86					
13.0	Growth allowance	92	123	274					
14.0	Personnel provisions			832					
15.0	Crew sta control and pan			70					
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	1,946	2,602	3,038	2,359	-			9,945
17.0	Personnel Crew			363					363
18.0	Cargo Food, etc.			735		4,536			5,271
19.0	Ordnance N_2 and Tk			9					9
20.0	Ballast EVA			163					163
21.0	Resid prop. and serv items	7	442						449
	SUBTOTALS (INERT WEIGHT)	1,953	3,044	4,308	2,359	4,536			16,200
22.0	Res prop. and serv items								
23.0	Inflight losses		113						113
24.0	Thrust decay propellant		18						18
25.0	Full thrust propellant		27,978						27,978
26.0	Thrust prop. buildup		36						36
27.0	Pre-ignition losses								
	Aux propellant	20	1,760						1,780
	TOTALS (GROSS WEIGHT)(KG)	1,973	32,949	4,308	2,359	4,536			46,125
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:	
CODE, SYSTEM; REF. MIL-M-38310A OR SP-6004	
ITEM OR MODULE	
A	IM
B	PM
C	CM
D	LLG
E	PL
F	
SPACECRAFT	
M	MANNED LAUNCH
U	UNMANNED LAUNCH

NOTES & SKETCHES:

Concept 1-LLAm

FORM 3942-Z NEW 2-70

Table 3-12. Concept 1 - Lunar Landing B, Summary Weight Statement - Metric Units

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	236	1,413	1,150					
3.0	Induced envir prot	64	313	154					
4.0	Launch recov and docking		34	218	1,452				
5.0	Main propulsion		1,111						
6.0	Orient control sep and ull	162							
7.0	Prime power source	391							
8.0	Power conv and distr	45		23					
9.0	Guidance and navigation	186	8	95					
10.0	Instrumentation								
11.0	Communication	424		136					
12.0	Environmental control	346	31	86					
13.0	Growth allowance	92	145	274					
14.0	Personnel provisions			832					
15.0	Crew sta control and pan			70					
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	1,946	3,055	3,038	1,452	-			9,491
17.0	Personnel Crew			363					363
18.0	Cargo Food, etc.			735		4,536			5,271
19.0	Ordnance N_2 and Tk			9					9
20.0	Ballast EVA			163					163
21.0	Resid prop. and serv items	7	442						449
	SUBTOTALS (INERT WEIGHT)	1,953	3,497	4,308	1,452	4,536			15,746
22.0	Res prop. and serv items								
23.0	Inflight losses		113						113
24.0	Thrust decay propellant		18						18
25.0	Full thrust propellant		27,267						27,267
26.0	Thrust prop. buildup		36						36
27.0	Pre-ignition losses								
	Aux propellant	20	1,760						1,780
	TOTALS (GROSS WEIGHT)(KG)	1,973	32,691	4,308	1,452	4,536			44,960
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:	
CODE, SYSTEM, REF, MIL-M-38310A OR SP-6004	
ITEM OR MODULE	
A	LM
B	PM
C	CM
D	LLG
E	PL
F	
SPACECRAFT	
M	MANNED LAUNCH
U	UNMANNED LAUNCH

NOTES & SKETCHES:

Concept 1-LLBm

FORM 3942-Z NEW 2-70

Space Division
North American Rockwell

Table 3-13. Concept 5, Lunar Landing Summary Weight Statement - Metric Units

CODE	SYSTEM	ITEM OR MODULE						SPACECRAFT	
		A	B	C	D	E	F	M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	236	984	939	1,150				
3.0	Induced envir prot	64	243	243	154				
4.0	Launch recov and docking		34	34	218	1,452			
5.0	Main propulsion		1,095	165					
6.0	Orient control sep and ull	162							
7.0	Prime power source	391							
8.0	Power conv and distr	45			23				
9.0	Guidance and navigation	136	8	8	95				
10.0	Instrumentation								
11.0	Communication	424			136				
12.0	Environmental control	346	31	31	86				
13.0	Growth allowance	92	120	70	274				
14.0	Personnel provisions				832				
15.0	Crew sta control and pan				70				
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	1,946	2,515	1,490	3,038	1,452	-		10,441
17.0	Personnel Crew				363				363
18.0	Cargo Food, etc,				735		4,536		5,271
19.0	Ordnance N_2 and Tk				9				9
20.0	Ballast EVA				163				163
21.0	Resid prop. and serv items	7	395	395					797
	SUBTOTALS (INERT WEIGHT)	**1,953**	**2,910**	**1,885**	**4,308**	**1,452**	**4,536**		**17,044**
22.0	Res prop. and serv items								
23.0	Inflight losses		**11**	**102**					**113**
24.0	Thrust decay propellant		9	9					**18**
25.0	Full thrust propellant		18,824	9,993					28,817
26.0	Thrust prop. buildup		18	18					36
27.0	Pre-ignition losses								
	Aux propellant	20	181	1,579					1,780
	TOTALS (GROSS WEIGHT) (KG)	1,973	21,953	13,586	4,308	1,452	4,536		47,808
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:	
CODE, SYSTEM, REF. MIL-M 38310A OR SP-6004	
ITEM OR MODULE	
A	IM
B	PM
C	TS
D	CM
E	LLG
F	PL
SPACECRAFT	
M	MANNED LAUNCH
U	UNMANNED LAUNCH

NOTES & SKETCHES:

Concept 5-LLm

FORM 3942-Z NEW 2-70

Table 3-14. Concept 11 - Lunar Landing Summary Weight Statement - Metric Units

CODE	SYSTEM	ITEM OR MODULE A	B	C	D	E	F	SPACECRAFT M	TOT
1.0	Aerodynamic surfaces								
2.0	Body structure	236	415	1,080	1,150				
3.0	Induced envir prot	64	64	263	154				
4.0	Launch recov and docking		34	34	218	1,452			
5.0	Main propulsion		672	124					
6.0	Orient control sep and ull	162							
7.0	Prime power source	391							
8.0	Power conv and distr	45		21	23				
9.0	Guidance and navigation	186	8		95				
10.0	Instrumentation								
11.0	Communication	424			136				
12.0	Environmental control	346	31		86				
13.0	Growth allowance	92	59	75	274				
14.0	Personnel provisions				832				
15.0	Crew sta control and pan				70				
16.0	Range safety and abort								
	SUBTOTALS (DRY WEIGHT)	1,946	1,283	1,597	3,038	1,452	-		9,316
17.0	Personnel Crew				363				363
18.0	Cargo Food, etc.				735		4,536		5,271
19.0	Ordnance N_2 and Tk				9				9
20.0	Ballast EVA				163				163
21.0	Resid prop. and serv items	7	100	324					431
	SUBTOTALS (INERT WEIGHT)	1,953	1,383	1,921	4,308	1,452	4,536		15,553
22.0	Res prop. and serv items								
23.0	Inflight losses		11	102					113
24.0	Thrust decay propellant		9	9					18
25.0	Full thrust propellant		4,890	21,841					26,731
26.0	Thrust prop. buildup		18	18					36
27.0	Pre-ignition losses								
	Aux propellant	20	181	1,579					1,780
	TOTALS (GROSS WEIGHT)(KG)	1,973	6,492	25,470	4,308	1,452	4,536		44,231
	Design envelope volume (ft^3)								
	Pressurized volume (ft^3)								
	Design envel surf area (ft^3)								
	Pressurized surf area (ft^2)								
	Design q, max (lb/ft^2)								
	Design g, max								
	Design power, max (kw)								
	Design no. men/days								

DESIGNATIONS:

CODE, SYSTEM: REF. MIL-M-38310A OR SP-6004

ITEM OR MODULE

A	IM
B	PM
C	TS
D	CM
E	LLG
F	PL

SPACECRAFT

M	MANNED LAUNCH
U	UNMANNED LAUNCH

NOTES & SKETCHES:

Concept 11-LLm

FORM 3942-Z NEW 2-70

3.4 SPECIAL STUDIES

3.4.1 HIGH PERFORMANCE COMPROMISE CONCEPTS

Simplified Operational Concepts

Background and Objectives

In developing baseline Concepts 1, 5, and 11, it became apparent that the features of modular, multipurpose, flexibility for the autonomous space-based reusable space tug imposed difficult gross weights and dimensions in terms of current EOS capabilities. The question then became, "what is the price being paid for these desirable operational features in terms of concept weights and geometry"? A special parallel study was therefore conducted to evaluate the effects of operational compromises which could lead to lighter, smaller designs. The compromises considered are summarized as follows with a brief explanation of their justification:

1. Integrated IM: Assumes that a separate IM module structure is not necessary

2. Reduced autonomy: Assuming that complete autonomy is not required nor cost-effective

3. Reduced systems redundancy: Permitted by ground-based operations with excellent surveillance between missions

4. Deletion of base docking: Assumes that a soft docking approach may be taken for EOS cargo

5. Reduced engine redundancy: Use of 1 or 2 engines; based on no docking, no lunar compromises

6. Overall improvement of base structural design: Permitted by deleting docking, 2 (or 1) engines, single O_2 tank, no landing gear provisions

As a result of this study it was expected that the single stage recoverable concept (Concept 1-A) might require 60,000 pounds (27,200 kilogram) to perform the geosynchronous mission and therefore might insure EOS compatibility with ground-based operations.

Initial Design Study

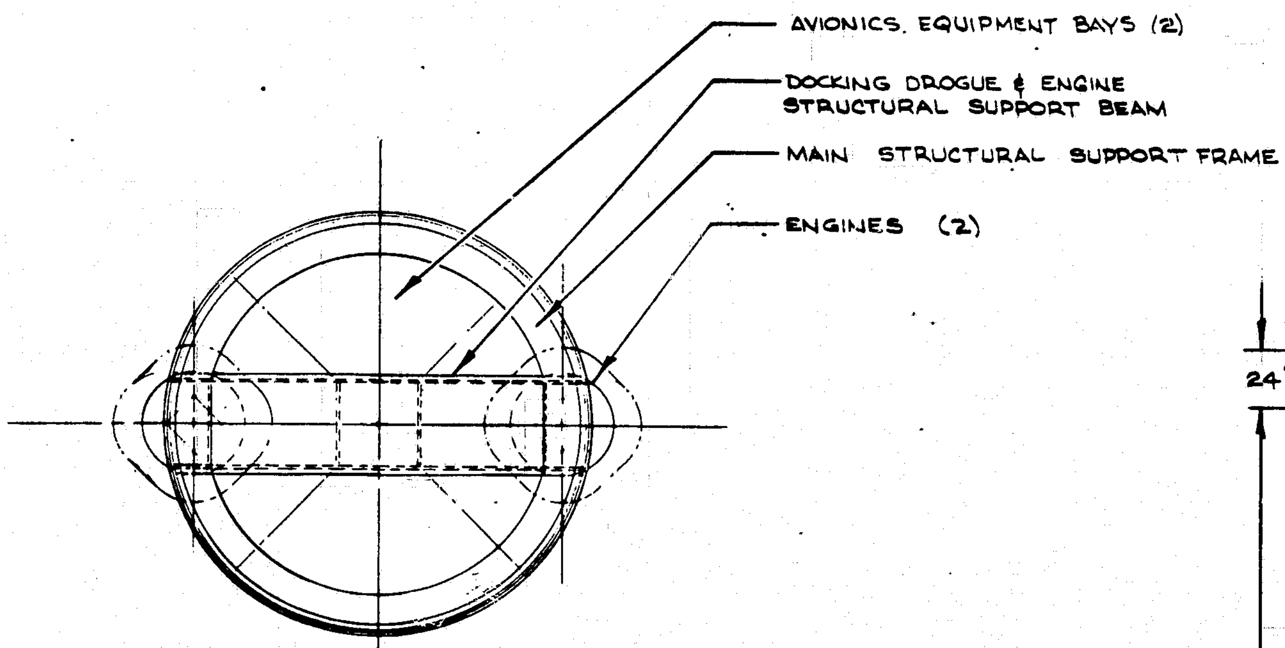
The first drawing prepared in this study (Figure 3-7) investigated the differences in using one, two, and six LO_2 tanks with dual and single engines. The first configuration (A) is an unmanned single stage geosynchronous vehicle. The general arrangement of this vehicle is based on an initially assumed 60, 000 pound (27, 200 kilogram) propellant requirement to perform the mission. The vehicle structure incorporates the IM structure to obtain the minimum vehicular weight. The propellant tanks are not an integral part of the basic vehicle structure.

The propellant capacity is based on 60, 000 pounds (27, 200 kilograms) of usable propellant for the main propulsion engines, 500 pounds (227 kilograms) for attitude control engines, 150 pounds (68 kilograms) for the electrical power system, and the normal residual and ullage allowances. The propellant tank volumes and sizes are as follows: The liquid hydrogen tank has an internal volume of 2306 cubic feet (65. 26 cubic meters), an inside diameter of 163 inches (4. 14 meters), and an internal length of 230 inches (5. 842 meters). The single liquid oxygen tank has an internal volume of 798 cubic feet (22. 58 cubic meters), an inside diameter of 154 inches (3. 91 meters), and an inside length of 110 inches (2. 99 meters). The propellant tanks are made of aluminum, and suspended from titanium support skirts with low thermal conductance sway braces at the opposite end. Further weight saving could result from use of glass-epoxy skirts or even boron-epoxy. The LH_2 tank shape has a cylindrical center section with 1. 4 to 1 elliptical end heads. The LO_2 tank shape is a 1. 4 to 1 ellipsoid.

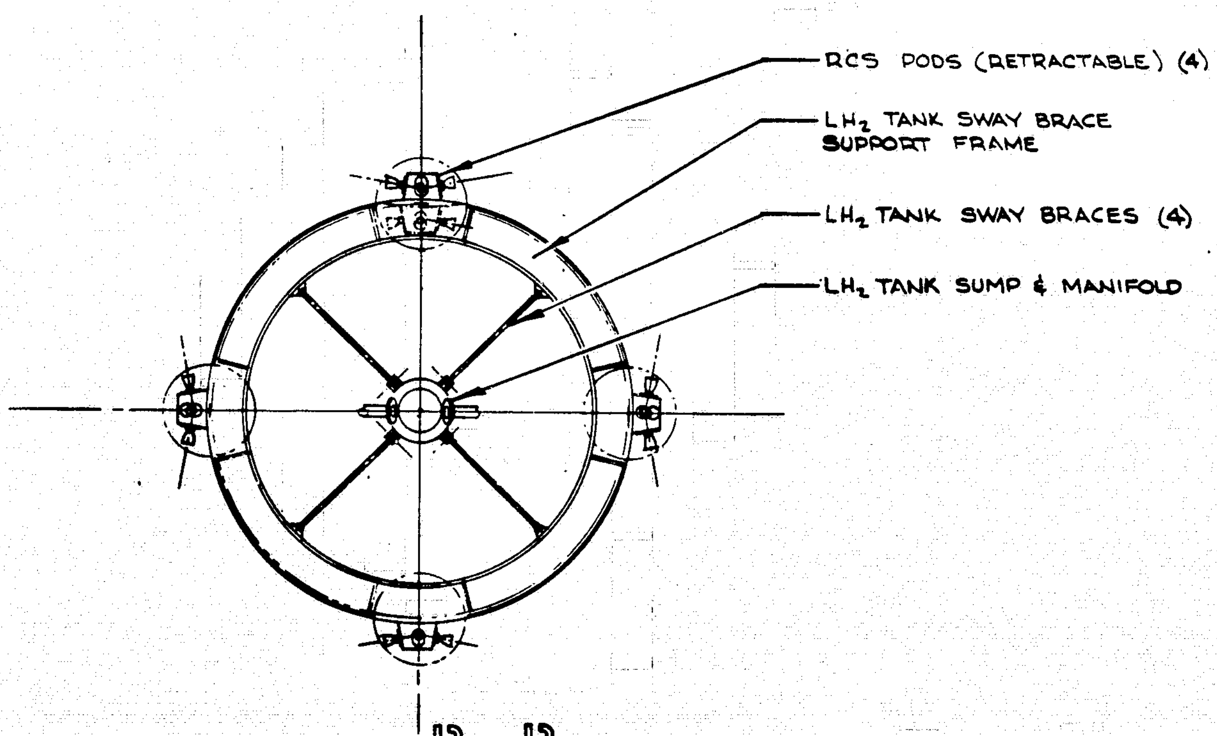
The basic structure of the vehicle is alluminum skin stringer cylindrical structure with a closeout bulkhead at the top, a docking thrust cone, LH_2 tank support ring frame, ACS and avionics support frame, the liquid oxygen tank support frame, and the docking cone and engine support beam.

The avionics equipment will be located in various areas of the vehicle. The communications, and guidance and navigation systems with antennae are located in the upper structure between the closeout bulkhead and the docking thrust cone. The data management avionics, the electrical power systems, and the attitude control system are located between the LH_2 and the LO_2 tanks. The attitude control pods, which have five nozzles per pod, are retracted inside the vehicle mold line during stowage in the EOS; retraction prevents interference with the EOS cargo bay structure.

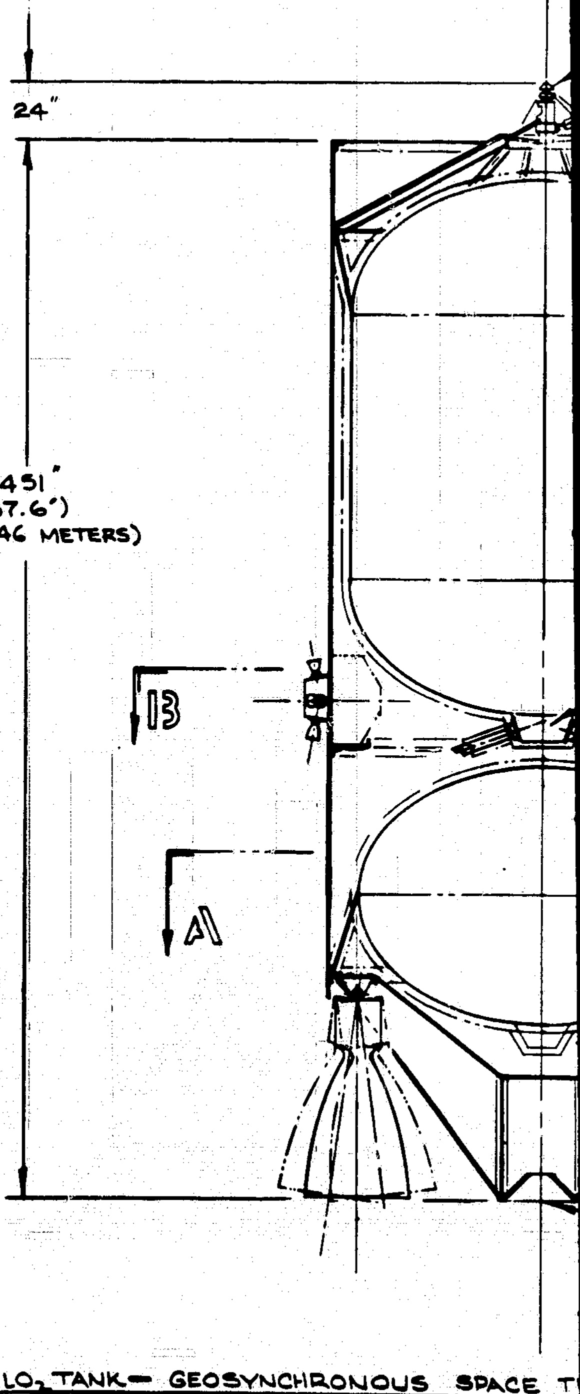
The two engines are mounted on each end of the engine-docking drogue support beam (docking is still possible with 2 engines). The engines gimbal inboard during stowage due to the 15-foot maximum diameter stowage envelope. The passive docking drogue, which is similar to the Apollo system,



SECTION A-A



SECTION B-B



SINGLE STAGE - DUAL ENGINES - SINGLE LO_2 TANK - GEOSYNCHRONOUS SPACE T

(60,000 LBS. USABLE MAIN PROPELLANT)

(27,216 KILOGRAMS)

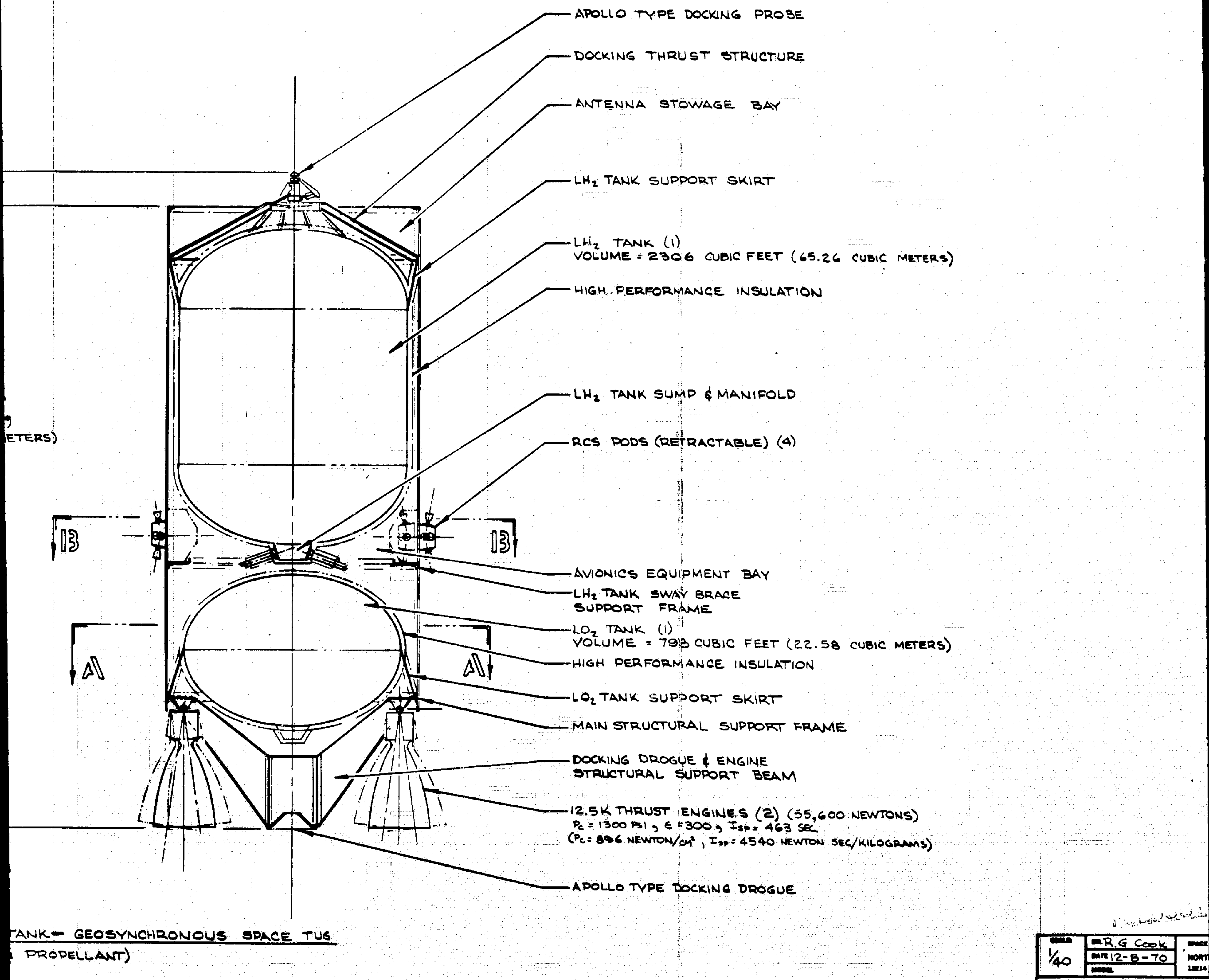


Figure 3-7. Geosynchronous Mission, 60,000 Pounds
Stage Recoverable

FOLDOUT FRAME 2

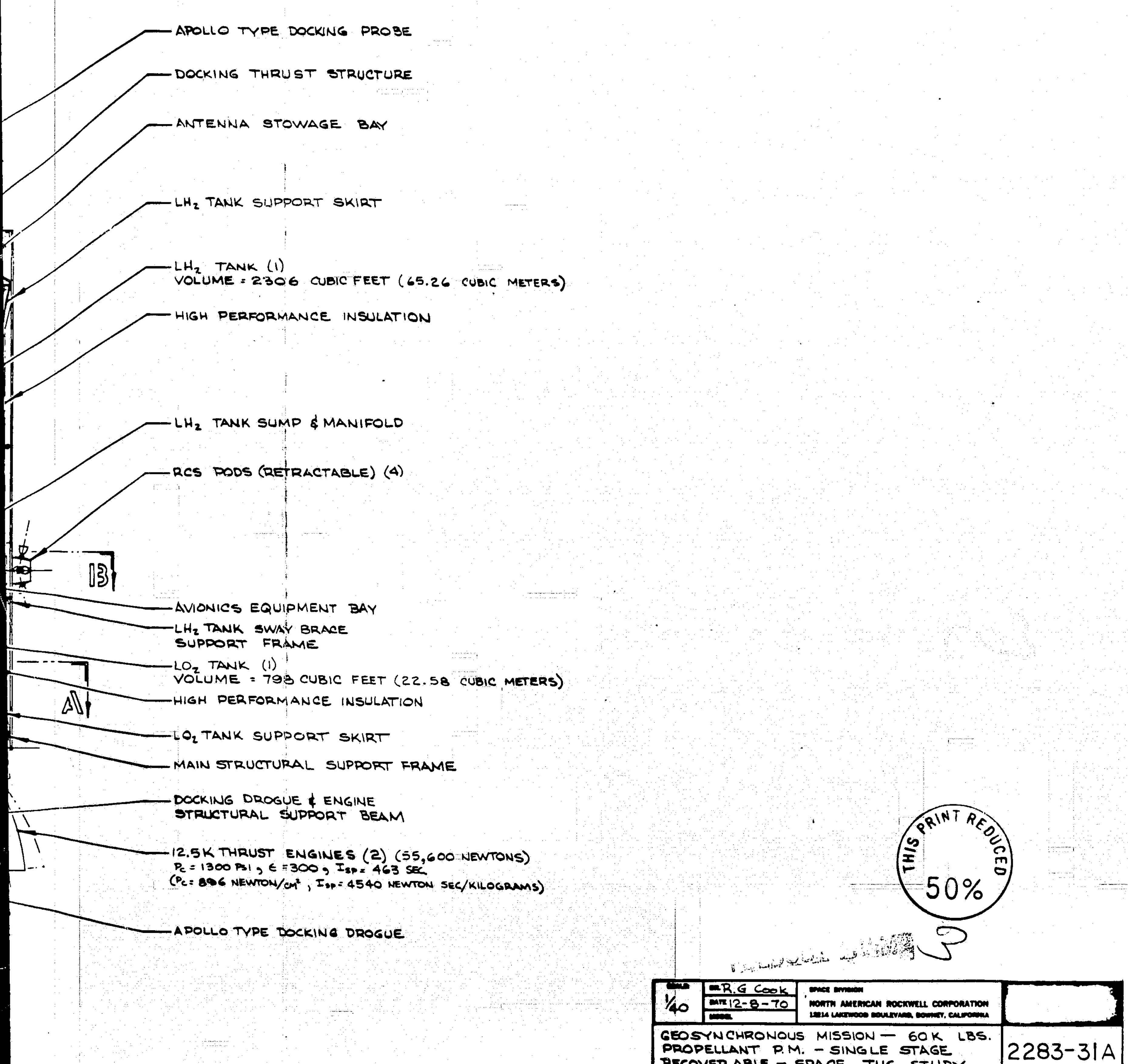
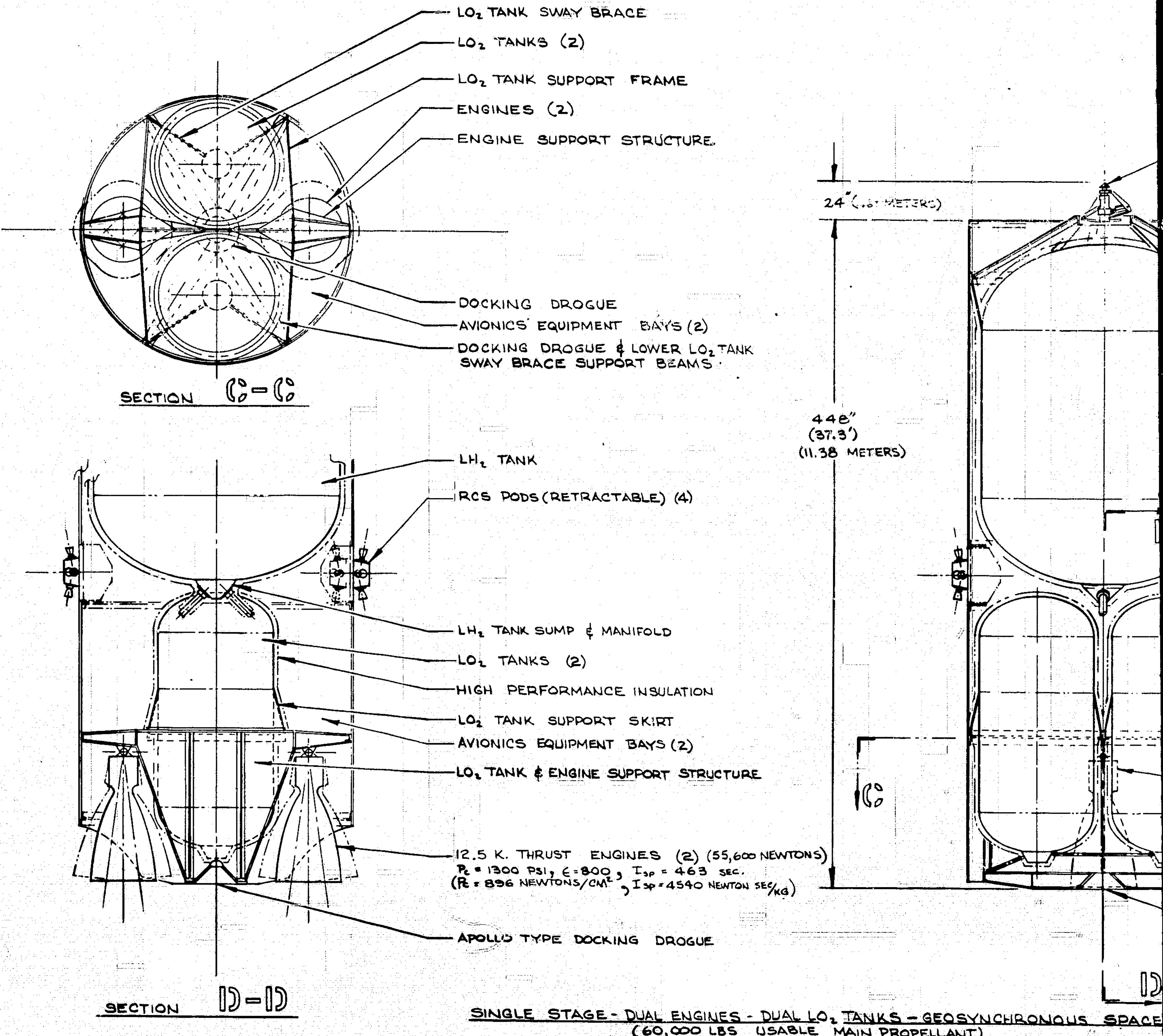


Figure 3-7. Geosynchronous Mission, 60,000 Pounds Propellant PM, Single Stage Recoverable

1 OF 3



SINGLE STAGE - DUAL ENGINES - DUAL LO_2 TANKS - GEOSYNCHRONOUS SPACE
(60,000 LBS USABLE MAIN PROPELLANT)
(27,160 KILOGRAMS)

Figure 3-7.

FOLDOUT FRAME 1

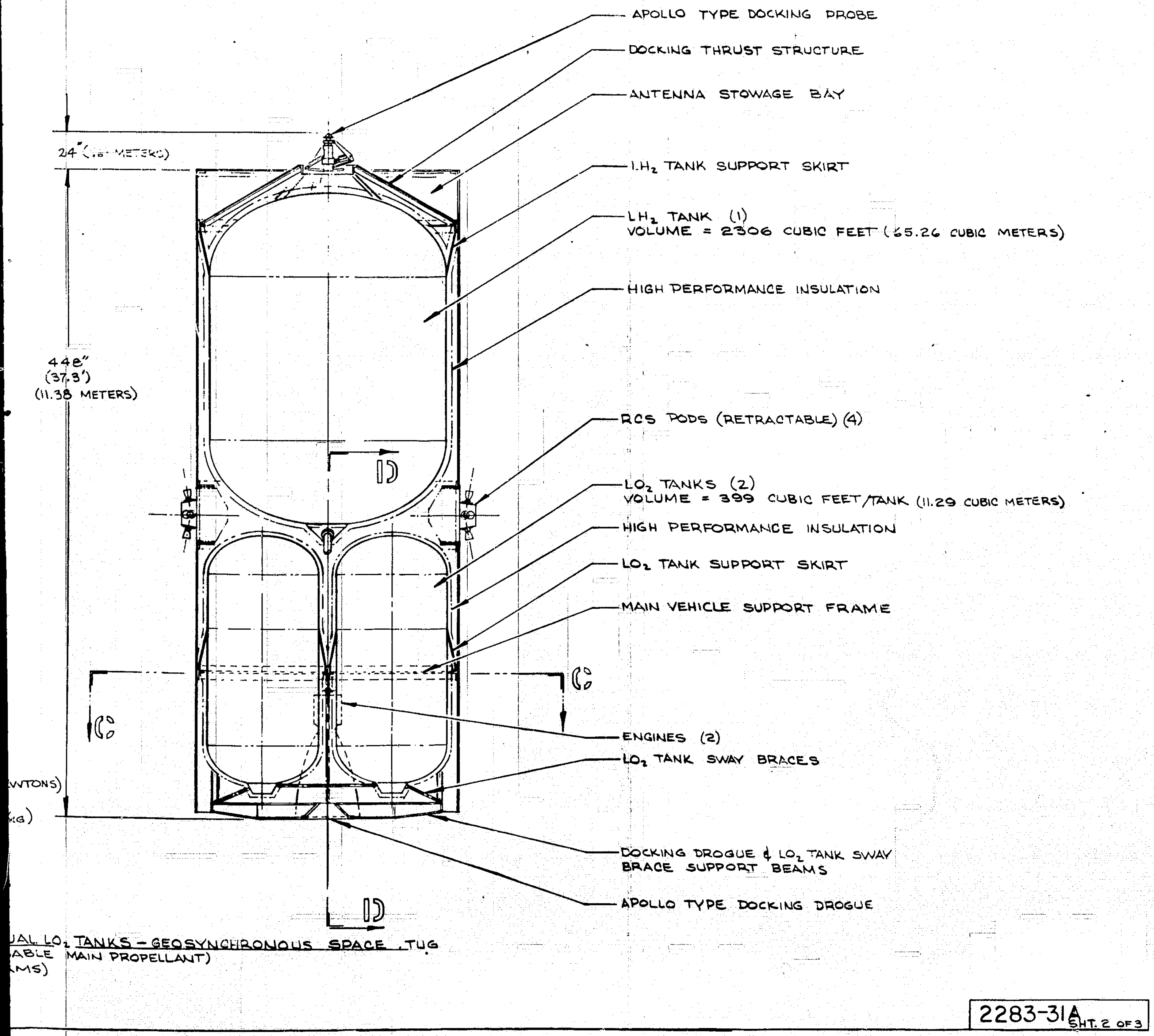


Figure 3-7. Geosynchronous Mission, 60,000 Pounds Propellant PM, Single Stage Recoverable

2 OF 3

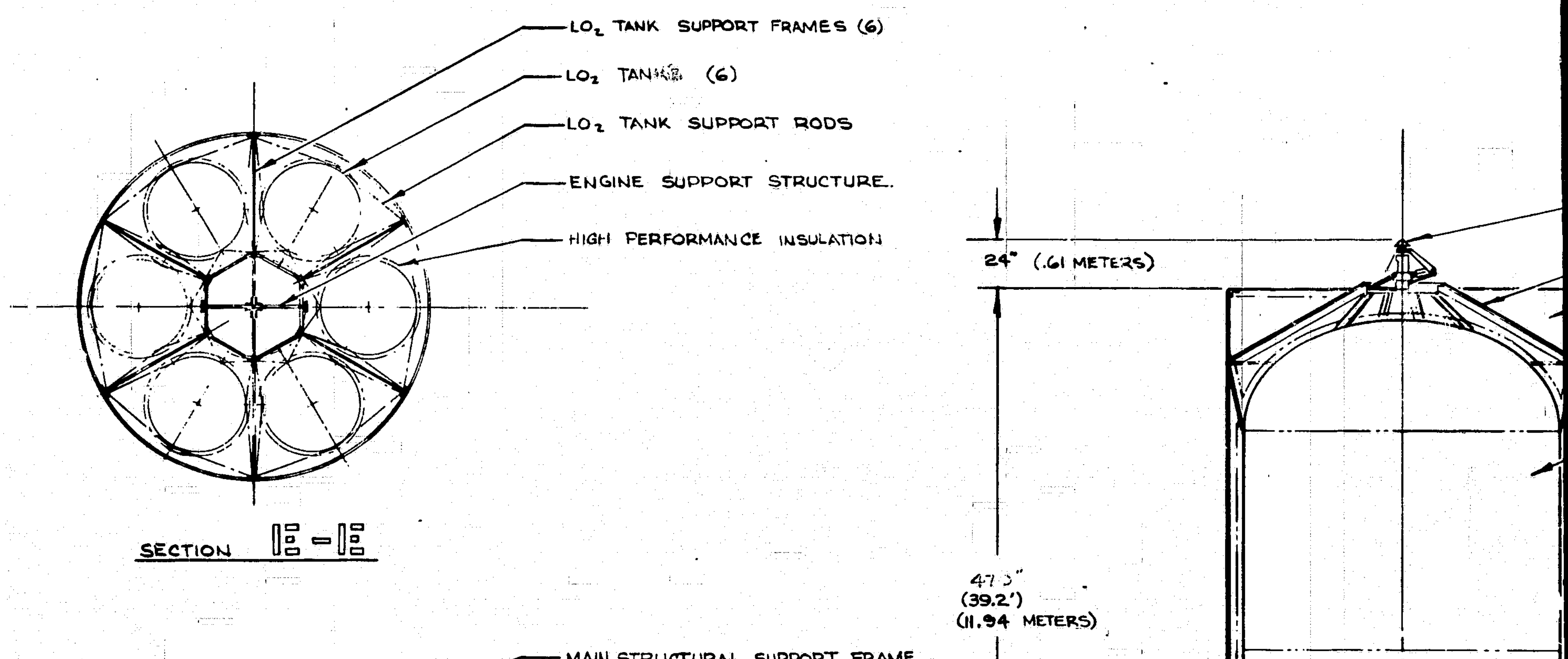


Figure 3-7. Geosynchronous Mi
Stage Recoverable

FOLDOUT FRAME /

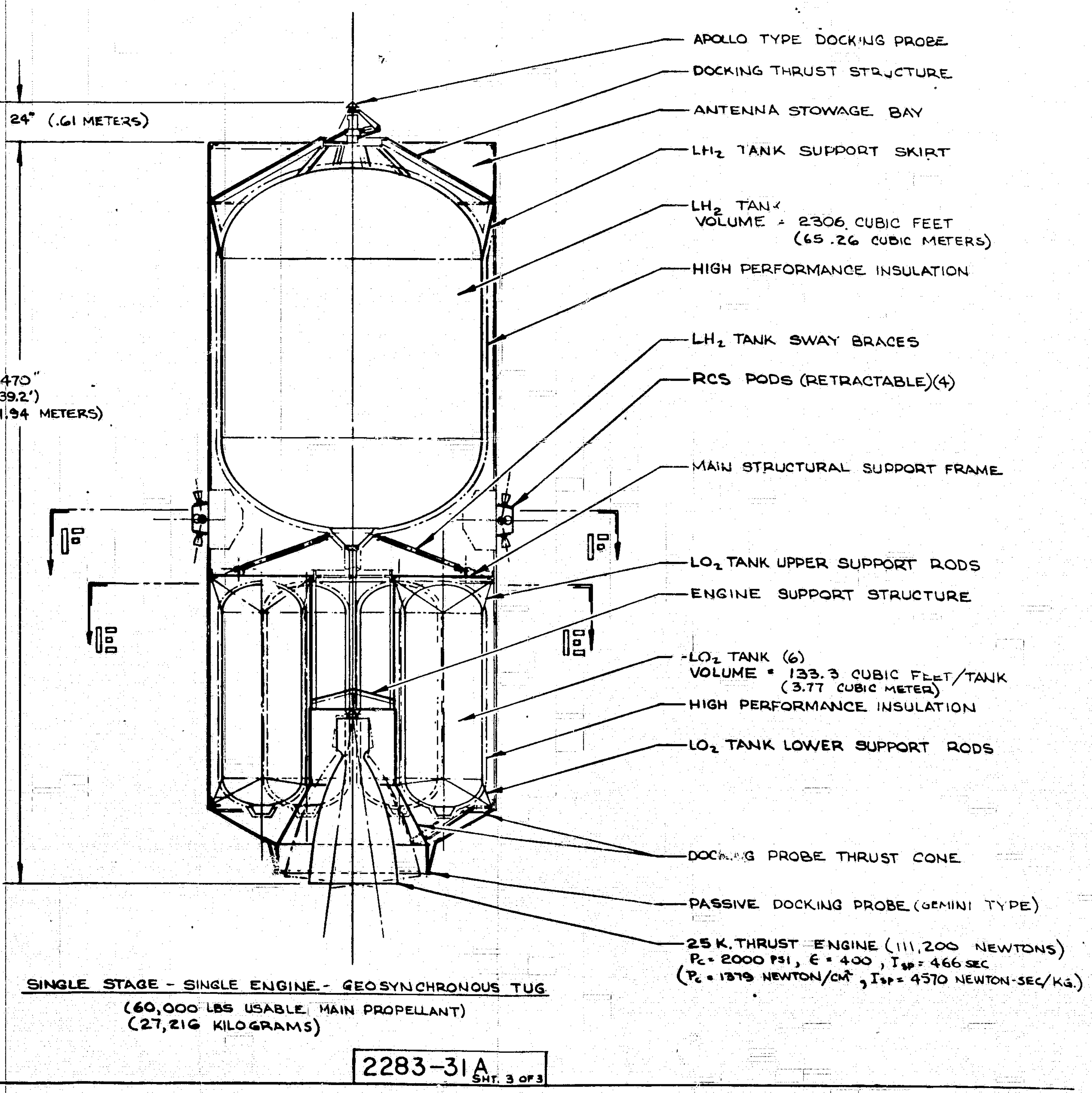


Figure 3-7. Geosynchronous Mission, 60,000 Pounds Propellant PM, Single Stage Recoverable

is mounted on the engine-docking system support beam on the aft center line of the vehicle.

Each main engine has a 12,500 pound (55,600 newton) thrust rating, 1300 psi (896 newton/square centimeters) chamber pressure, an expansion ratio of 400 to 1, a specific impulse of 463 seconds (4540 newton seconds per kilogram) and a mixture ratio of 6 to 1. The total thrust of the main engines is equal to 25,000 pounds (111,200 newtons).

The overall dimensions of the configuration are 15 feet (4.57 meters) in diameter by 451 inches (11.46 meters) long, which allows for a maximum payload length of 269 inches (6.83 meters).

The second configuration shown in Figure 3-7 is based on the same parameters as the first configuration. The only difference between configurations A and B are the liquid oxygen tanks. In configuration B there are two tanks to permit increased volumetric efficiency of the lower portion of the vehicle, and a reduced overall length. Each liquid oxygen tank has an internal volume of 399 cubic feet (11.29 cubic meters), an inside diameter of 76 inches (1.93 meters) and an internal length of 172 inches (4.37 meters). These tanks are aluminum, supported by titanium skirts which are attached to the main vehicle support frame located in the center of the tanks. This frame is also the main engine support structure, and incorporates a box section extending to the aft bulkhead for supporting the passive Apollo type docking drogue.

The avionics equipment, electrical power systems, and the attitude control tanks and equipment are located on each side in the bays above the engines. The engines are the same as those used in the A configuration. The overall dimensions of the B configuration are 15 feet (4.57 meters) in diameter and 448 inches (11.38 meters) in length.

The third configuration, C, is based on the same parameters as configuration A. The major differences are: six liquid oxygen tanks, a single engine, and a large diameter engine-concentric passive docking probe. The six tanks permit "swallowing" the single engine to yield a short overall vehicle length.

The liquid oxygen tanks have an internal volume of 133 cubic feet (3.77 cubic meters) each, an inside diameter of 48 inches (1.22 meters), and an internal length of 139 inches (3.53 meters). These tanks are supported by a multiple rod suspension system attached to each end of the tanks and to radial frames located between each tank.

The engine with 25,000 pounds (111,200 newtons) thrust, chamber pressure of 2000 psi (1379 newton per square centimeter), expansion ratio of 400 to 1, and a specific impulse of 466 seconds (4570 newton seconds per

kilogram), is nested in between the liquid oxygen tanks in a shroud, on the center line of the vehicle. The large docking probe, which is approximately 100 inches (2.54 meters) in diameter, is a Gemini type. The large diameter is required to permit sufficient clearance around the engine for normal operation. The overall dimension of the C configuration is 15 feet (4.57 meters) in diameter and 470 inches (11.94 meters) long.

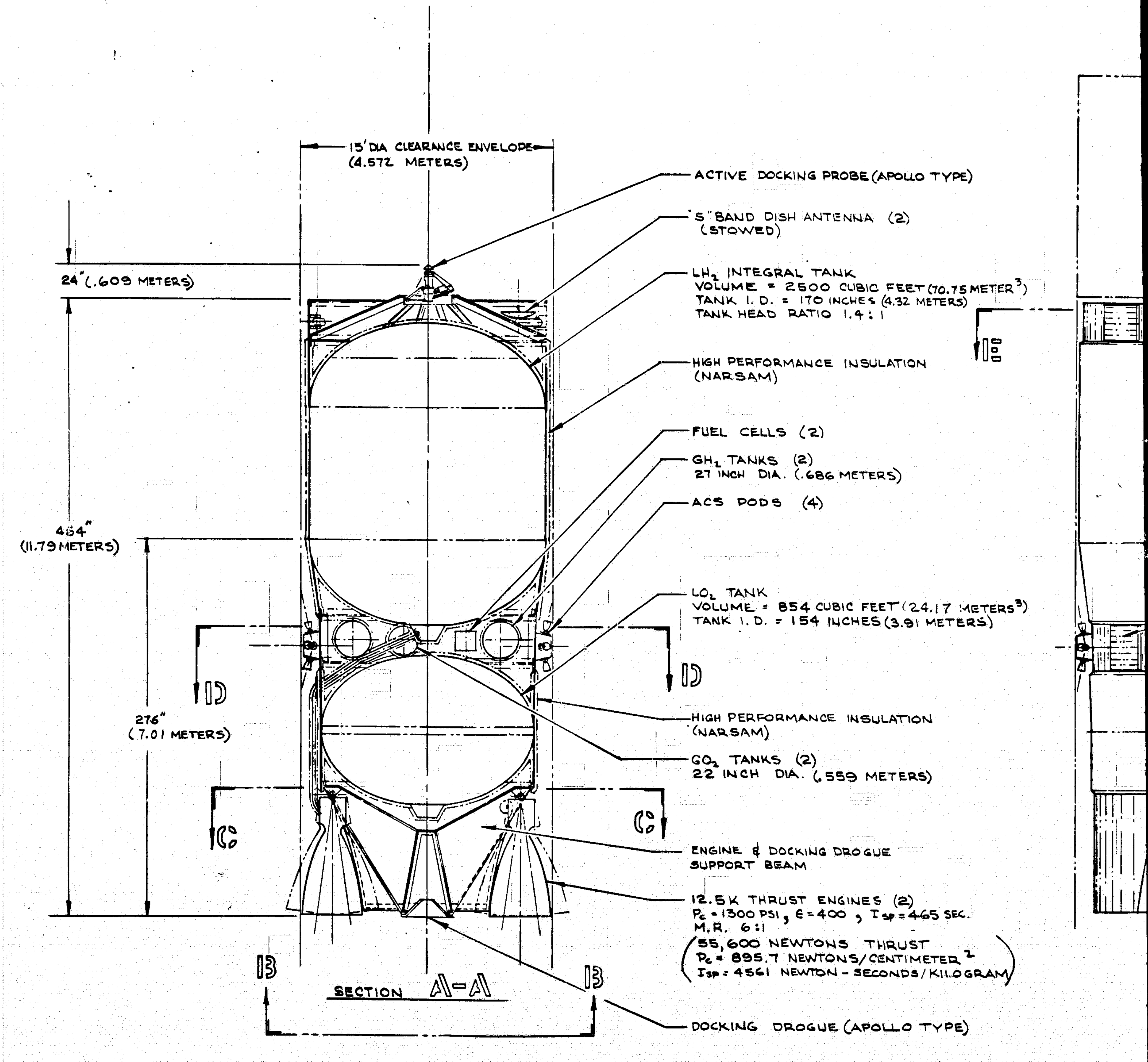
Second Design Study

The second configurational approach in this study (Figure 3-8) is based on integrating the propellant tanks into the vehicle basic structure. The configuration is an alternate to the first Concept 1 design refinement shown in Figure 3-2. This concept, identified as 1-A, investigates the integrated tank approach as well as one and four LO_2 tanks and one and two engines. The intelligence equipment and attitude control equipment are also integrated into the basic structure of the PM. The ACS engine modules do not retract in this concept but are placed on the vehicle within the 15 foot (4.57 meter) mold line in a section of the outer shell where the LH_2 tank necks down to a smaller diameter.

Concept 1A is an unmanned, single stage geosynchronous vehicle with an assumed capacity of 60,000 pounds (27,215 kilograms) of usable propellant for the main engines, 500 pounds (227 kilograms) of propellant for the attitude control engines, and 150 pounds (68 kilograms) of propellant for the electrical power systems.

The propellant tank volumes and sizes are as follows: The single liquid hydrogen tank has an internal volume of 2306 cubic feet (65.3 cubic meters), an inside diameter of 170 inches (4.3 meters), an internal length of 217 inches (5.5 meters), and is an integral part of the basic vehicle structure. The liquid oxygen tank is semi-integrated into the basic vehicle structure, and has a volume of 798 cubic feet (22.6 cubic meters), an internal diameter of 154 inches (3.9 meters), and an internal length of 110 inches (2.8 meters).

The liquid hydrogen tank configuration has a cylindrical center section with 1.4 to 1 ellipsoidal end heads, and the liquid oxygen tank is an elliptical tank with a 1.4 to 1 ratio. These tanks are made of aluminum and the structural tank support skirts are made of titanium with an option of using glass or boron filament epoxy structure. All propellant tanks and structural attach skirts are fully insulated from the space environment by high performance insulation, a vent space and a combination meteoroid barrier, and a vent shroud or purge bag. The insulation system used is known as NARSAM.



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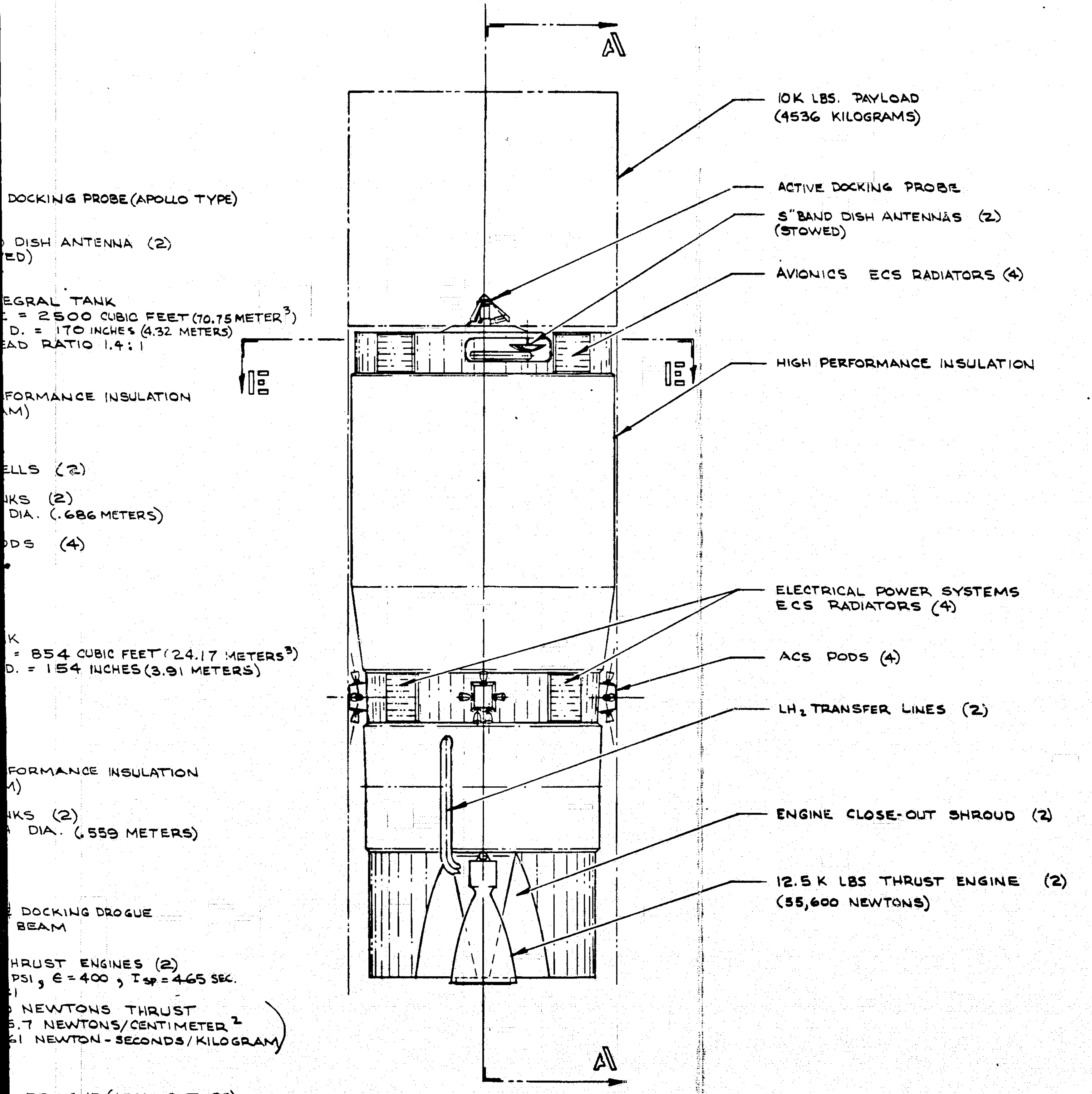


CONCEPT 1A CON

CONFIGURATION DESCRIPTION	PR
SINGLE LH_2 TANK SINGLE LO_2 TANK DUAL ENGINES	
SINGLE LH_2 TANK FOUR LO_2 TANKS SINGLE ENGINE	
SINGLE LH_2 TANK SINGLE LO_2 TANK SINGLE ENGINE	

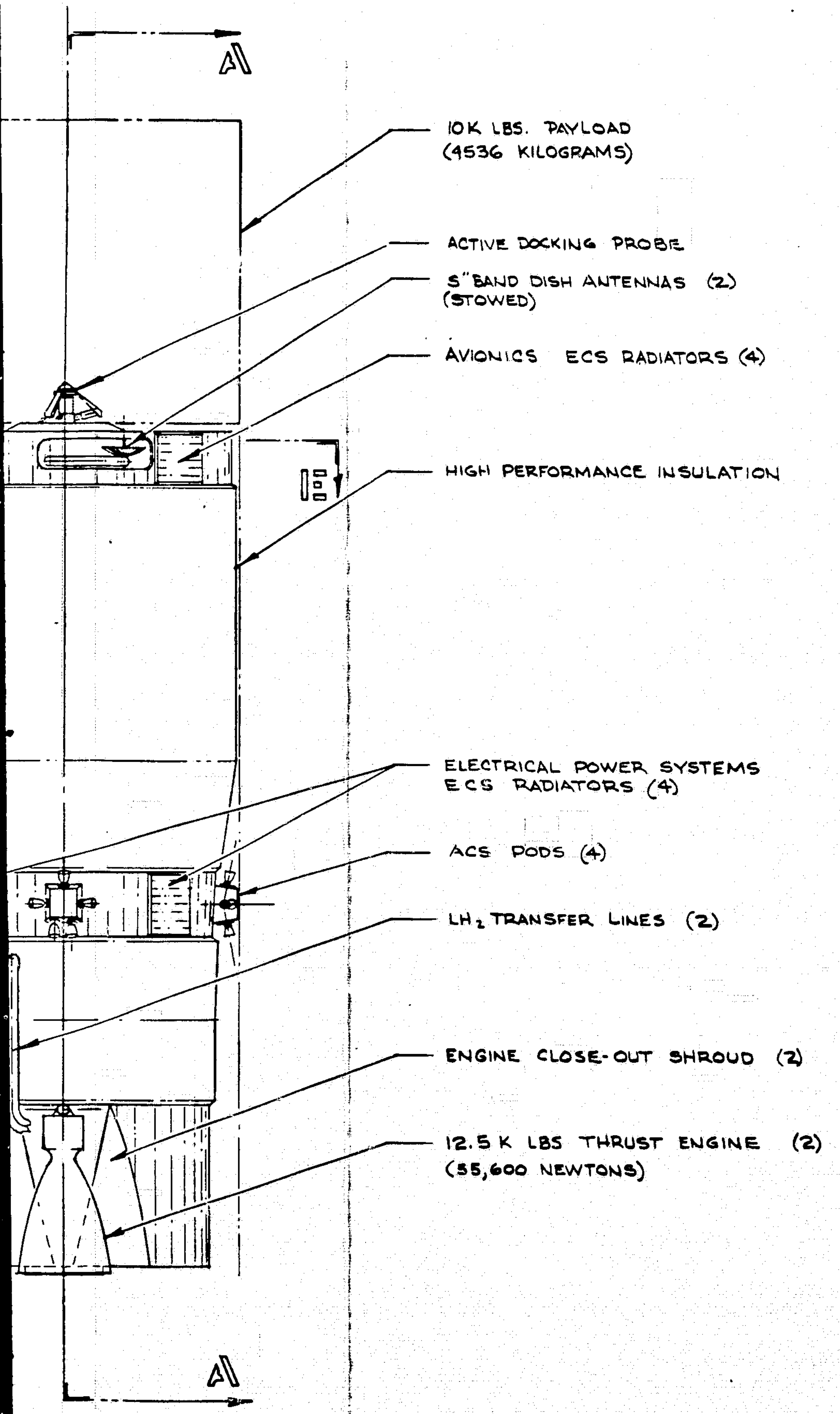
CONCEPT 1A

SCALE	DR. R.G. COOK	SPACE DIV
1/40	DATE 1-12-71	NORTH A
	MODEL	12214 LAKE

GEOSYNCHRONOUS MISSION
INTEGRAL TANKS-RECOVER
DUAL & SINGLE ENGINE VERSIO

Figure 3-8. Geosynchronous Mission, 65,000 Pounds Pro
Recoverable Unmanned Tug, Dual and Single

Space Division
North American Rockwell



CONCEPT # 1A CONFIGURATIONS:

CONFIGURATION DESCRIPTION	PROPELLANT CAPACITY	PROPULSION MODULE LENGTH
SINGLE LH_2 TANK SINGLE LO_2 TANK DUAL ENGINES	65K LBS	464 INCHES 38.5 FEET 11.79 METERS
SINGLE LH_2 TANK FOUR LO_2 TANKS SINGLE ENGINE	61 K LBS	480 INCHES 40.0 FEET 12.20 METERS
SINGLE LH_2 TANK SINGLE LO_2 TANK SINGLE ENGINE	60 K LBS	505 INCHES 42.0 FEET 12.83 METERS

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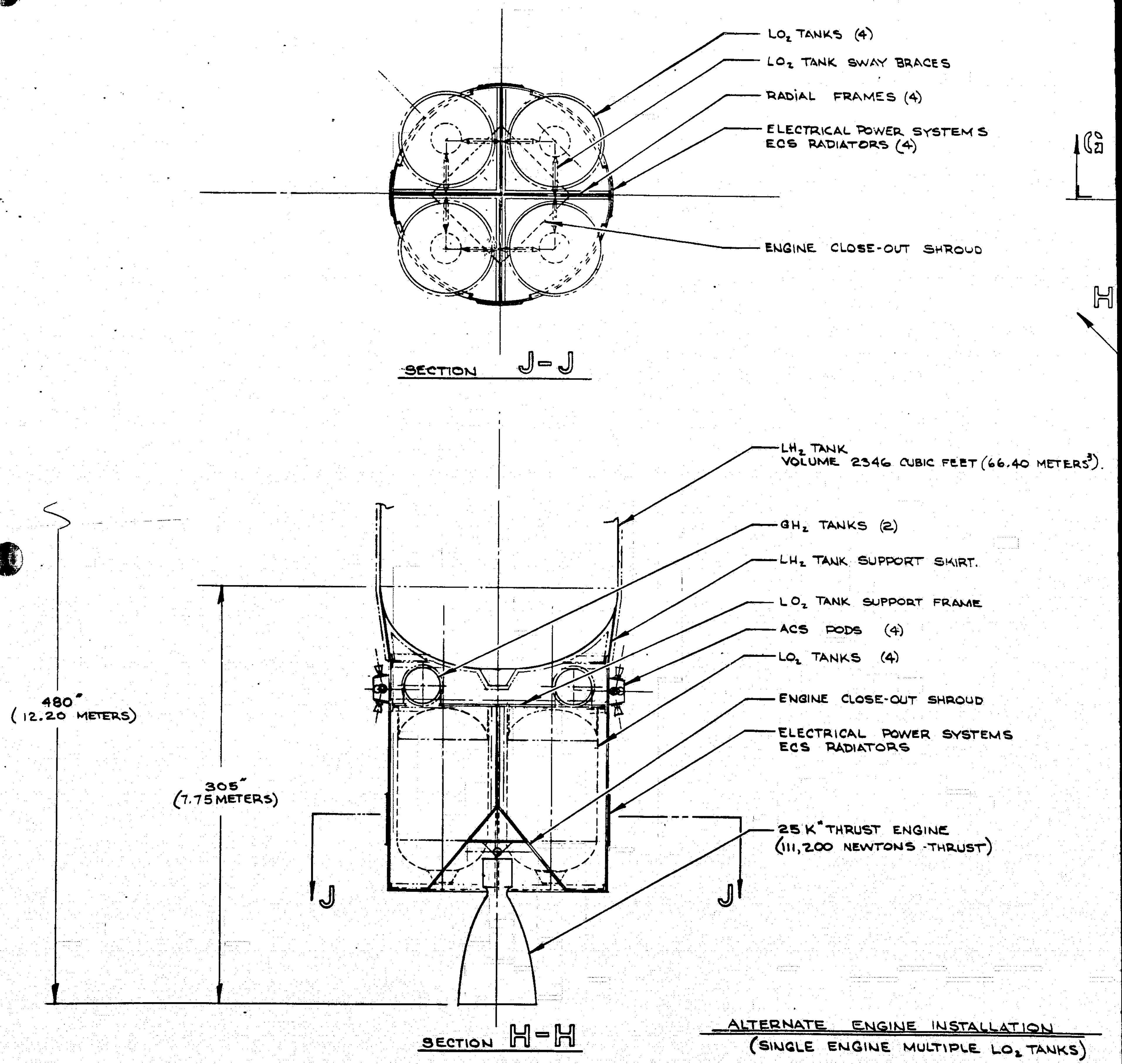
CONCEPT 1"A"

SCALE 1/40	DR R.G. COOK DATE 1-12-71 MODEL	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
GEOSYNCHRONOUS MISSION - 65K PROPELLANT, INTEGRAL TANKS - RECOVERABLE UNMANNED TUG DUAL & SINGLE ENGINE VERSIONS - SPACE TUG STUDY -			2283-36A SHEET # 1 OF 2

Figure 3-8. Geosynchronous Mission, 65,000 Pounds Propellant, Integral Tanks, Recoverable Unmanned Tug, Dual and Single Engine Versions



ALTERNATE ENGINE INSTALLATION
(SINGLE ENGINE MULTIPLE LO_2 TANKS)

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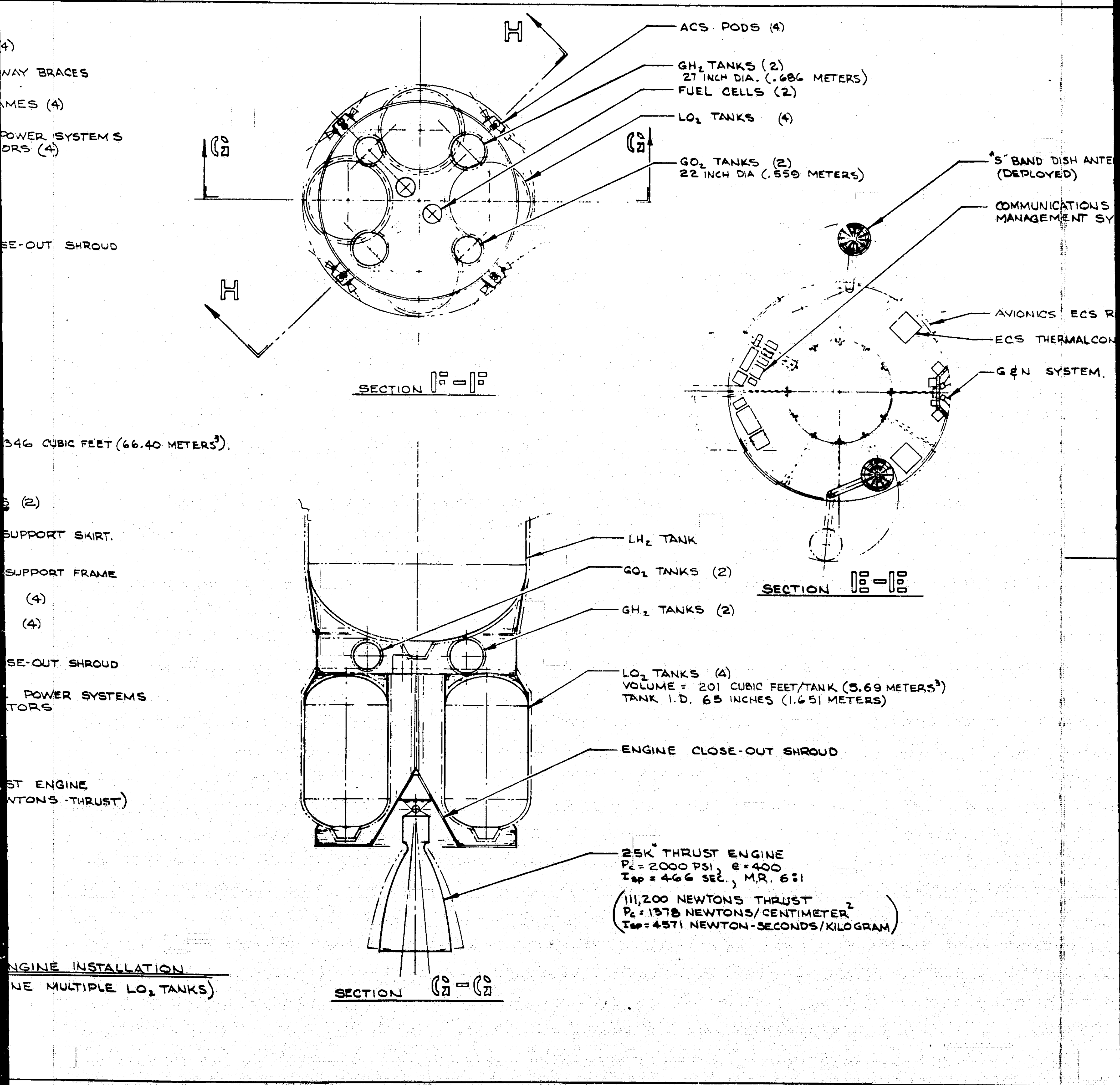

Figure 3-8. Geosyn
Recove

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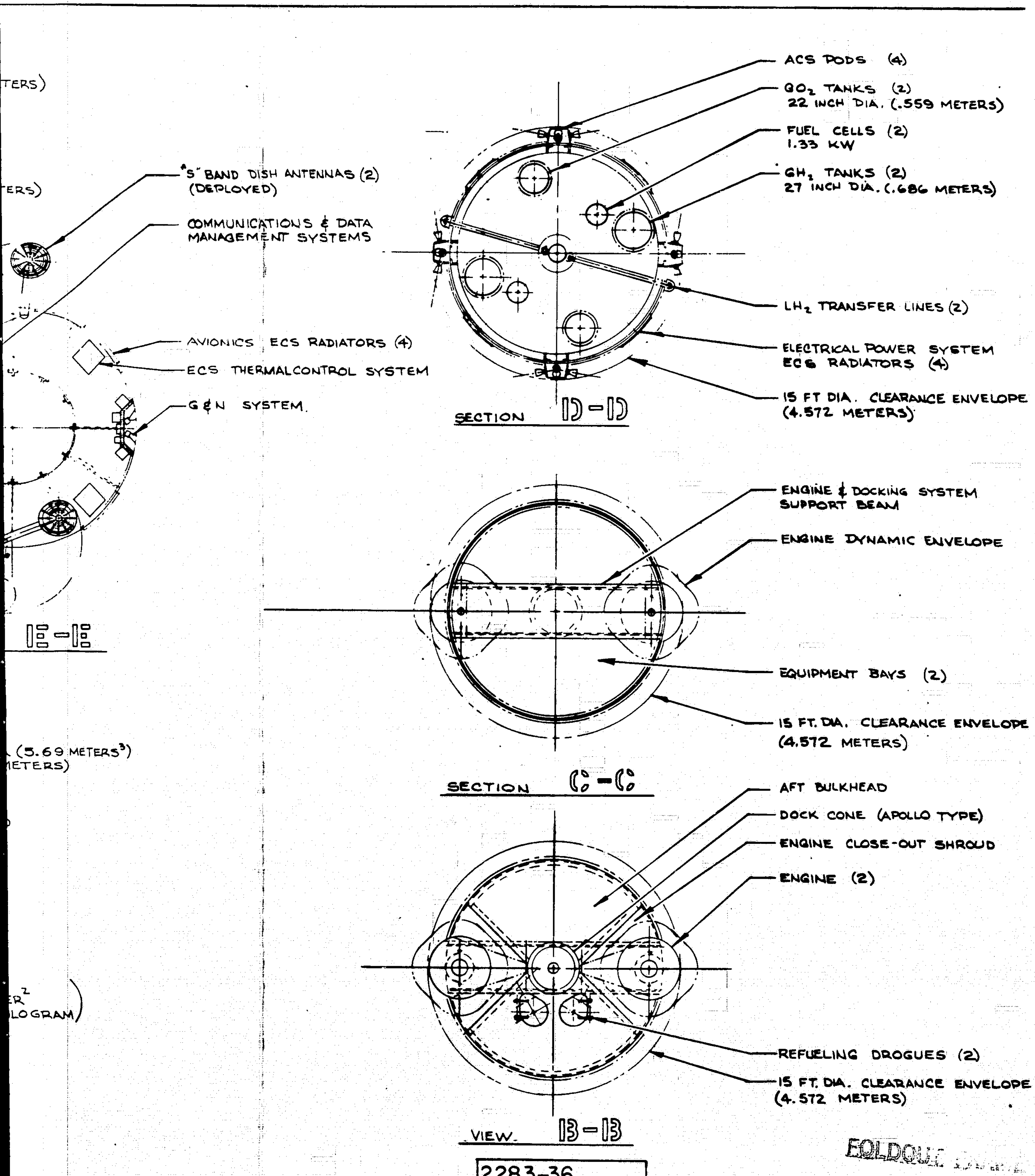

2283-36 SHT 2 OF 2

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Figure 3-8. Geosynchronous Mission, 65,000 Pounds Propellant, Integral Tanks, Recoverable Unmanned Tug, Dual and Single Engine Versions

The vehicle structure below the liquid hydrogen tank tapers down to 155 inches (3.9 meters) in diameter thus permitting the four attitude control pods to be rigidly mounted without protruding beyond the 15 foot (4.57 meters) clearance diameter for the EOS cargo bay. The attitude control pods have five nozzles on each pod, two facing aft, one facing forward, and one facing each side. The ACS equipment and tanks are located between the LH_2 and LO_2 tanks. There are two 27-inch (0.69 meters) diameter gaseous hydrogen tanks and two 22-inch (0.56 meters) gaseous oxygen tanks which are shrouded with NARSAM insulation. The electrical power system equipment such as fuel cells and batteries, and the associated environmental control equipment including radiators are also located between the LH_2 and the LO_2 tanks.

Mounted to the lower LO_2 tank skirt is the engine and docking support structural beam. Auxiliary equipment bays are located on each side of this structural member. Two 12,500 pound (55,603 newton) thrust engines with 1300 psi (896 newtons per square centimeter) chamber pressure, 400 to 1 expansion ratio, a specific impulse of 465 seconds (4560 newton seconds per kilogram), and a mixture ratio of 6 to 1, are attached to the support beam, one at each end. These engines fit into fairing cavities in the aft end of the propulsion stage. The liquid hydrogen transfer lines are routed around the liquid oxygen tank externally to the engine cavity. The docking drogue is a passive Apollo type located on the aft bulkhead and supported by the structural support beam.

Located on the aft closeout bulkhead are the refueling drogues and their control valves. The refueling equipment is fully insulated and enclosed to prevent contamination and to reduce boil-off losses. The controls for each drogue are separate to enable each drogue to be operated sequentially, preventing any mixing problem between LH_2 and LOX propellants. Fuel transfer will be performed only during a hard dock condition, that is, when the tug is rigidly docked to the propellant depot. At the forward end of the propulsion stage located on the center line is an active Apollo type docking probe, which is supported by a skin-stringer thrust cone to the upper LH_2 tank skirt and frame.

The communications, guidance and navigation, and data management avionics equipment is mounted between the forward bulkhead and the forward docking thrust structure. The 24-inch (0.61 meter) diameter S-band dish antennas are retracted into this cavity during launch in the EOS. Ten S-band omni antennas are located on the ACS pods, the dish antenna booms, and on each end of the propulsion stage. The environmental control system for the avionics is located in the same compartment as the avionics with four radiators located 90 degrees apart on the outer periphery of the structure.

The overall size of the tug is 443 inches (11.3 meters) or 36.9 feet long by 175 inches (4.4 meters) or 14.6 feet in diameter.

The alternate concept is a single engine version of Concept 1-A. The configuration is identical to the first from the lower liquid hydrogen tank skirt forward, and have the same propellant capacities. There are four liquid oxygen tanks with a volume of 199.5 cubic feet (5.6 cubic meters) per tank, an inside diameter of 65 inches (1.7 meters) and an internal length of 119 inches (3.0 meters). The liquid oxygen tanks are made of aluminum and suspended on titanium skirts from the main mid-vehicle structural frame. This structural frame and the engine support member are stabilized by four radial members, which also stabilizes the lower end of the liquid oxygen tanks with sway braces.

The main engine has 25,000 pounds (111,205 newtons) thrust, 2000 psi (1379 newtons per square centimeter) chamber pressure, 400 to 1 expansion ratio, a specific impulse of 466 seconds, and a mixture ratio of 6 to 1. The engine is mounted in a closeout shroud located between the liquid oxygen tanks on the aft end of the vehicle. The single engine location on the center line eliminates the possibility of aft docking and inflight refueling in space. The overall size of this tug is 475 inches (12.1 meters) or 39.6 feet long and 175 inches (4.4 meters) or 14.6 feet in diameter.

3.4.2 EXPENDABLE OMS TANKAGE GROUND-BASED CONCEPT (FIGURE 3-9)

The design configurations prepared during the early stages of this study and those developed during the design refinement phase were considered to be space-based vehicles where the most expensive modules or all the modules are recovered when performing the geosynchronous mission. Another approach is to design a ground-based expendable stage. This vehicle would have a reduced propellant quantity and overall vehicle weight. The structural approach should be the lightest and lowest cost concept feasible. Because the stage is expended to perform the mission the various components of the stage, avionics, propellant tanks, engine components, etc., should be of the lowest cost. A desirable feature would be the use of components that are used in other vehicles of the same operational period as the tug. The prime candidate for such components is the earth-orbital shuttle, the vehicle which will launch this expendable tug.

Much of the equipment used in the tug is based on that which will be used in the EOS. Consequently, a great savings can be realized in this area for the expendable stage. For instance, the Orbital Maneuvering System of EOS currently utilizes an engine size which is approximately the size and type needed for tug. An investigation was made of the various EOS orbiter

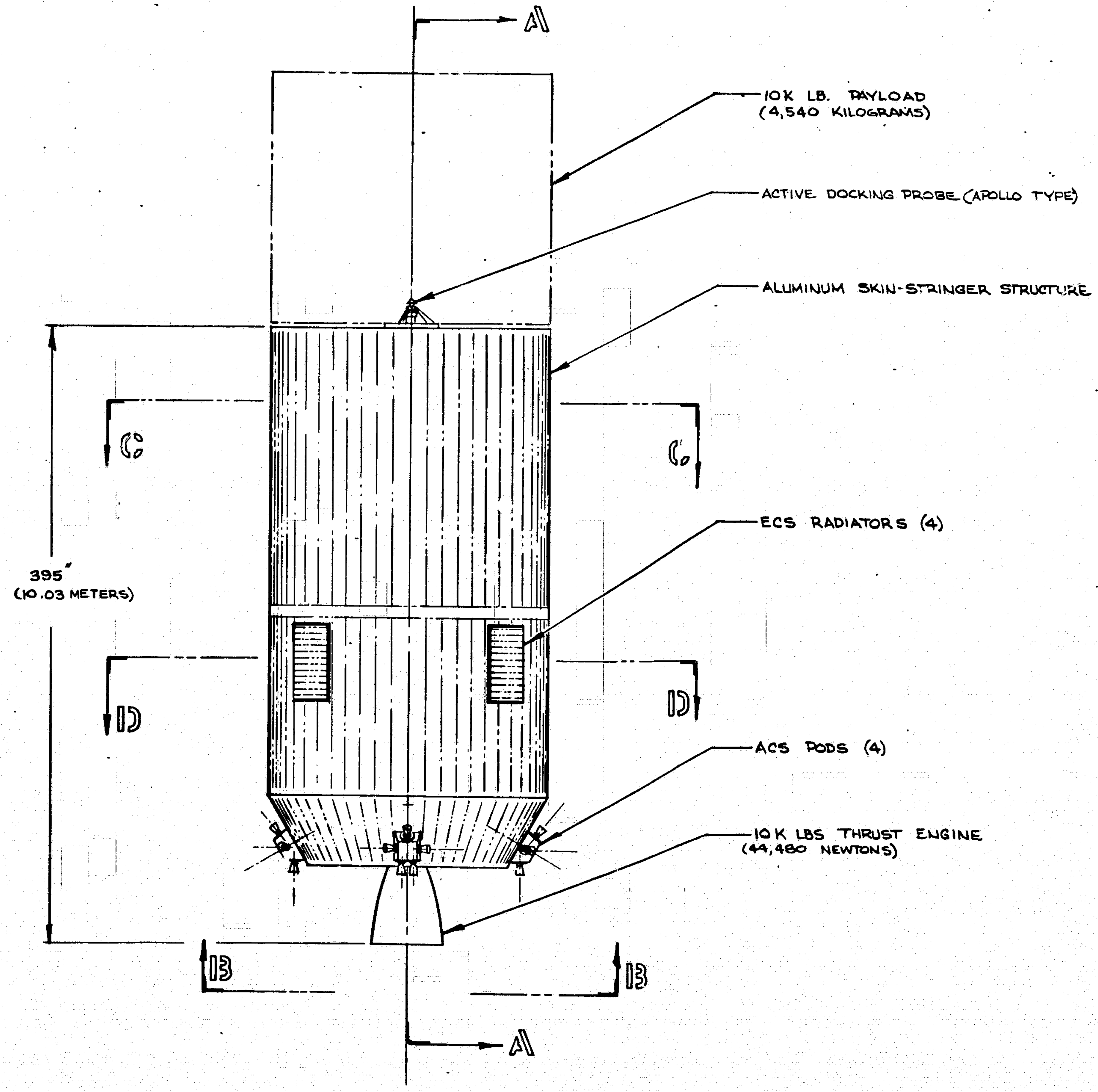

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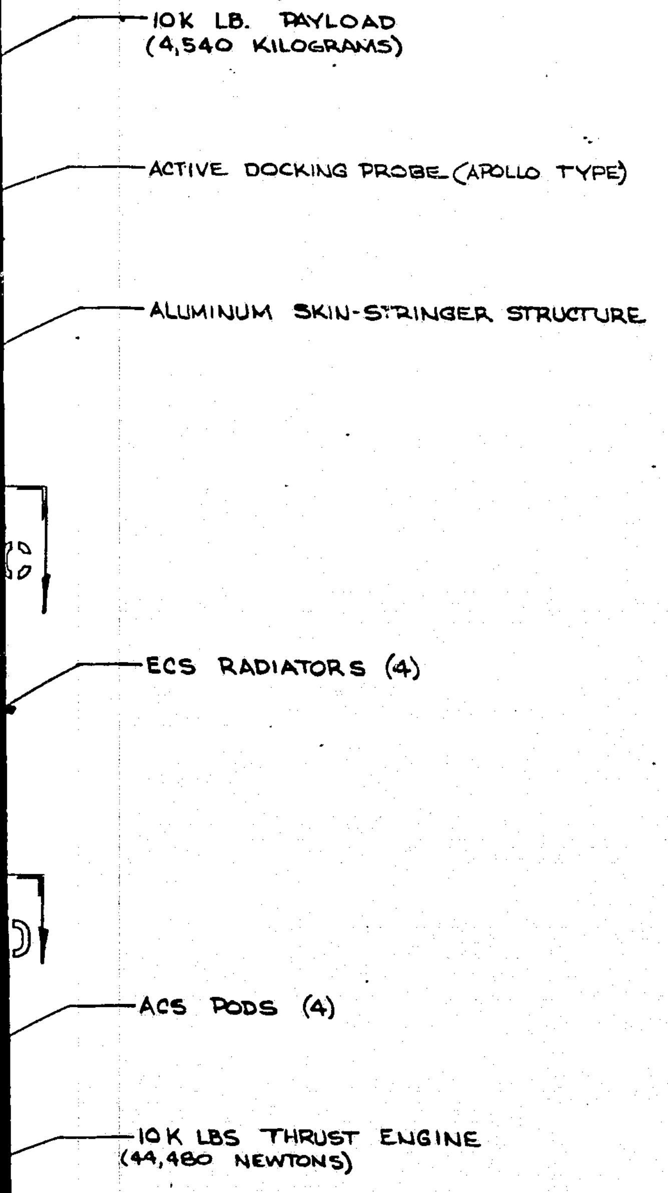
Figure 3-9. Geosynch
Ground B



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CONCEPT -7A

SCALE 1/40	DR. R G. COOK DATE 1-19-71 MODEL	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
GEOSYNCHRONOUS MISSION-30K* PROPELLANT CAPACITY - GROUND BASED, EXPENDABLE SPACE TUG (SPACE TUG STUDY)			2283-38 SHEET # 1 OF 2

Figure 3-9. Geosynchronous Mission, 30,000 Pound Propellant Capacity, Ground Based Expendable Space Tug

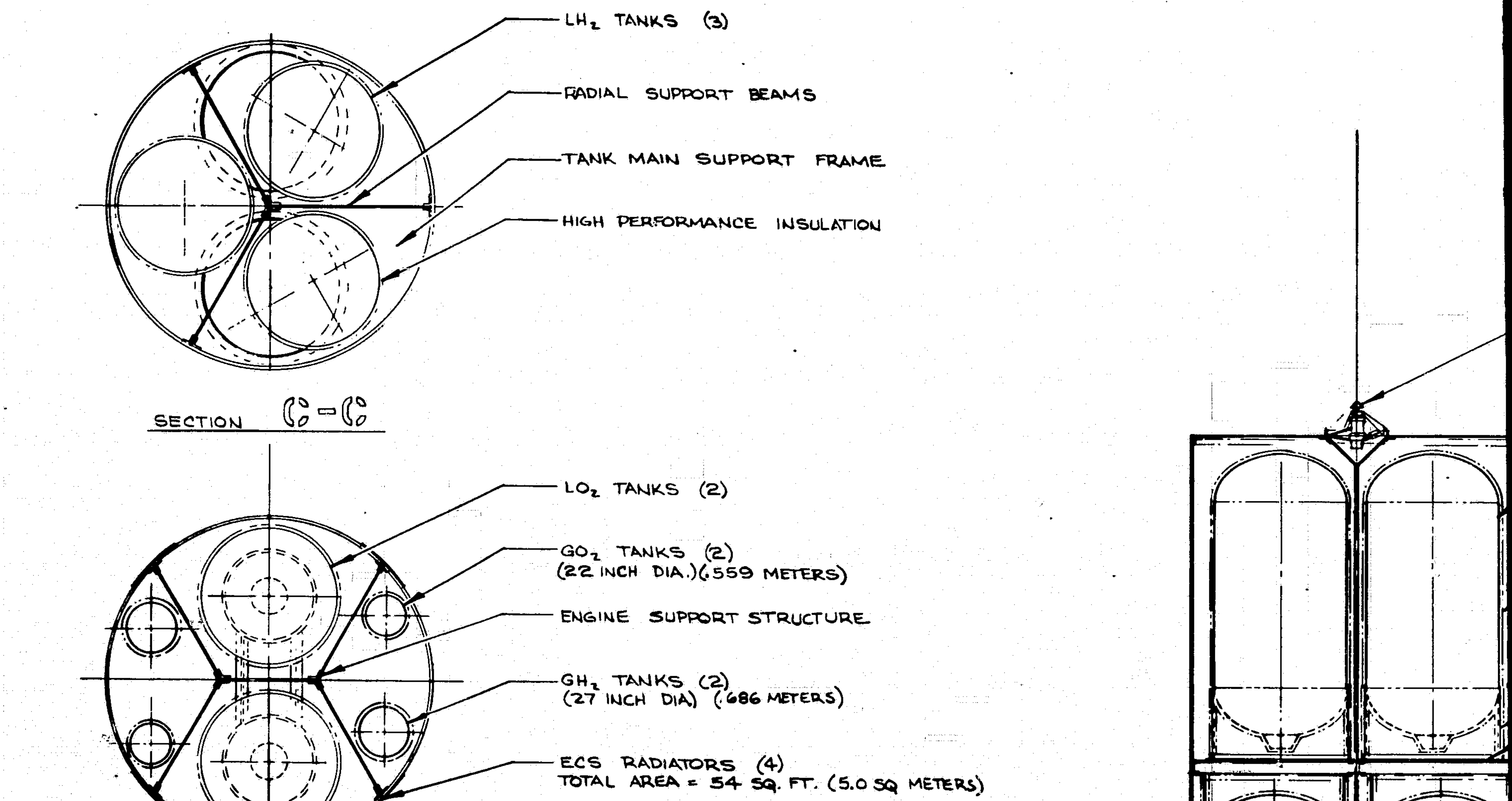

Figure 3-9. Geosynchron
Ground Base

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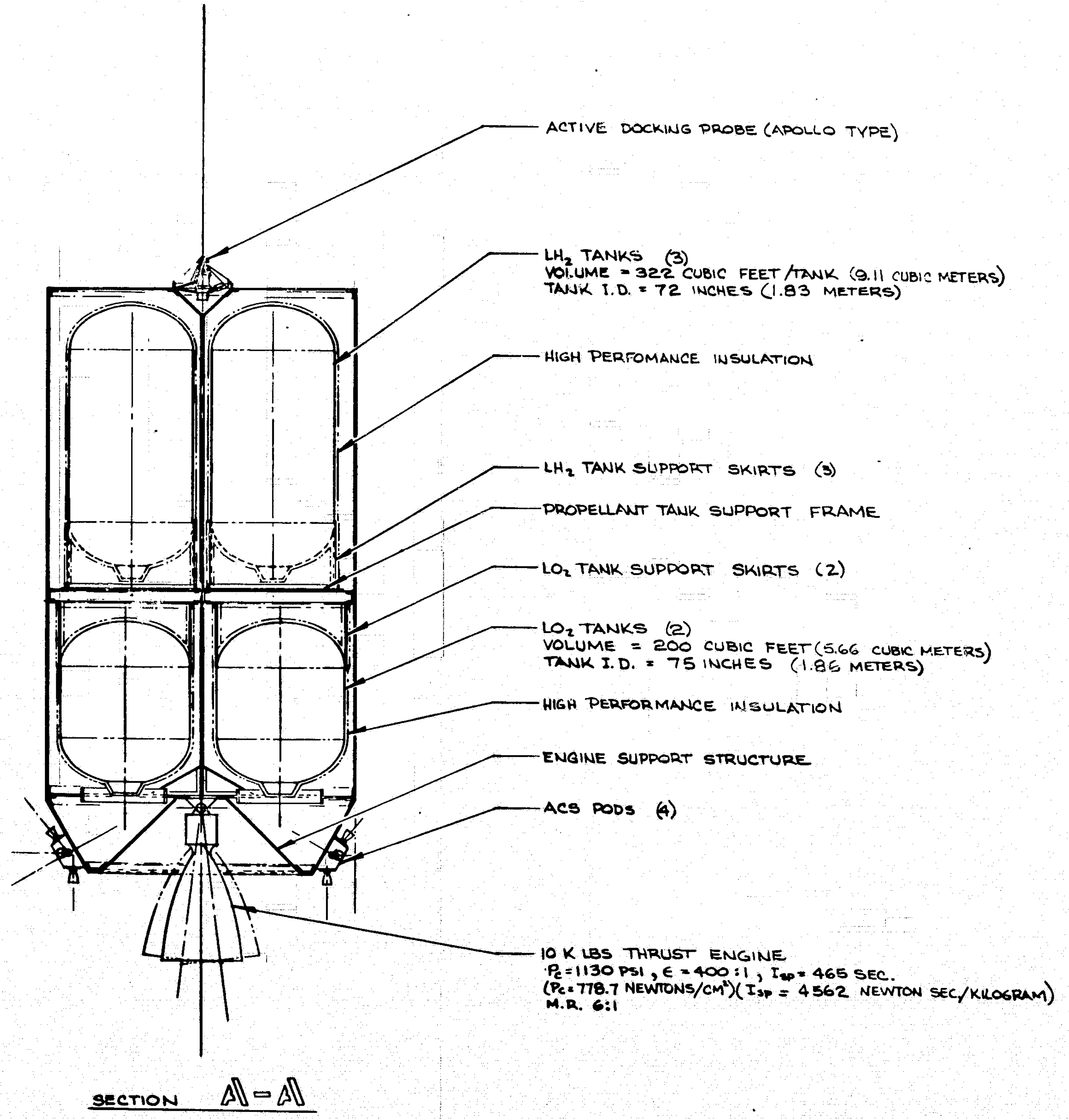

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2283-38
SHT 2 OF 2

Figure 3-9. Geosynchronous Mission, 30,000 Pound Propellant Capacity, Ground Based Expendable Space Tug

configurations to determine if there existed an OMS O_2/H_2 tankage concept of sufficient volumes to use in the expendable tug vehicle. While the OMS tankage design is still undergoing study and is subject to change, the idea of common usage appears to have merit as a potential item of commonality. A total of about 30,000 pounds (13,608 kg) is required to perform the mission. An EOS orbiter configuration was found which contained OMS tankage of sufficient size for use in this vehicle. The OMS tankage selected had an LH_2 tank with an internal volume of 322 cubic feet (9.1 cubic meters) and a LOX tank with an internal volume of 200 cubic feet (5.7 cubic meters). To satisfy the propellant load required by this expendable stage, three LH_2 and two LOX tanks are required.

The configured concept is shown on Figure 3-9 as an unmanned expendable single-stage geosynchronous mission tug vehicle which incorporates the EOS orbiting maneuvering system propellant tanks and engines for its main propulsion system. The LH_2 tank is 72 inches (1.8 m) inside diameter with a volume of 322 cubic feet (9.1 cubic meters) and three are clustered at the forward end of the vehicle. The LOX tanks are 75 inches (1.9 m) inside diameter with an internal volume of 200 cubic feet (5.7 cubic meters) and two are required and are located aft of the LH_2 tanks. Each of the tanks contains a center cylindrical section with 1.401 ratio elliptical bulkheads.

The propellant tanks are supported on cylindrical skirts attached to a common tank support bulkhead. The three LH_2 tanks are located forward of the common bulkhead and are separated by radial structural frames. Lateral stabilizing struts for each tank forward end are attached to these radial frames. The two LOX tanks are located aft of the common bulkhead and attached to it through cylindrical struts also. The LOX tank literal stabilizer struts are located at the aft end of each tank and are attached to a cruciform type structure between the tanks. This aft section is divided into four compartments. Two of these contain the LOX tanks and the remaining two contain the attitude control and avionics equipment.

The structure of the aft compartment supports the tanks and equipment, and also is the thrust structure for the main propulsion engine. The attitude control engine modules are rigidly attached to the outer part of the aft conical section closeout shroud. This location keeps them within the EOS cargo bay 15-foot (4.57 m) payload envelope without requiring retraction.

An active docking probe, similar to the Apollo docking probe, is located on the forward end of the propulsion module on the vehicle center line to space attachment to payloads or other equipment. There are no provisions for docking on the aft end of the vehicle, nor are there any refueling provisions since the concept is assumed to be ground-based.

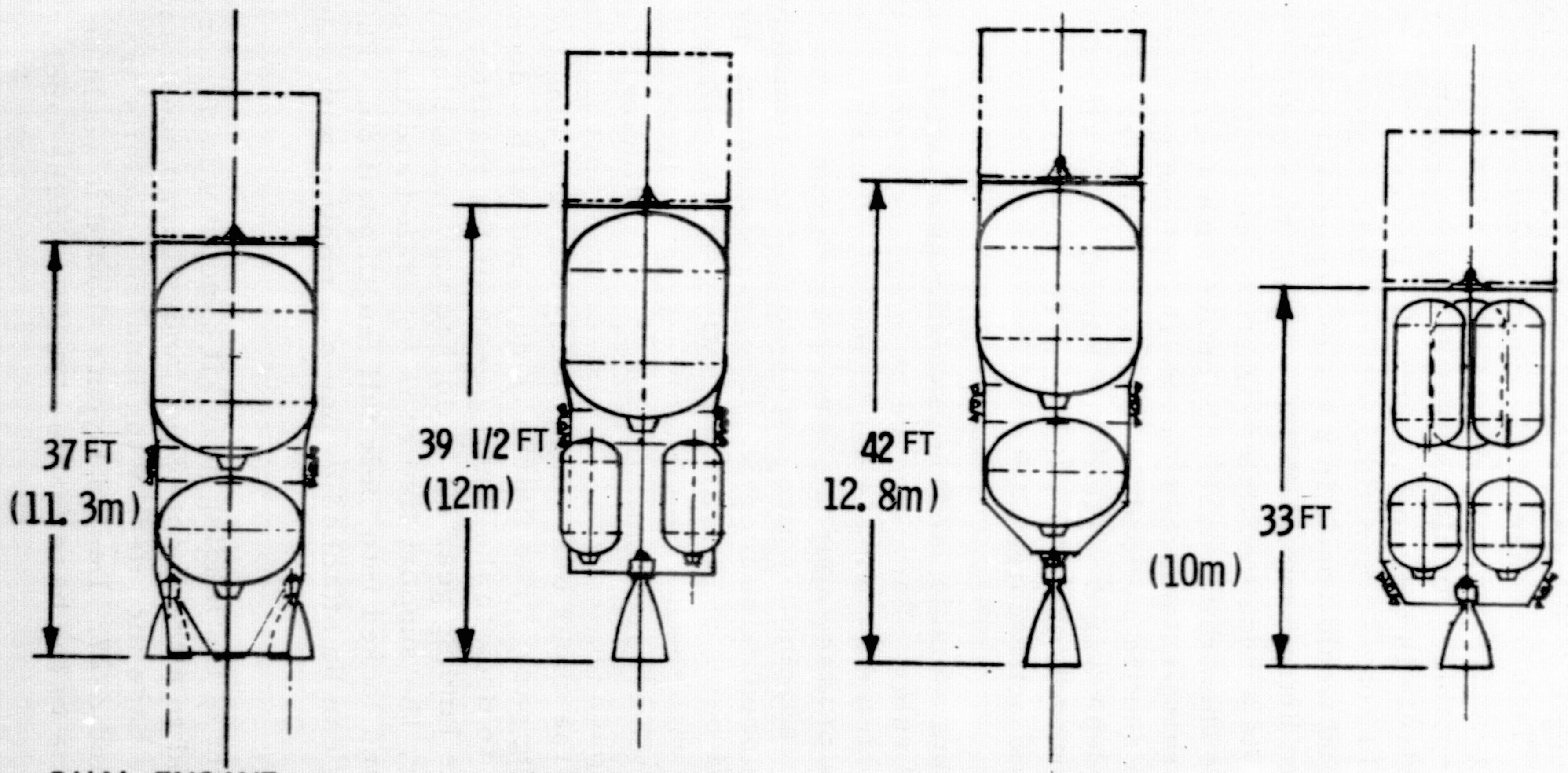
The main propulsion system is a single engine which has a 10,000 pound (44,482 N) thrust rating, a chamber pressure of 1130 psia (778.7 N/CM^2), an area expansion ratio of 400 to 1 and a specific impulse of 465 seconds (4562 N-S/kg). Each attitude control module is equipped with five 200-pound (890 N) thrust engines, two of which face aft, one forward and one to each side. The ECS system equipment is located in the avionics equipment section and the spare radiators are located in four places on the periphery of the external shell structure. The radiators are just aft of the common bulkhead and are spaced 90 degrees apart and are offset radially 45 degrees from the attitude control engine modules. The radiators have a total area of 54 cubic feet (5 m^3) which is what is required for the expended mission.

The ground-based propellant tank insulation concept differs slightly from that used for the space-based systems. The major difference is in the insulation of the LH_2 tanks. The LOX tank insulation system consists of three 50-layer blankets of HPI insulation attached directly to the tank outer wall. A venting space filled with nitrogen gas separates this insulation from a combination meteoroid barrier and purge bay. The concept for the LH_2 tanks is identical with the exception that a foam substrate is first applied to the tank outer wall before the HPI blanket insulation is attached. The details of the thermal design philosophy are contained in Appendix F.

The results of these special operational compromise concepts were evaluated following the conceptual design effort. The results are summarized in Figure 3-10.

3.4.3 LANDING GEAR INTEGRATION

One of the key design study areas related to investigations of multi-purpose conceptual approaches is providing landing gear on the tug PM derivatives for the lunar landing mission. Initial landing gear studies indicated that the landing gear should be of a conventional four legged configuration consisting of a longitudinal strut and pair of lateral struts for each gear assembly, all strut elements providing stroking for landing attenuation. The addition of the landing gear to the basic tug PM presents a major configurational problem. Since the tug is constrained to be transported to earth orbit by the space shuttle vehicle, its diameter is established by the allowable payload envelope that can be accommodated in the shuttle cargo bay. Currently, this diameter is set at 15 feet (4.57 meters). Two options are available for addition of the previously described four legged landing gear to the basic PM; the landing gear can be integrated into the vehicle as a ground installation during fabrication of the tug lander, or the gear can be installed in earth or lunar orbit. The first approach impacts the basic tug PM design to a major degree. In the stowed configuration, the integrated



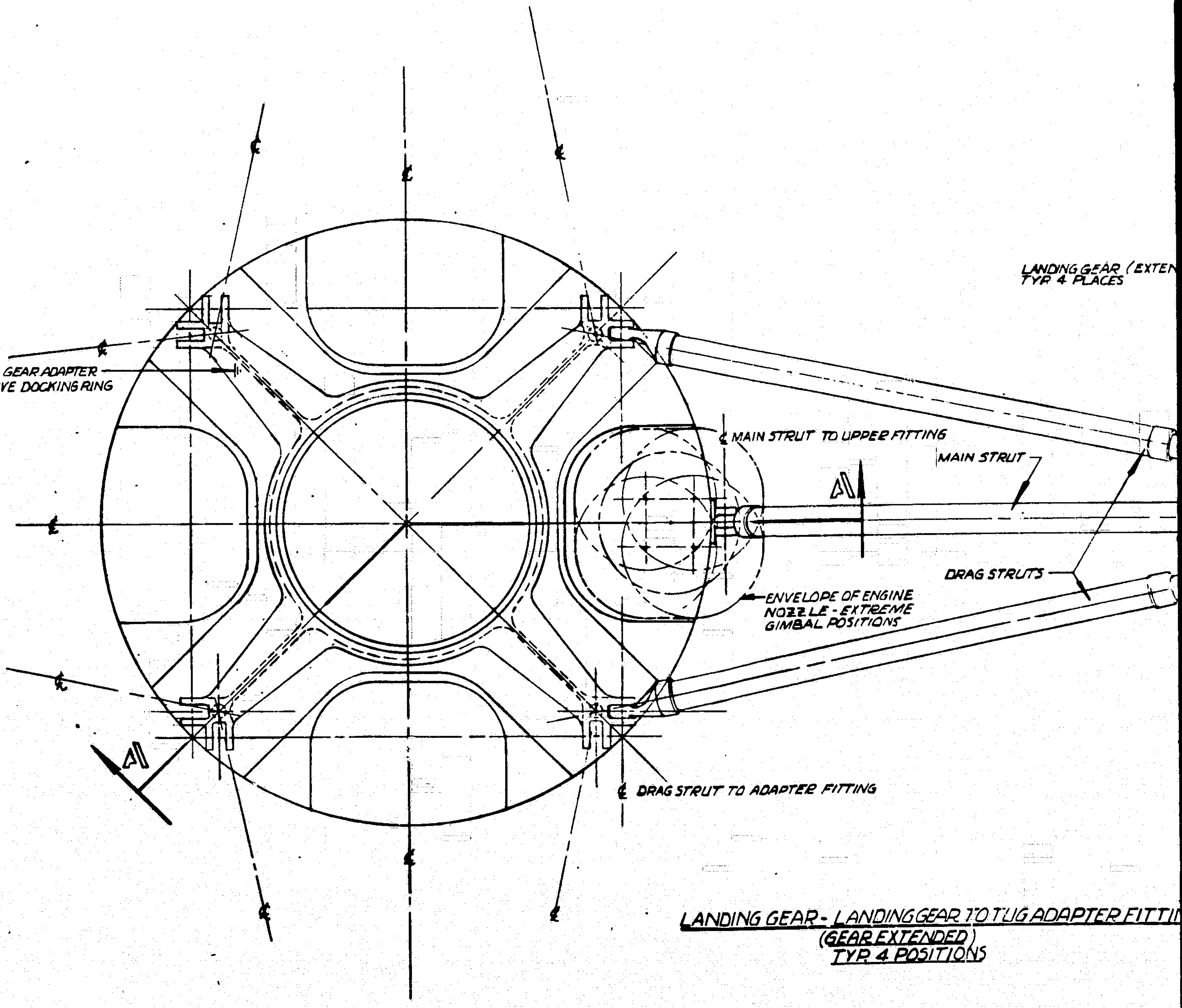
	DUAL ENGINE LIGHT AVIONICS RECOVERABLE	SINGLE ENGINE NO AFT DOCKING	SINGLE ENGINE SINGLE LOX TANK	SINGLE ENGINE LOW COST EXPENDABLE OMS TANKAGE
PROP WT:	65K (29.4 kg)	61K (27.6 kg)	60K (27.2 kg)	30K (13.6 kg)
GROSS WT:	84K (38.0 kg)	80K (26.2 kg)	78K (35.3 kg)	

Figure 3-10. Conceptual Design Summary

landing gear will require the longitudinal strut hinge to lie along side of the body. Since the allowable payload diameter is restricted to 15 feet (4. 57 meters) the basic diameter of the tug PM must be reduced so that the diameter across the landing gear struts is within the 15-foot (4. 57 meter) limit. Considering a reasonable landing gear strut diameter of 15 inches (0. 38 meter) this will result in a reduction in PM diameter of 16 percent and an increase in PM length of 40 percent in order to provide the same propellant tank volume. This increase in PM length would preclude accommodating many of the tug configurations in the shuttle cargo bay; therefore this approach is not acceptable. An alternative design approach would be to recess the longitudinal struts in the side of the PM. However, the four structural recesses would not be compatible with the four engine propulsion system arrangements, and would result in a major structural weight and configuration penalty on a bottom mounted CM. Again, this option is not considered acceptable. As a result of this evaluation, conceptual studies were initiated to examine the orbital assembly option with a landing gear kit. Conceptual ground rules established include the following: landing gear kit to be compatible with shuttle payload compartment diameter and length constraints and payload deployment system; the landing gear kit should be capable of being transported in orbit by a nominal Tug vehicle and the installation on the lander should not require EVA.

Figure 3-11 presents a viable landing gear kit which satisfies all of the previously stated ground rules. Basically the kit consists of a four legged landing gear assembly which does not exceed 15 feet (4. 57 meters) in diameter, and is folded into a package approximately 28 feet long. The landing gear kit features a frame assembly which can be docked and remotely latched to the aft bulkhead of the propulsion module. The frame assembly provides for attachment of the four pairs of lateral attenuation struts, and is configured to be compatible with the nozzle clearance requirements of the four rocket engines. In the stowed configuration, the longitudinal struts are folded to reduce overall package length. An adapter transport fixture is provided which supports the upper body hinge fitting for each longitudinal strut, and the four landing pads. The fixture is equipped with a docking interface to permit a tug to dock to the landing gear kit for transfer to the tug lander vehicle. After the landing gear kit is docked and mechanically attached to the tug lander, the support fixture is separated and the longitudinal struts are deployed and locked into their full length position. The longitudinal strut hinge fitting can be attached by use of a minitug with manipulator arms.

Further development of the landing gear kit concept was performed in subsequent studies related to attachment of the lander cargo pods, and transportation of lunar surface base modules by tug landers. Details are shown on Figures 2-25, 2-26, and 2-27. Design layouts showed it would be feasible to add hinged support arms to the base frame structure of the landing gear



LANDING GEAR - LANDING GEAR TO TUG ADAPTER FITTI
(GEAR EXTENDED)
TYP 4 POSITIONS

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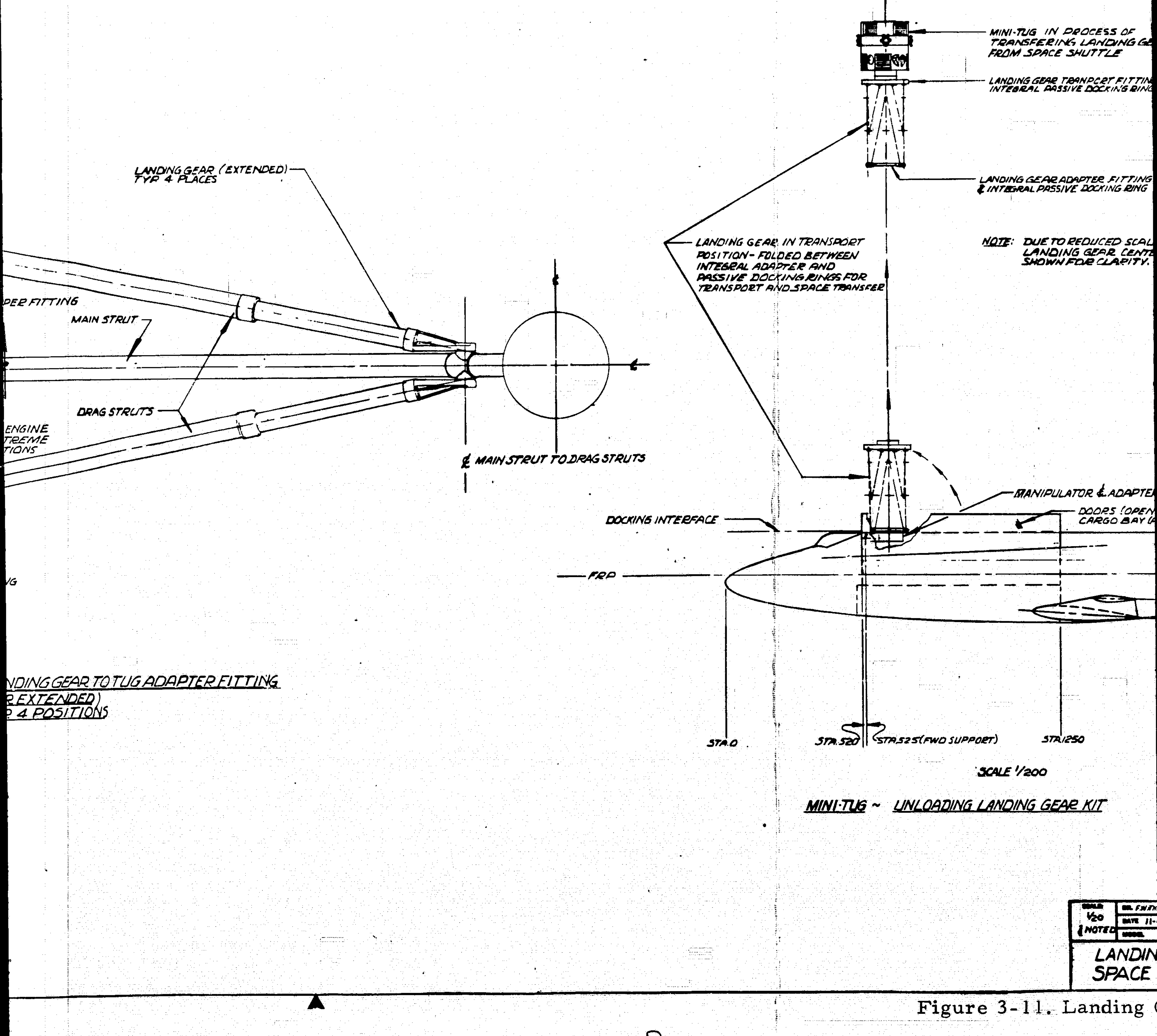


Figure 3-11. Landing (

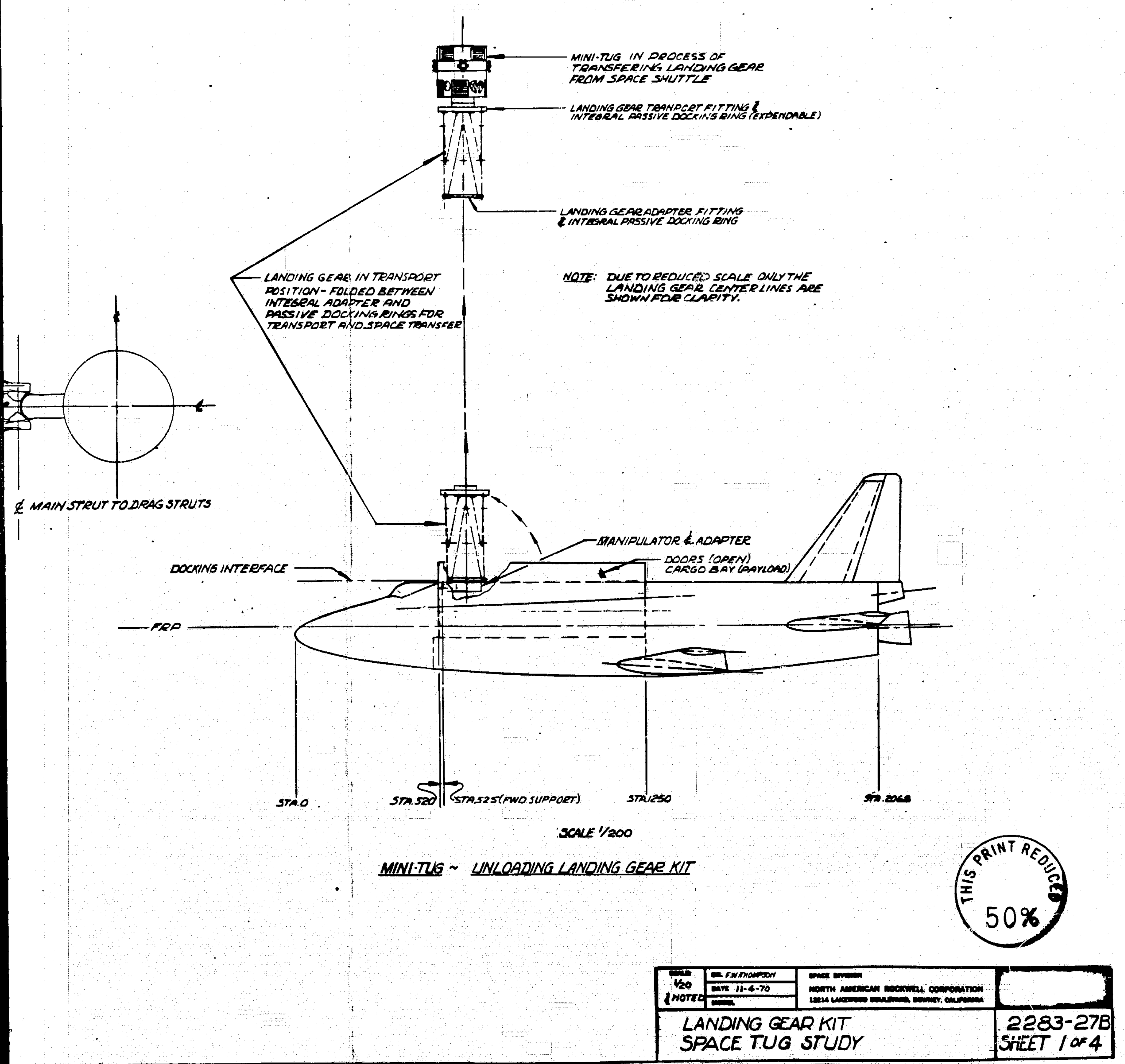
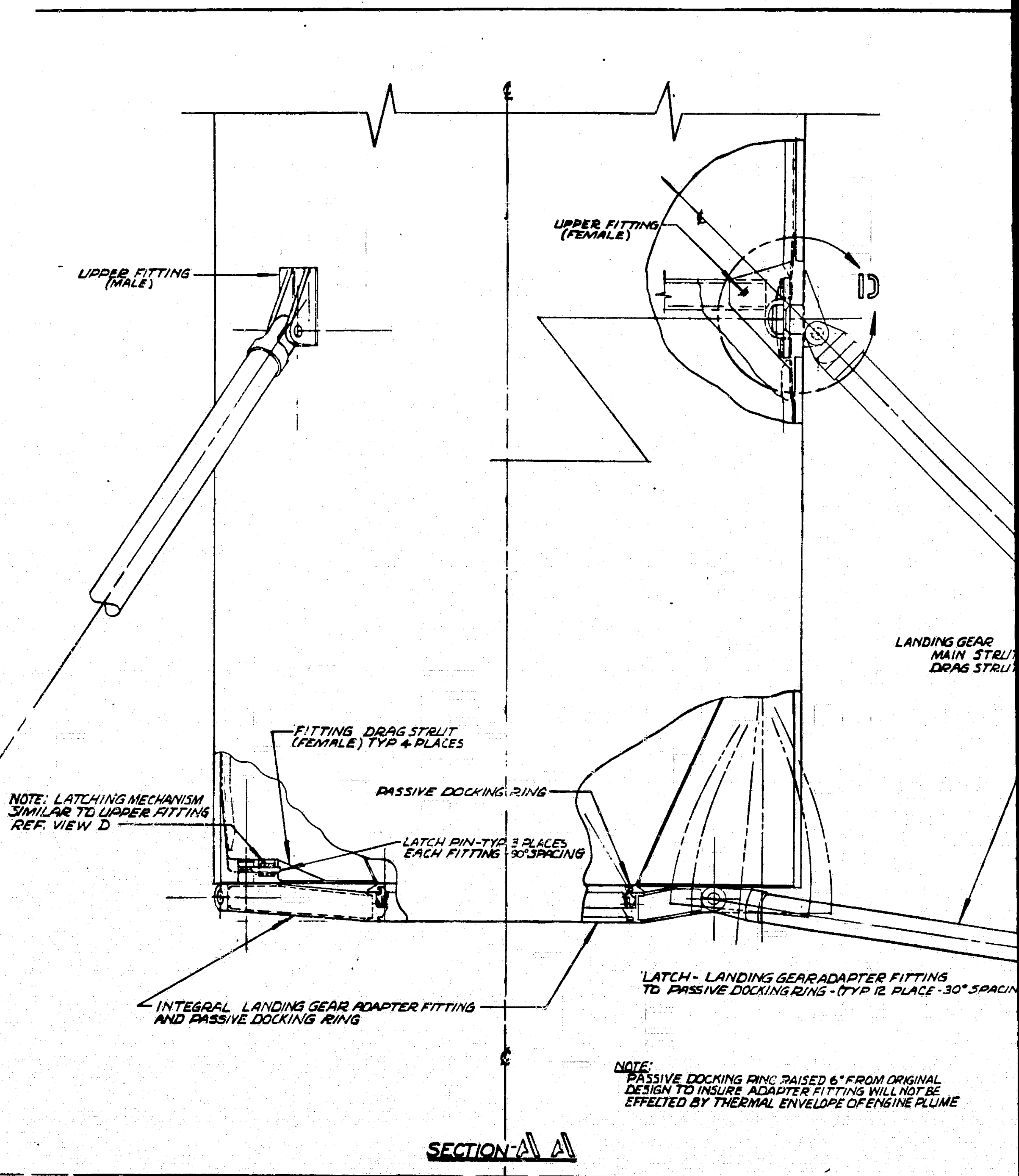


Figure 3-11. Landing Gear Kit



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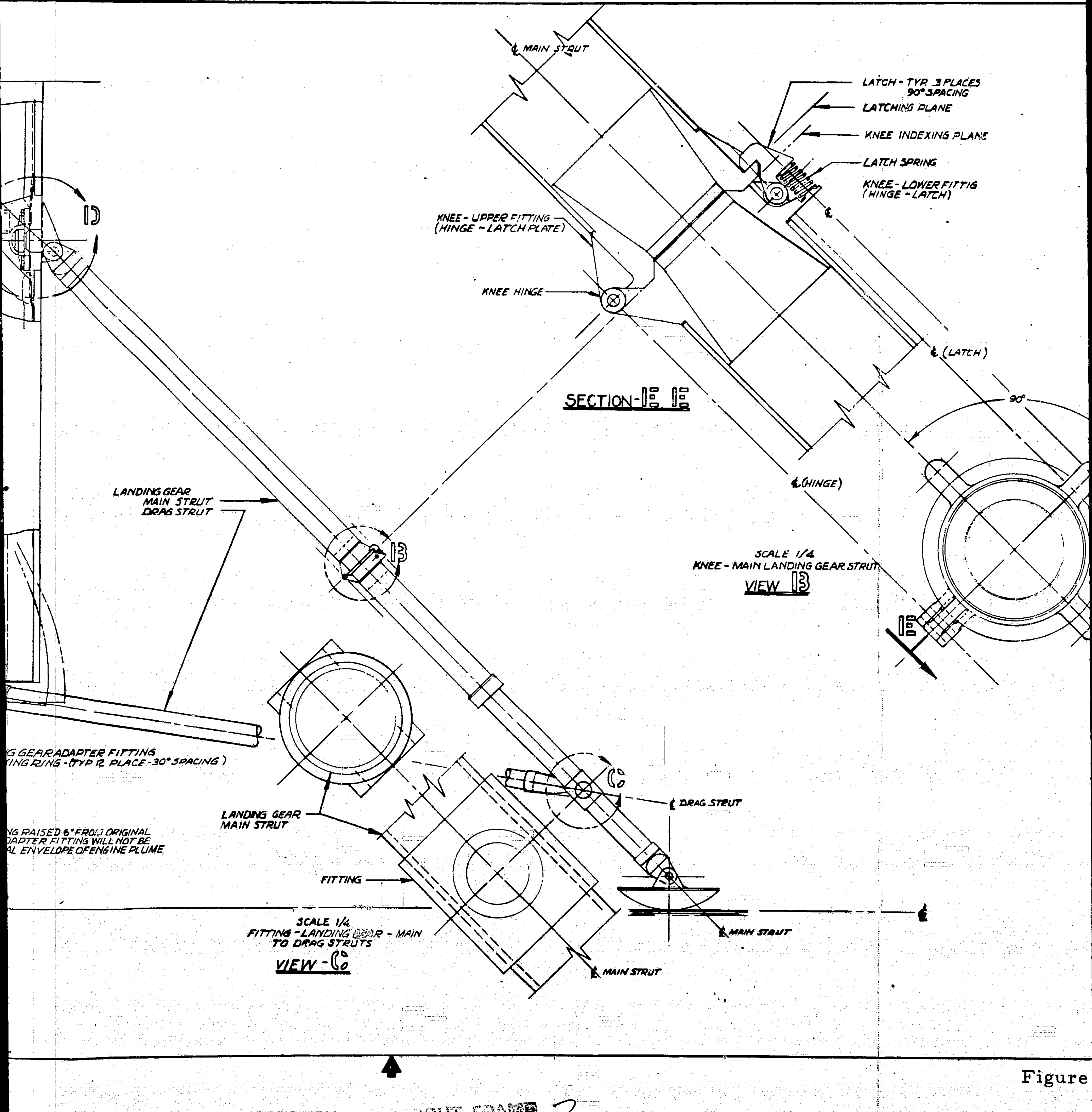

Figure

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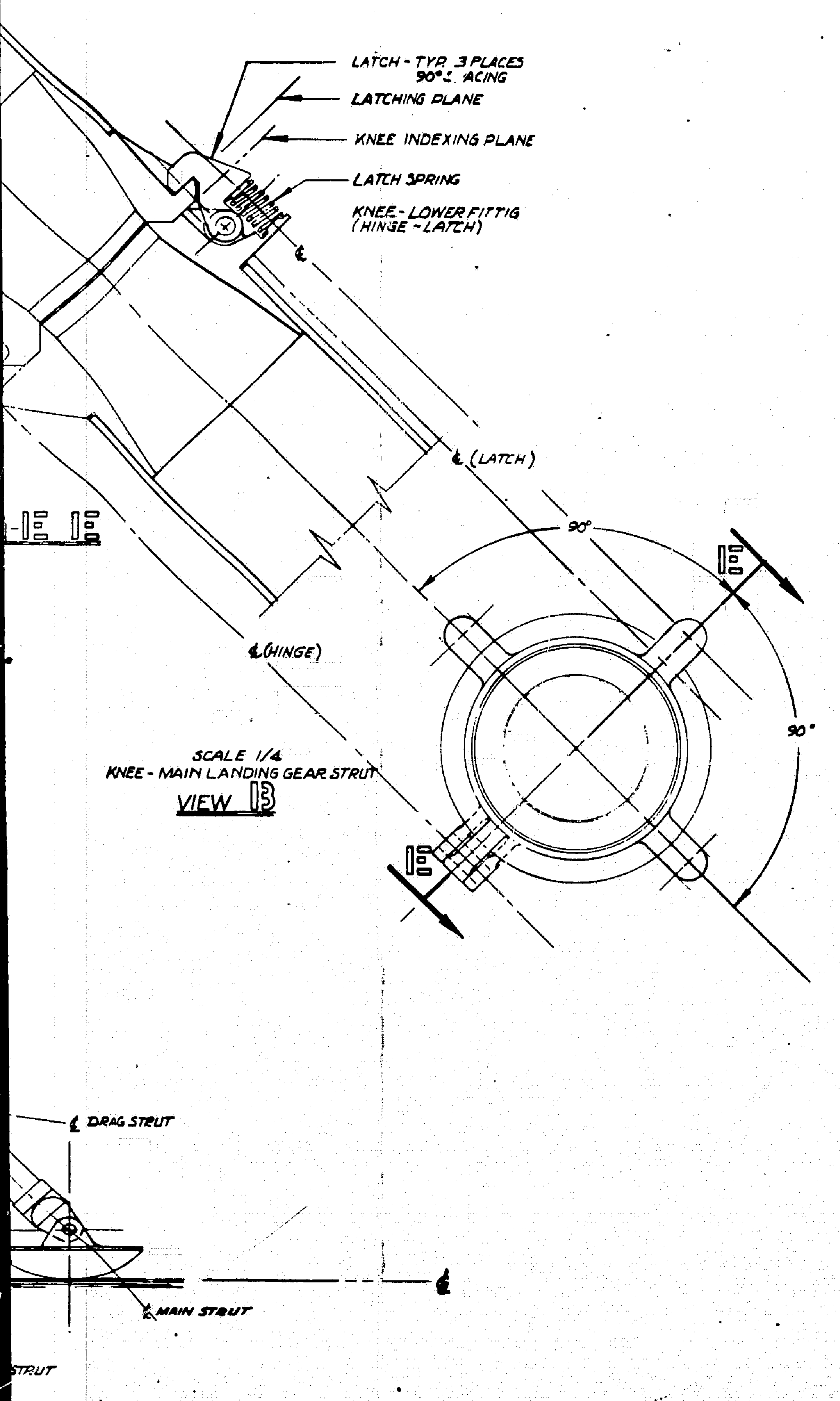

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SHEET 2 OF 4

Figure 3-11. Landing Gear Kit

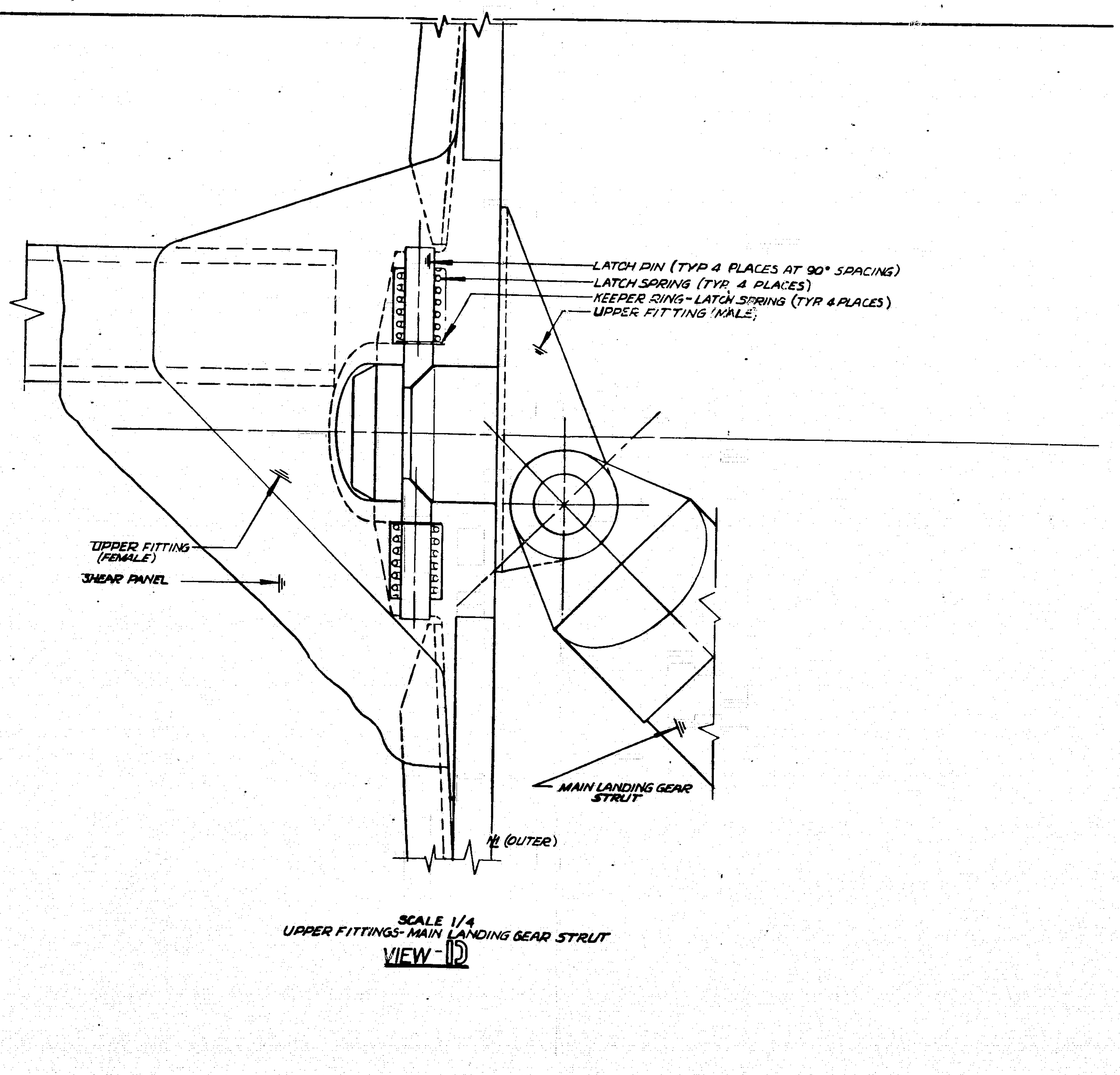
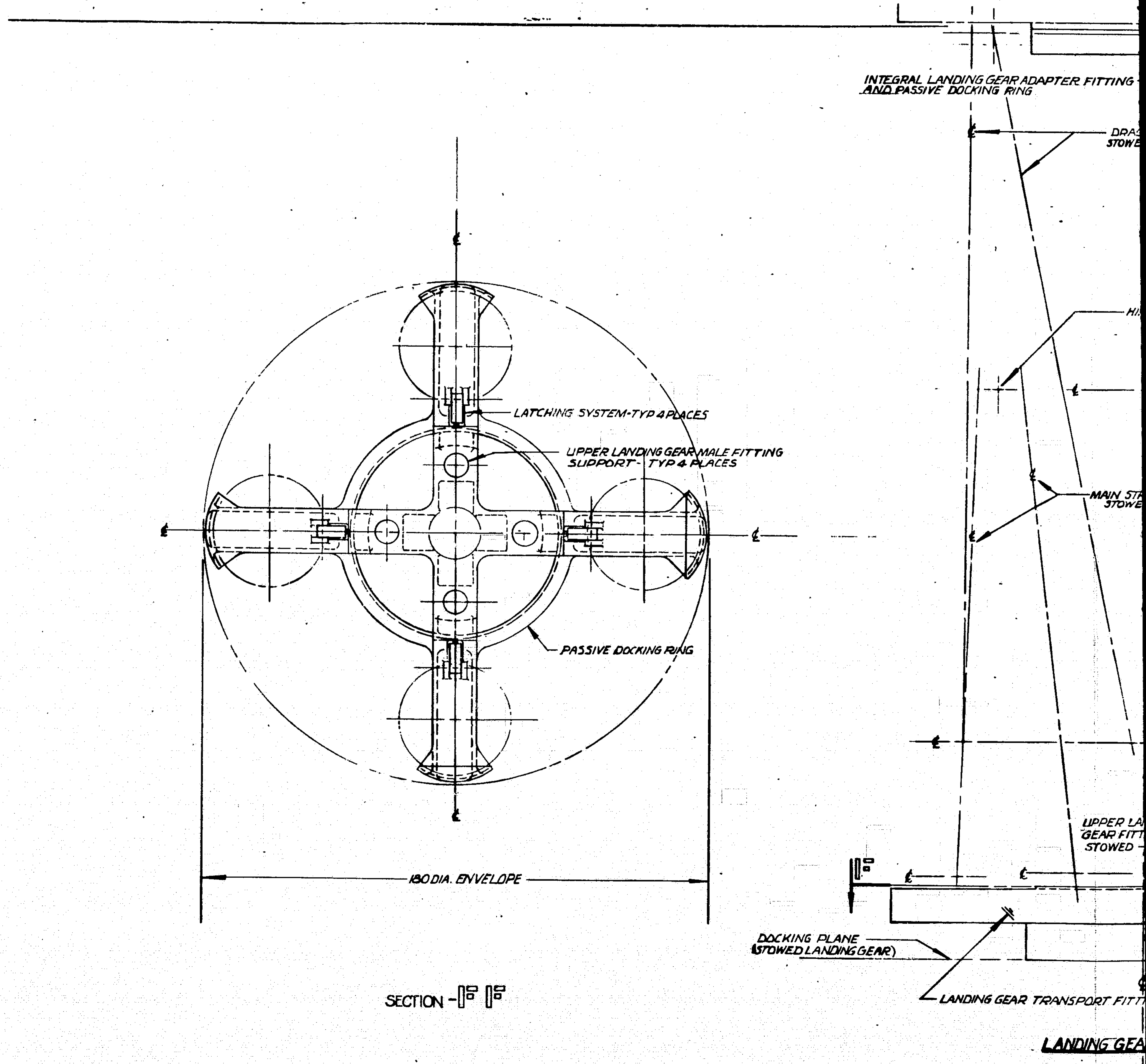


Figure 3-11. Landing Gear Kit



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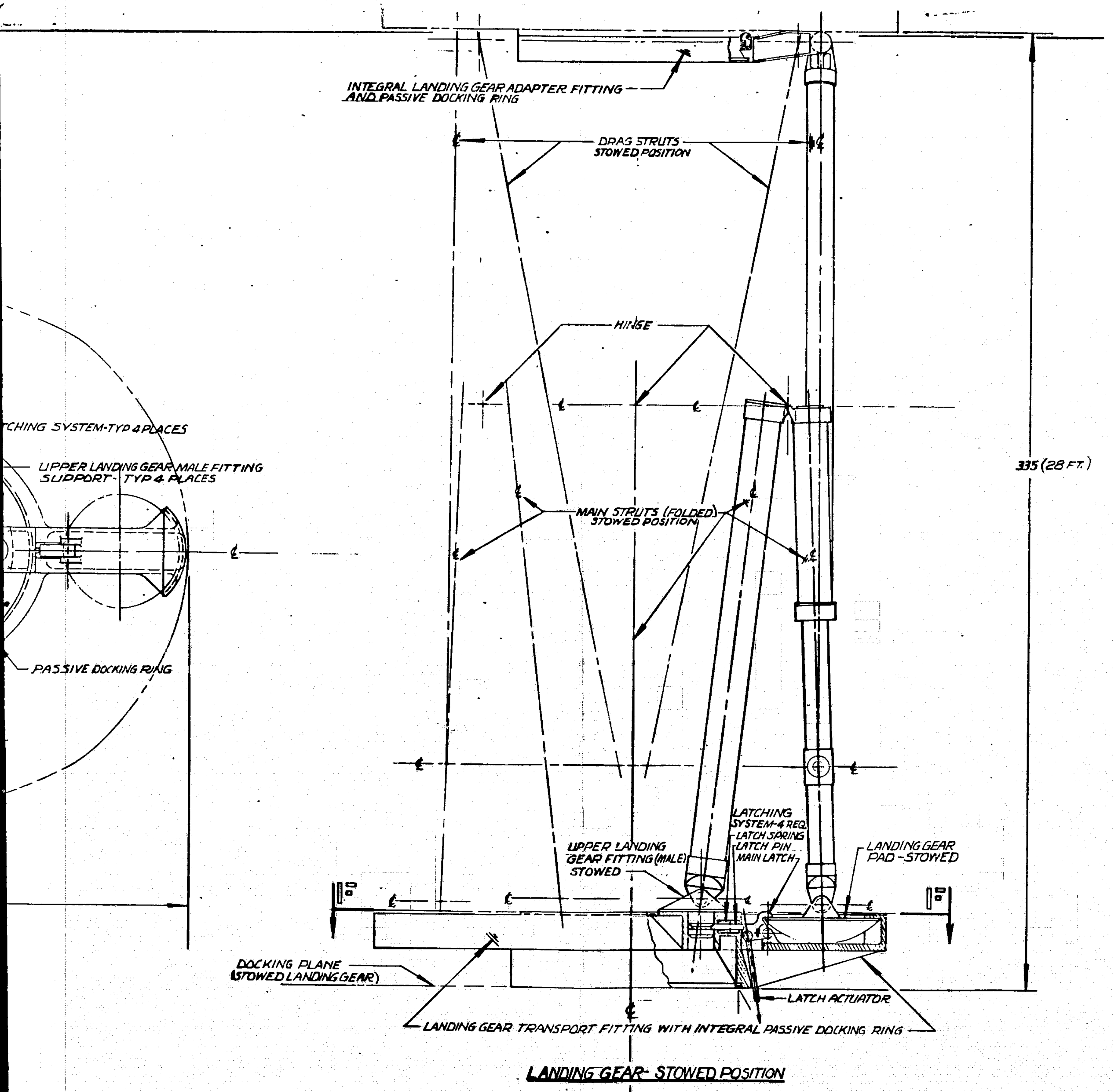

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Figure 3-11. Landing Gear Kit

kit which would provide the structural support and attach interface for the cargo pods or surface base modules. The support arms would be equipped with docking mechanisms to permit attachment of the modules by a tug. Each module or cargo pod would have a docking mechanism integrated into each end. The basic philosophy would be that the support arms on the landing gear kit would be equipped with an active mechanism and the modules would have the passive docking system. Modules providing manned entry probably would be equipped with a space station-compatible neuter docking system, and the cargo pods would be provided with Apollo drogue-type systems, primarily from weight considerations. It was felt that the high weight of the neuter docking system could only be accepted where manned passage through the docking interface was required. In all others the smaller and lighter Apollo probe and drogue system would be utilized.

3.4.4 HORIZONTAL LANDER CONCEPTS

During the early phase of the parametric configurational studies, a concerted effort was made to explore tug lander configurations that would be essentially horizontal in attitude in the lander mode. The basic objective was to place the crew module close to the surface to maximize crew visual assessment of the landing site, promote ease of surface access, and provide a low c. g. position. Two configuration arrangements were developed in this study and are shown on Figure 3-12 and Figure 2-10.

The first concept shown on Figure 3-12 represents the initial effort and was primarily concerned with developing a feasible packaging concept for the propellants, engines, avionics equipment, and crew module. The 15-foot (4. 57-meter) vehicle has been sized for a nominal propellant load of 60, 000 pounds (27, 000 kg) of LH_2/LO_2. To eliminate longitudinal cg shift from full to empty propellant condition, the propellant tank configuration has been made symmetrical about the cg. The LH_2 is accommodated in two elliptical ended short cylindrical tanks each with a volume of 2378 cubic feet (66. 5 cubic meters). To provide a stowage space for the two RL-10 sized engines, the LOX was accommodated in three cylindrical tanks positioned longitudinally to be symmetrical about the vehicle cg. The engines are stowed in the longitudinal space above the two lower LOX tanks and hinged to deploy out and then rotate 90 degrees downward into the lander operating position. For normal orbital tug operations the engines would remain in the initially deployed position with the thrust axis pointing aft. Because of the mid-length location of the engines and their proximity to the vehicle body, the outer surface of the aft body would require extensive thermal protection. A four-legged landing gear is provided which is stowed internally forward and aft of the hydrogen tanks. The avionics compartment, with RCS nozzles, is positioned adjacent to the crew module. The CM is positioned at the forward end and features an active docking system. Crew flight control windows,

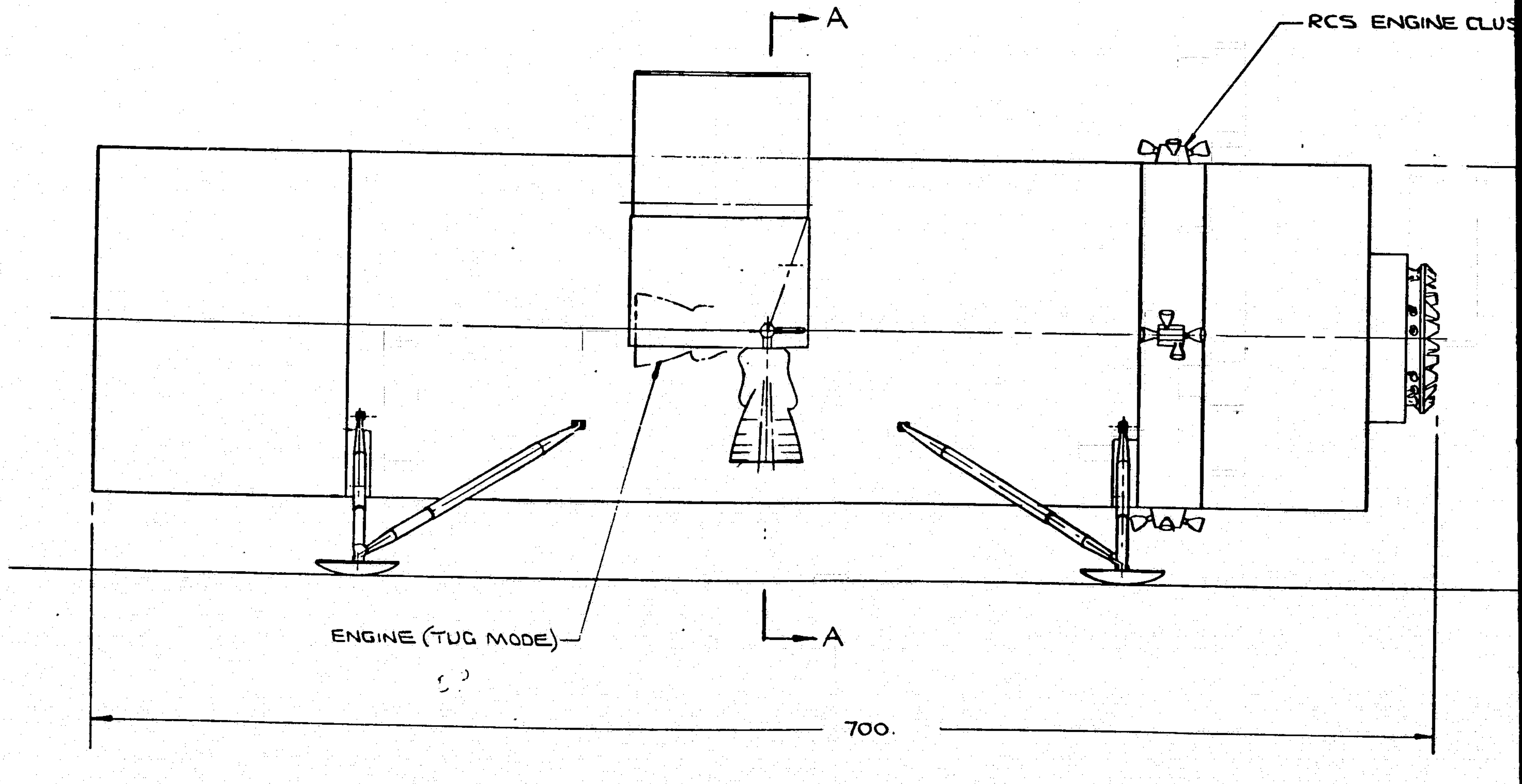

Figure 3-12. Lunar Lander, G
Horizontal Config

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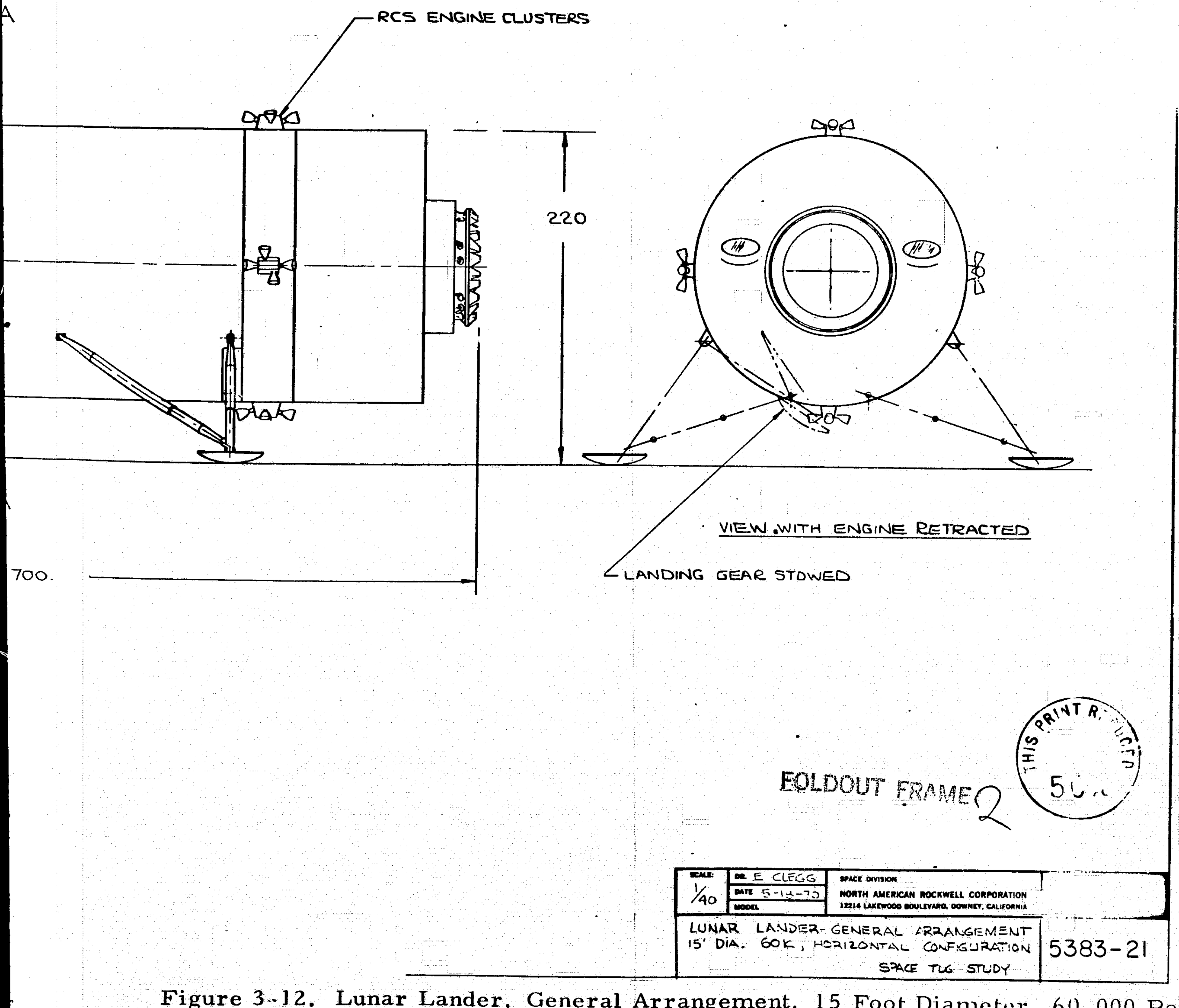
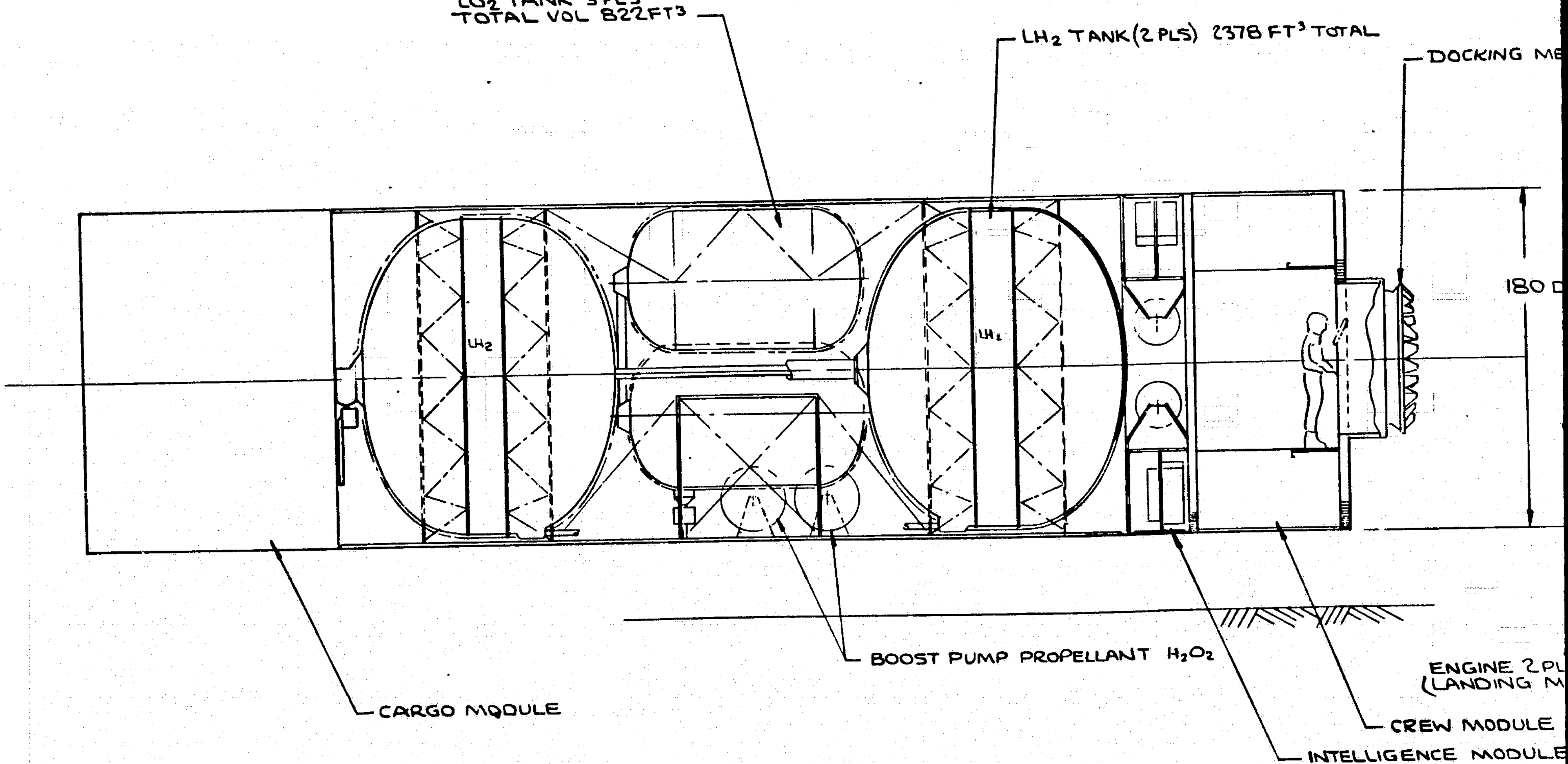


Figure 3-12. Lunar Lander, General Arrangement, 15 Foot Diameter, 60,000 Pound Horizontal Configuration



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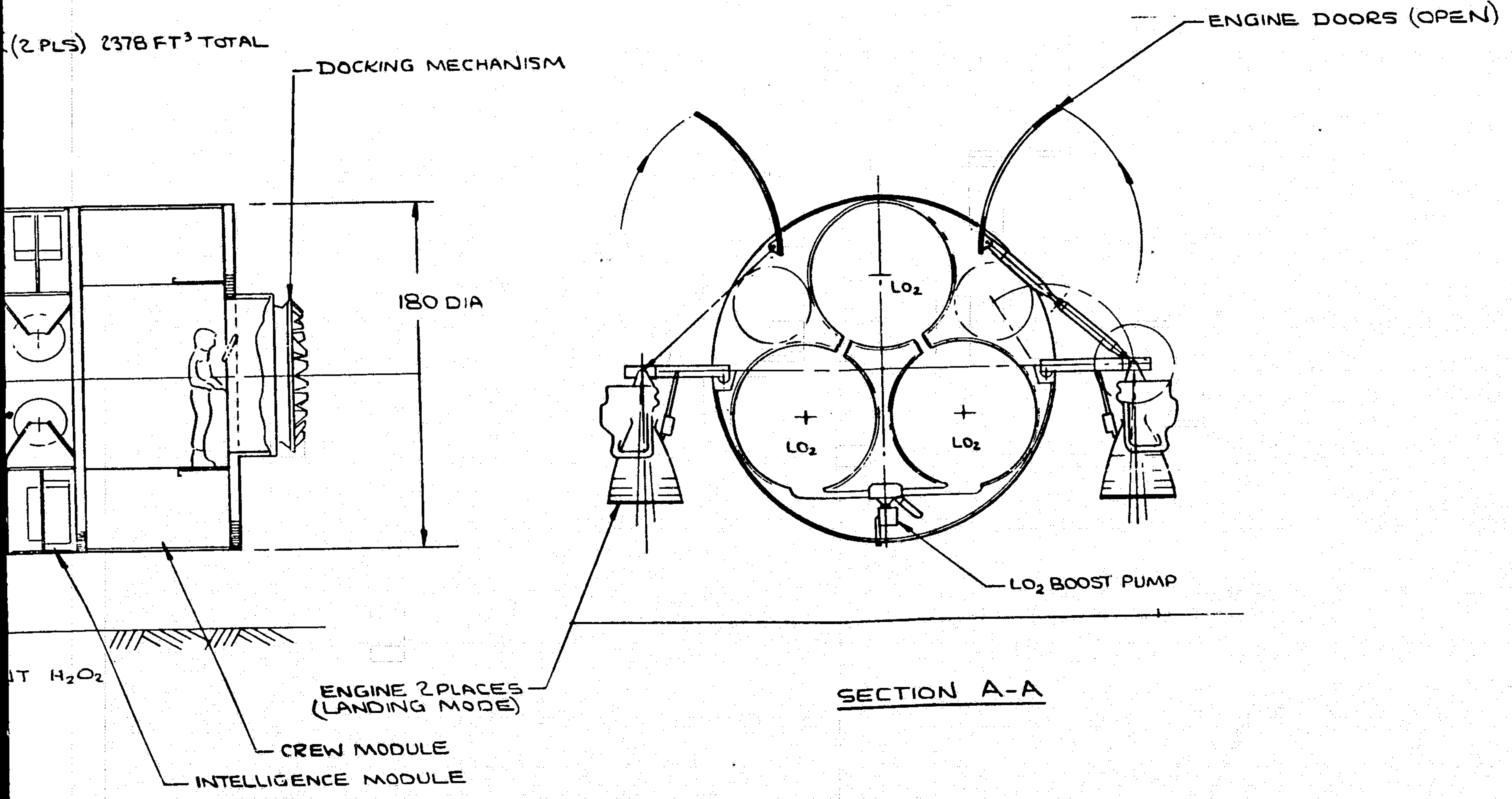

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Figure 3-12. Lunar Lander, General Arrangement, 15 Foot Diameter, 60,000 Poun
Horizontal Configuration

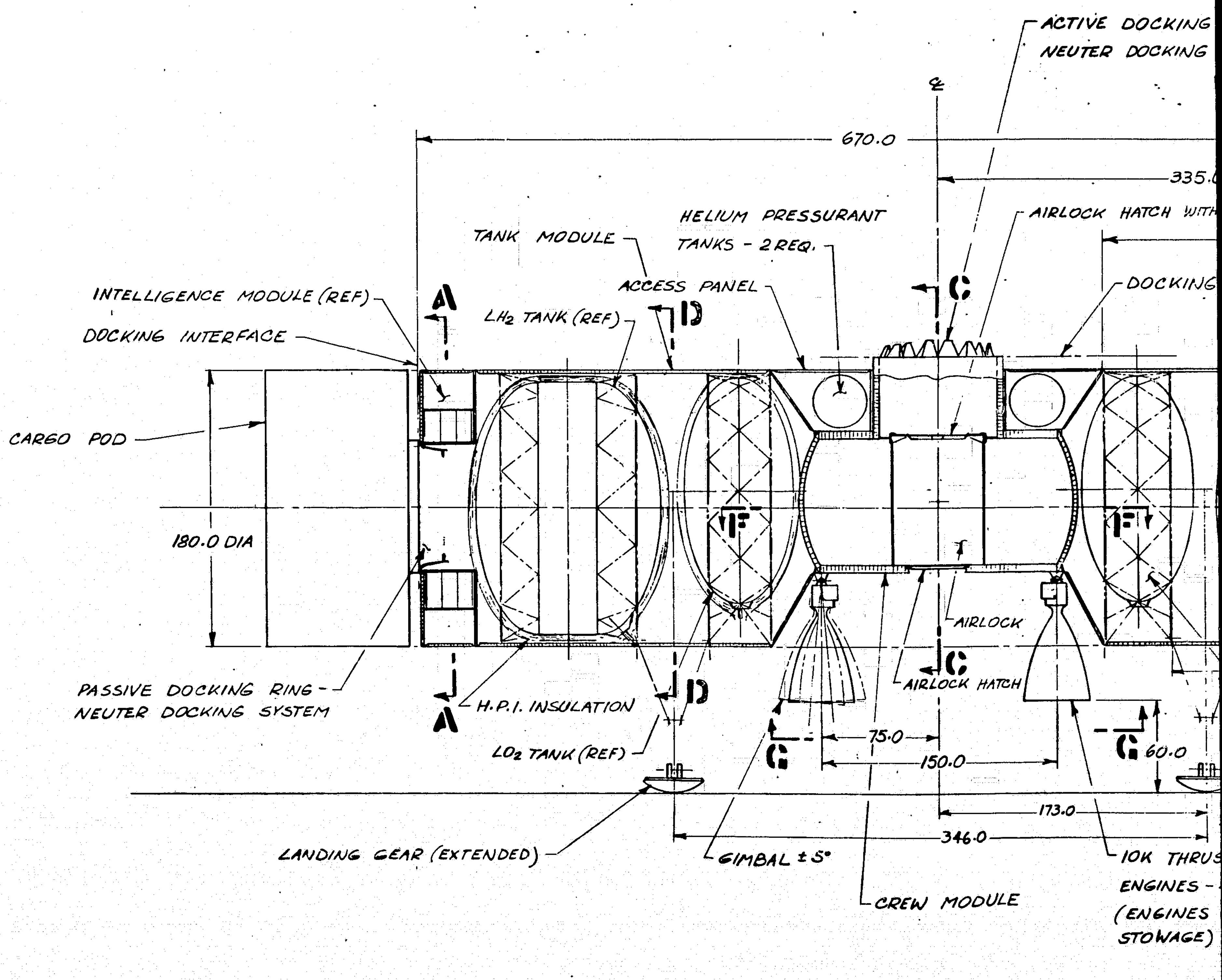
providing excellent ground visibility, are located in the forward bulkhead of the CM.

The major operational problem with this vehicle arrangement in the lander mode results from cg maintenance during the mission. The CAM is located at the aft end of the vehicle and must be basically maintained at the same weight as the avionics equipment module and CM to keep the cg coincident with the engine thrust vector. This could be a major problem in the case where no cargo is returned from the lunar surface to lunar orbit. Theoretically, additional RCS nozzles could be positioned at the forward end to compensate for cargo weight variations, but the required thrust level could be significant.

Subsequently, an alternative horizontal lander concept was configured that resolved some of the operational cg problems of the first concept. Figure 3-13 presents a four-engined horizontal lander sized to accommodate 60, 000 pounds (27, 200 kg) of usable propellant. The major configuration change is in the center section of the vehicle. The CM has been positioned laterally across the diameter of the vehicle, with the active docking system integrated into the upper bulkhead of the crew module. The four engines are attached to the lower bulkhead of the crew module and are hinged to fold into the diameter of the vehicle as shown in view of Figure 3-13. The four legged landing gear is stowed in the space between the LH_2 and LOX tank at each end. The avionics equipment is accommodated in two-ring sections with the deployable RCS nozzle clusters integrated into one of the modules. The major cg control problem has been alleviated by utilizing two cargo modules attached to the ends of the basic tug vehicle. Passive docking rings in the avionics modules permits orbital docking and attachment of the CAM. This would be necessary since the basic tug vehicle has an overall length of 55. 7 feet (17 meters) and must be accommodated in the 60 foot (18. 3 meters) shuttle payload compartment. The tug active docking system must be configured to be deployable in order to position the docking interface for enough outside of the tug mold line to comply with the nominal docking angular misalignment design conditions. Once the tug is deployed from the shuttle cargo bay, the tug docking system could be deployed and locked, subsequent retraction would not be necessary. Although the vehicle concept as shown here represents an improvement over the initial horizontal lander configuration, it does require that the cargo always be accommodated in two equal masses for lateral cg placement and control.

3. 4. 5 MINITUG CONCEPT

One of the tug operational derivatives identified as being of major interest was the minitug. Initial conceptual studies indicated that such a vehicle potentially could be derived from representative elements of a basic



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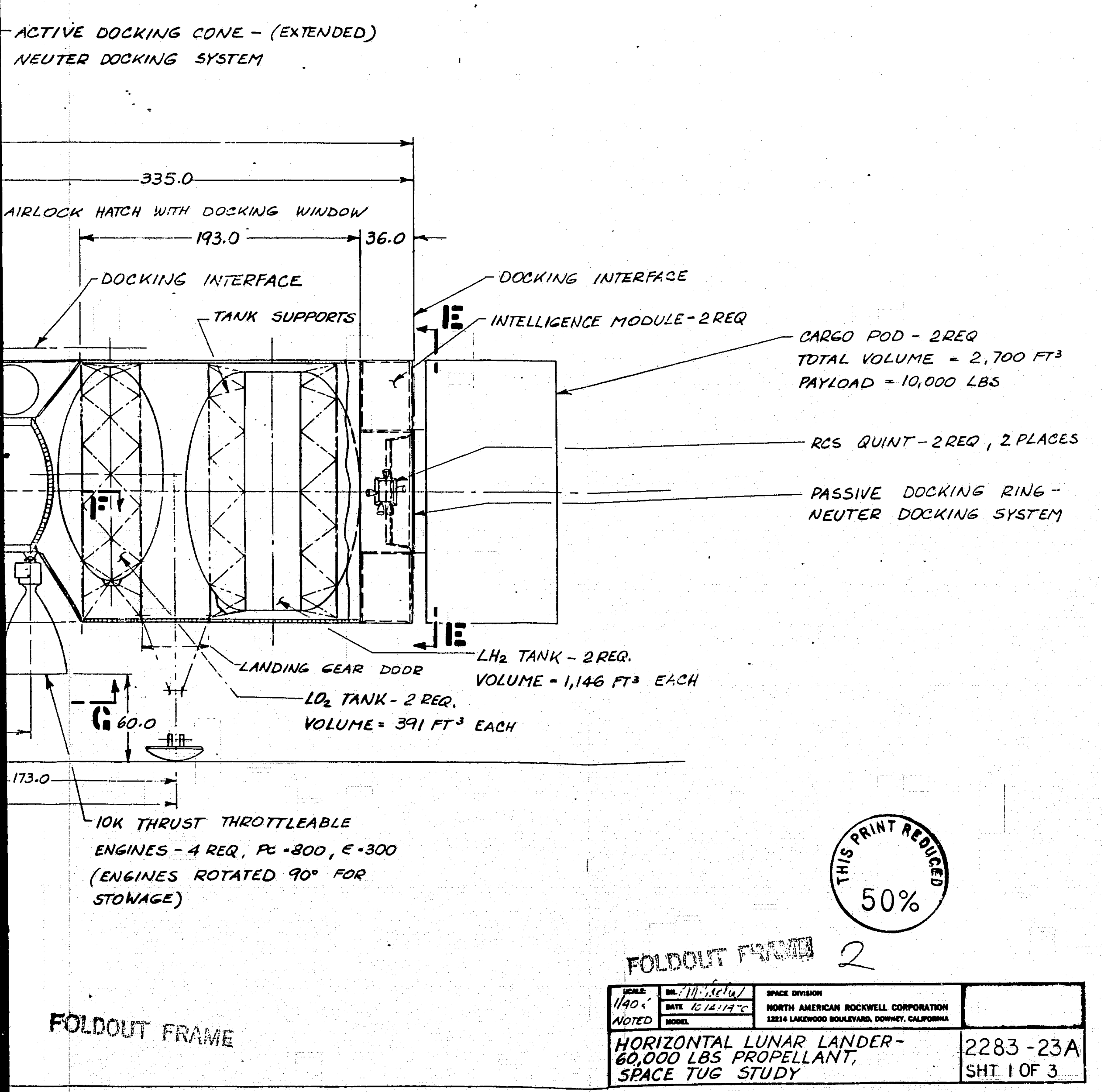
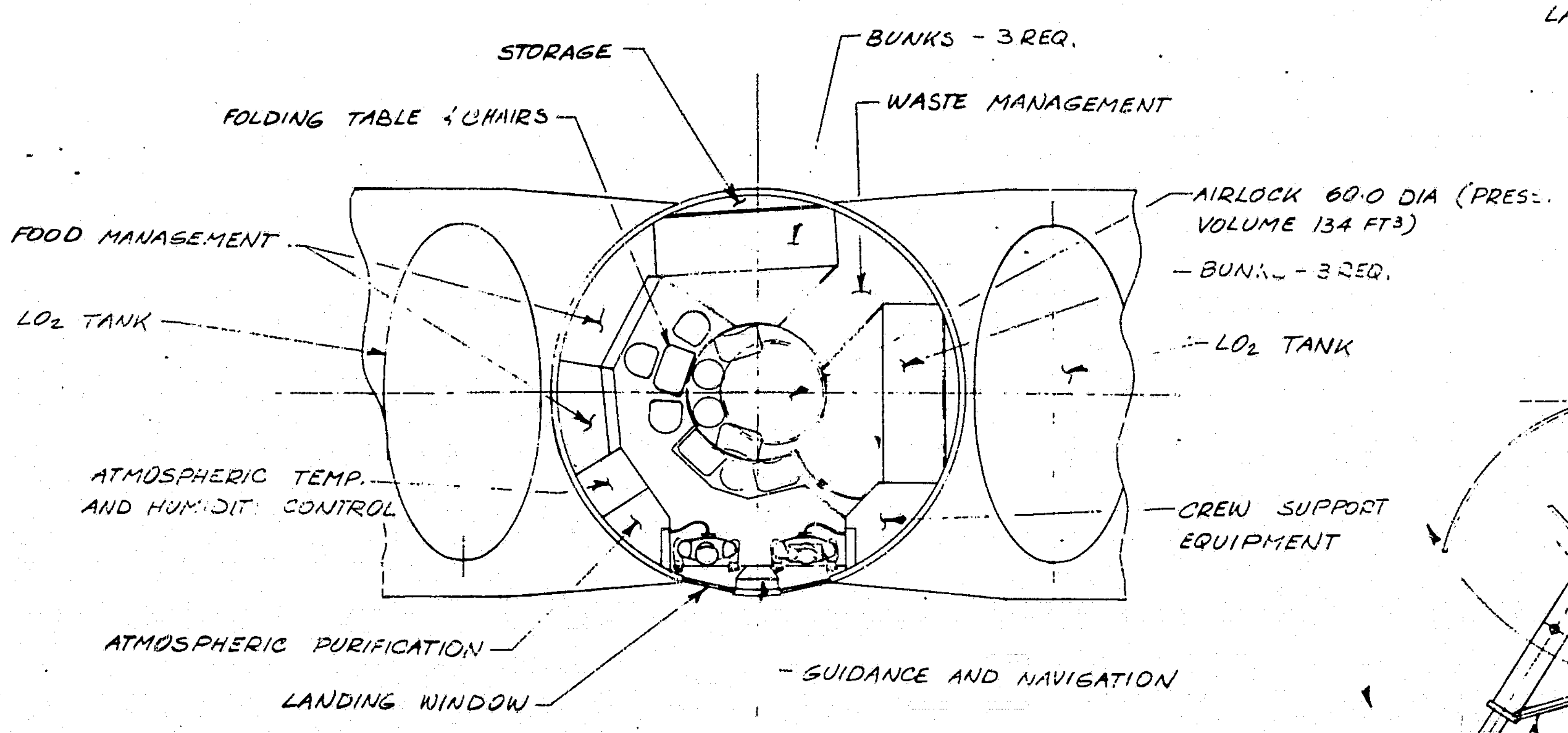


Figure 3-13. Horizontal Lunar Lander, 60,000 Pound Propellant



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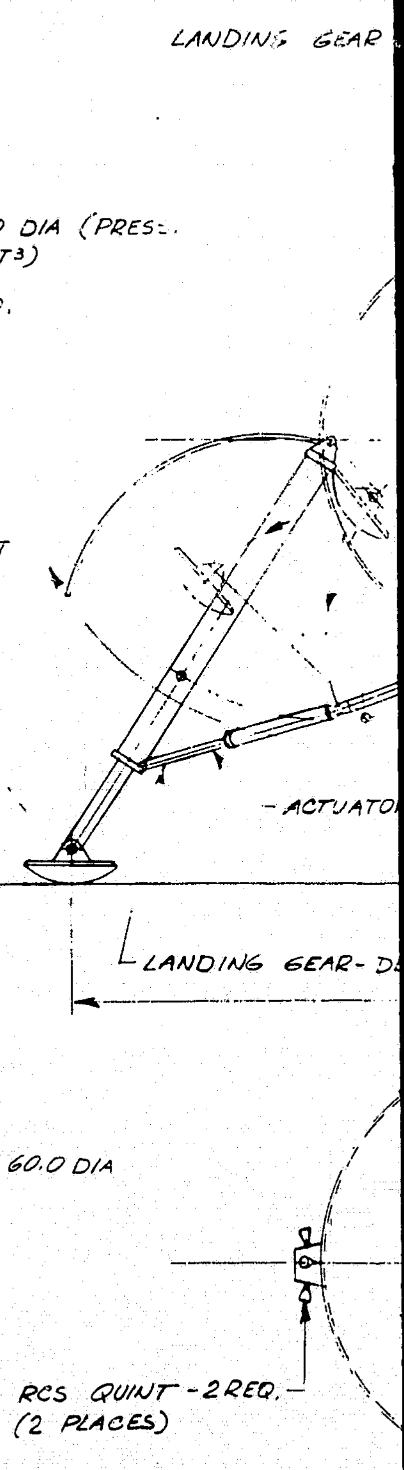

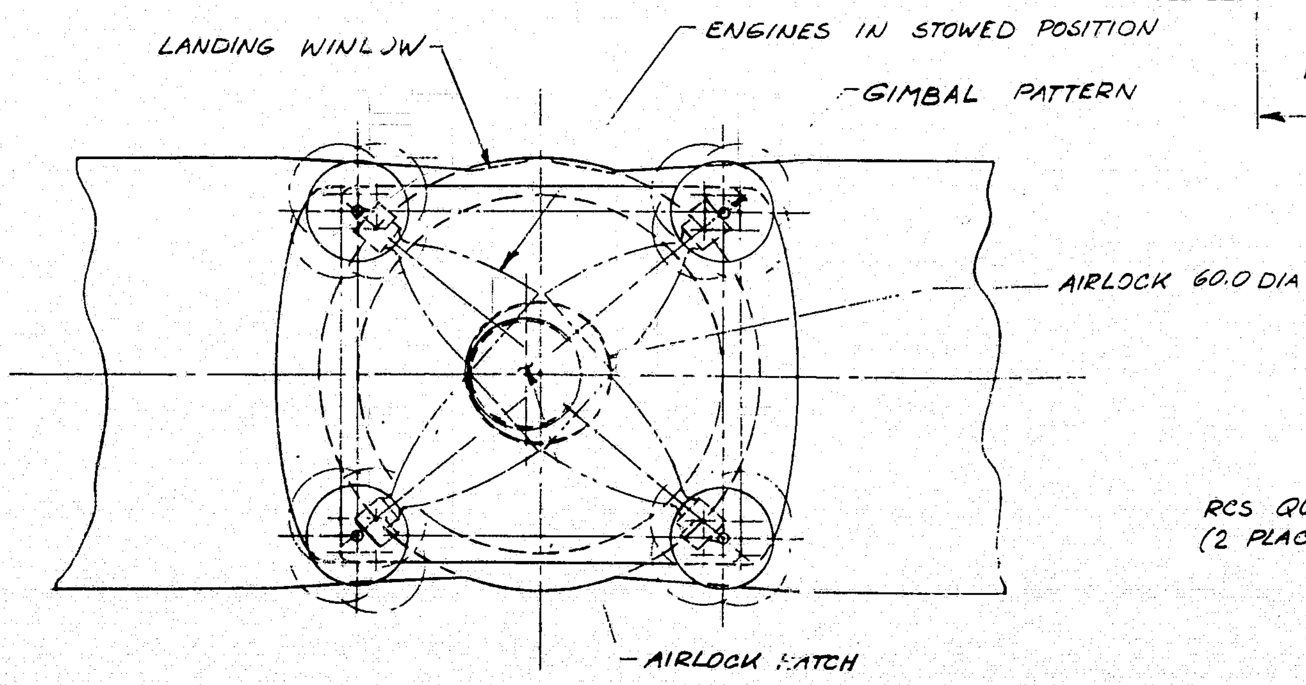

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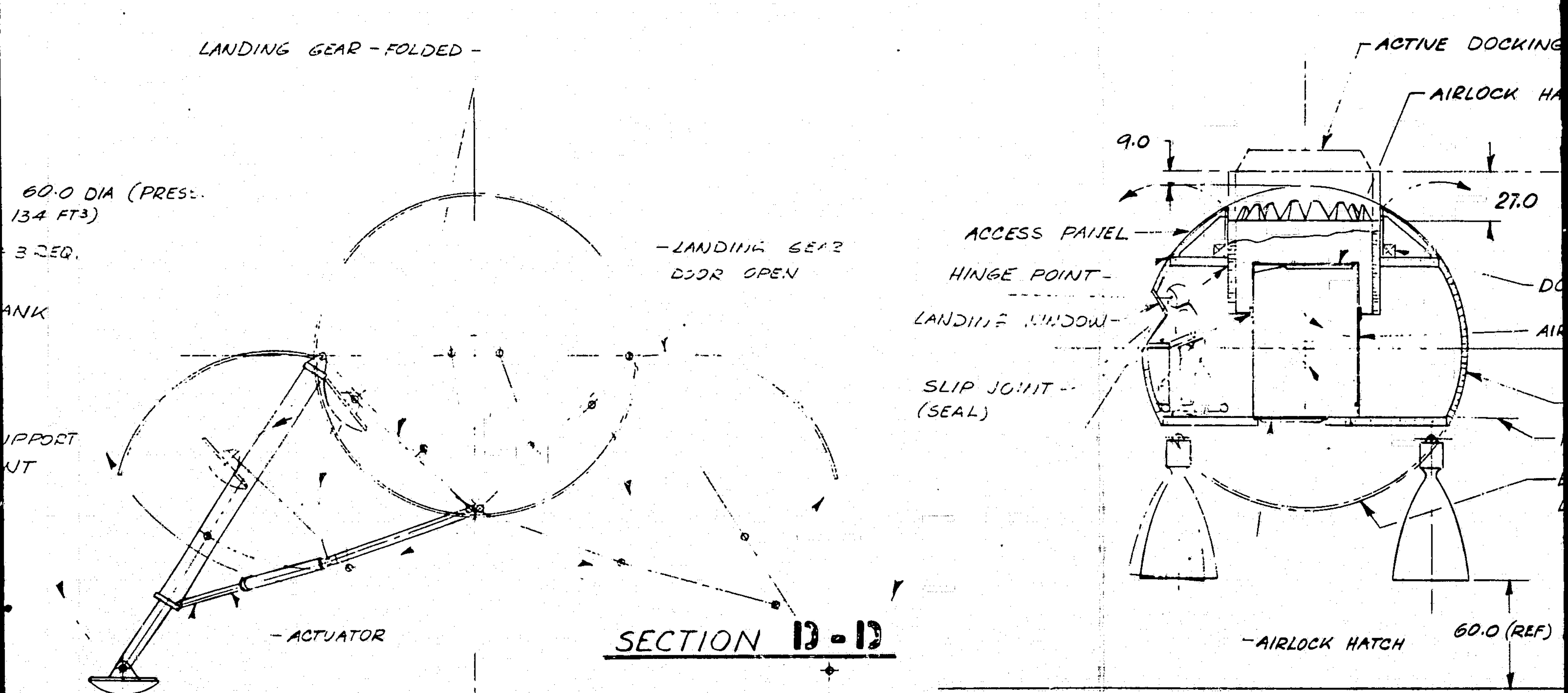

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Figure 3-13. Horizontal Lunar Lander, 6

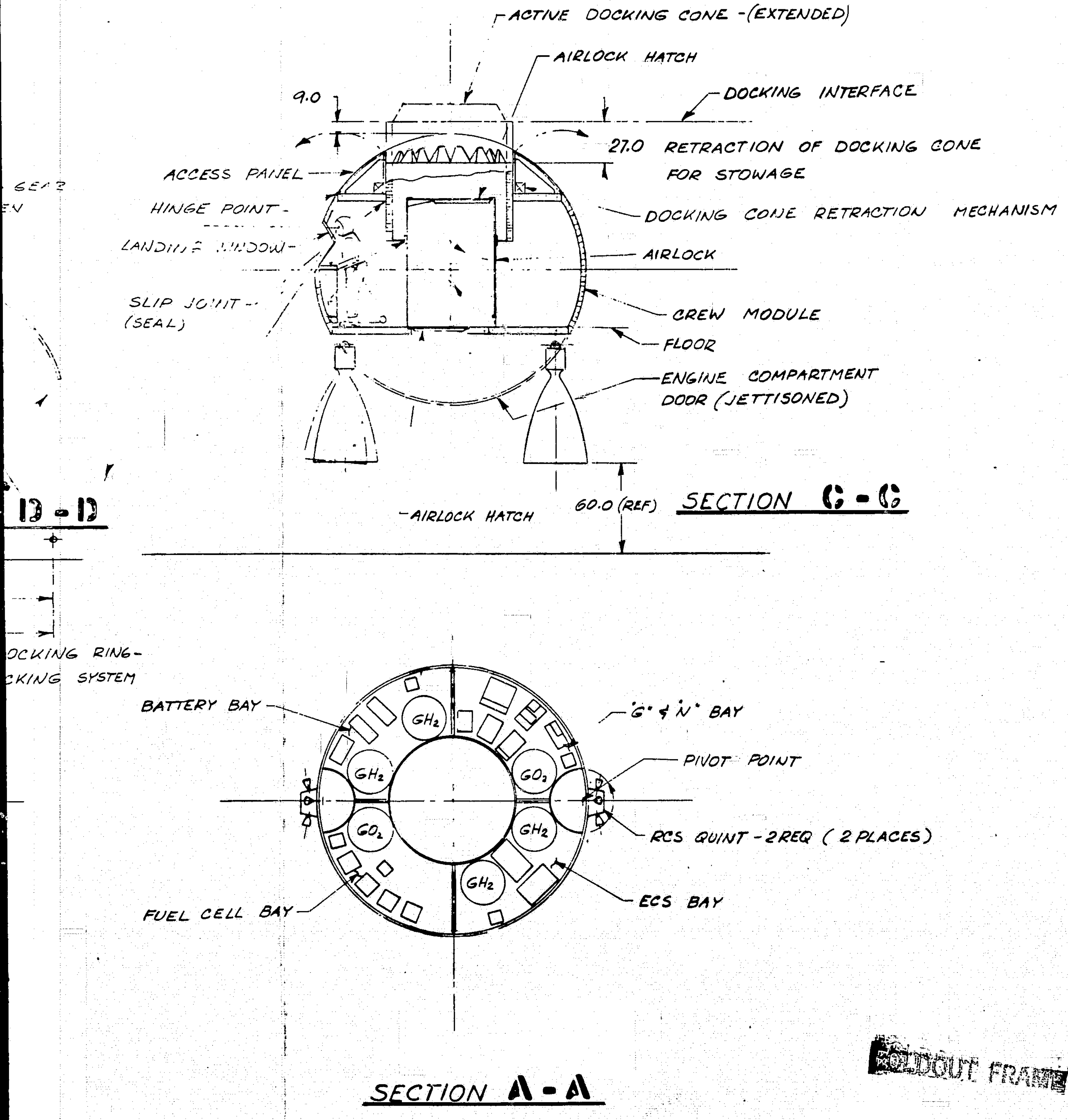

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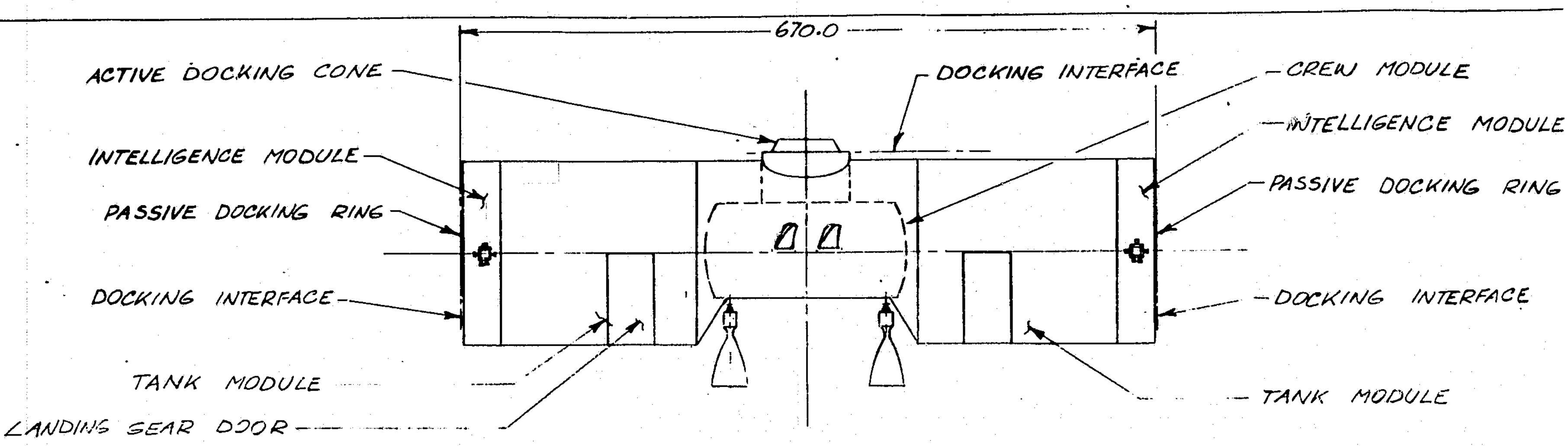
Figure 3-13. Horizontal Lunar Lander, 60,000 Pound Propellant



MANNED LOW EARTH ORBIT CONFIGURATION

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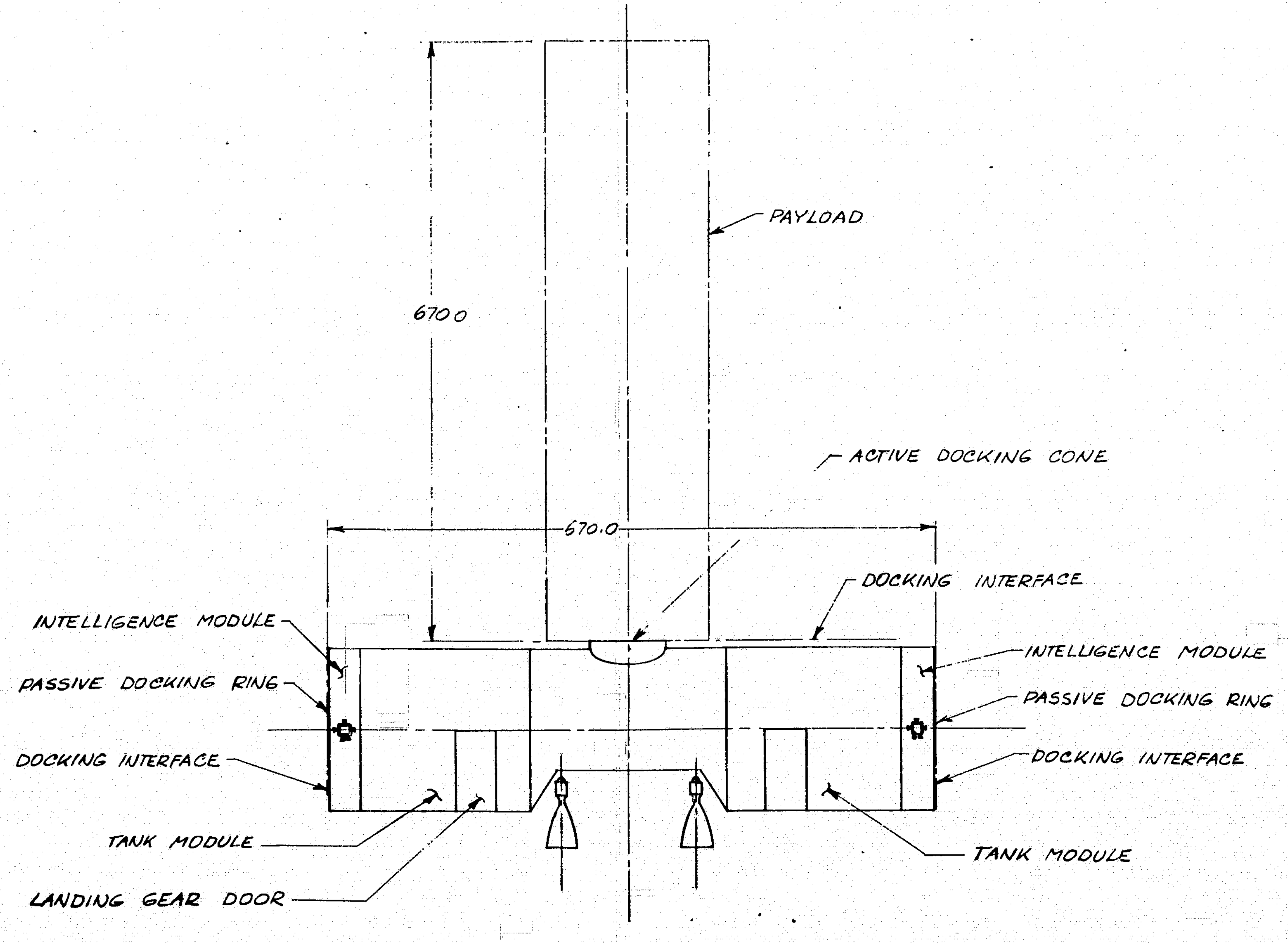


UNMANNED GEOSYNCHRONOUS ORBIT CONFIGURATION

SCALE: 1/100

2283-23
SHT 3 OF 3

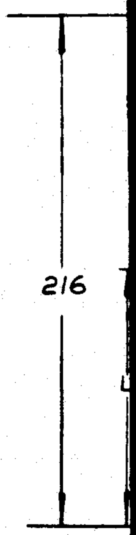
Figure 3-13. Horizontal Lunar Lander, 60,000 Pound Propellant

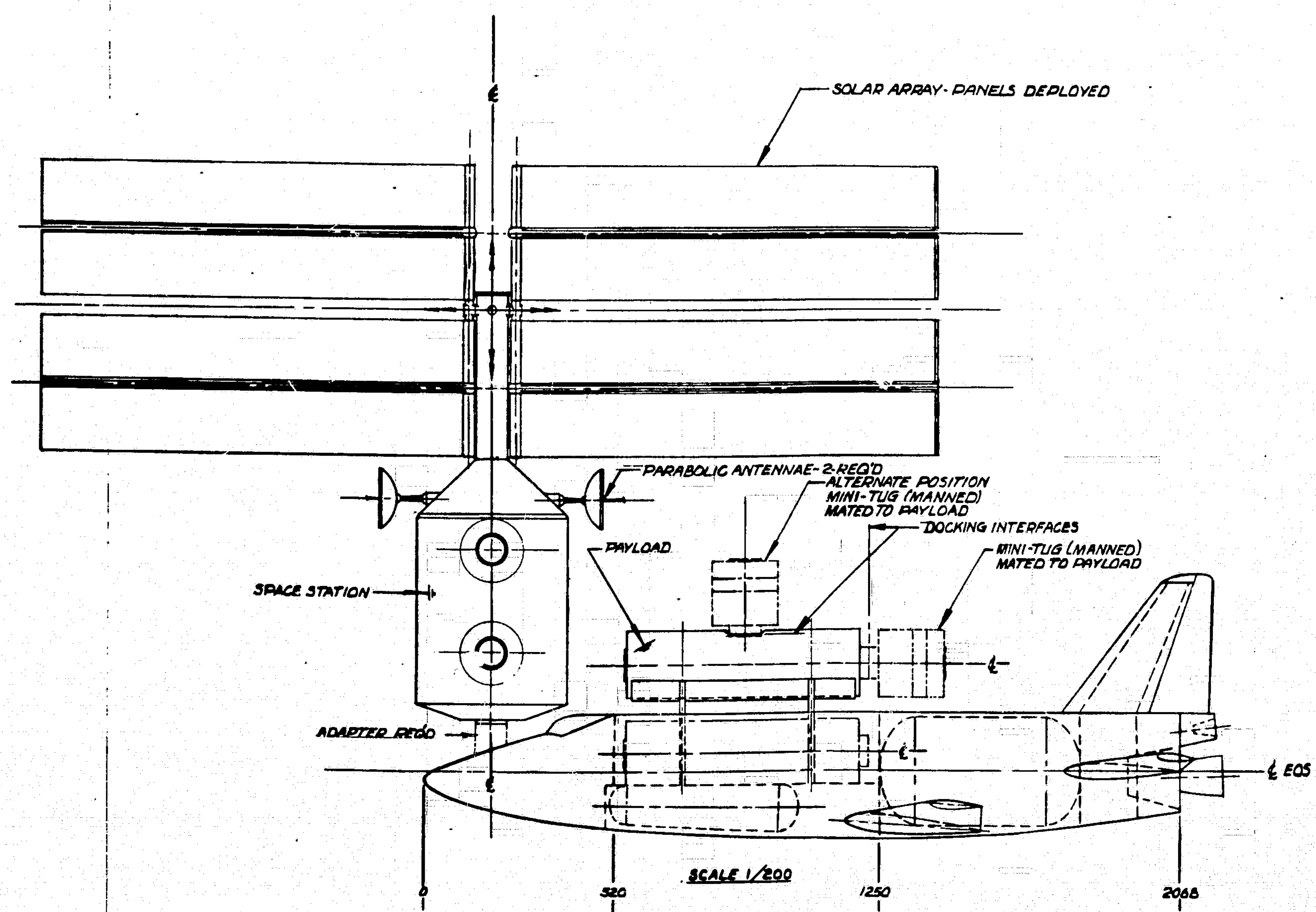
tug utilizing the modular approach. Figure 3-14 presents the major configurational features of this class of vehicle. The vehicle basically consists of the avionics or intelligence module with an auxiliary RCS propellant supply module (skirt module) attached to the aft interface. The existing RCS nozzle assemblies would be used for propulsion maneuvers of the minitug. The skirt module also would provide adequate surface for accommodation of the required radiator area.

Utilizing the basic propulsion module of the minitug, several configurational variations are possible. A manned minitug could be configured by the addition of a tug crew module. The CM would have a standard active docking system on the forward surface as shown for interfacing with orbital cargo or space station (or with appropriate docking adapters). The additional radiator area required for the manned configuration would be accommodated on the exterior surface of the CM. In addition to performing normal tug operations, consideration has been given to the capability of the minitug to perform other mechanical/assembly operations. This capability can be provided by the use of a manipulator module which can be docked to either the unmanned or manned tug by use of standard docking system. The manipulator module would feature active docking mechanisms at both ends to permit attachment of the manipulator module to the minitug and to permit the manipulator module to be attached to the space station or orbital propellant depot for temporary storage when not in use. The manipulator module would contain the manipulator arms, drive mechanisms and other support systems such as batteries, and the externally mounted TV cameras. More discussion of manipulator systems and concepts applicable to the tug are provided in Volume 5 of this final report.

On Figure 3-14 shows several orbital operation sequences which could effectively utilize the minitug. In these concepts, the minitug operating in the manned configuration would be used to transfer cargo modules from the shuttle to the space station. This operational concept is attractive in that it does not require the shuttle to necessarily dock to the station to effect cargo module transfer, nor does it require that the shuttle docking loads be accommodated by the shuttle payload. The minitug concept data has been furnished to the NR shuttle program personnel and has been used by them in logistics evaluation studies.

DIRECT DOCKING ~ CARGO TUG OPTION
(CONCEPT-2-D)





EOS DOCKED TO SPACE STATION ~ MINI-TUG ACTIVATED PAYLOAD TRANSFER

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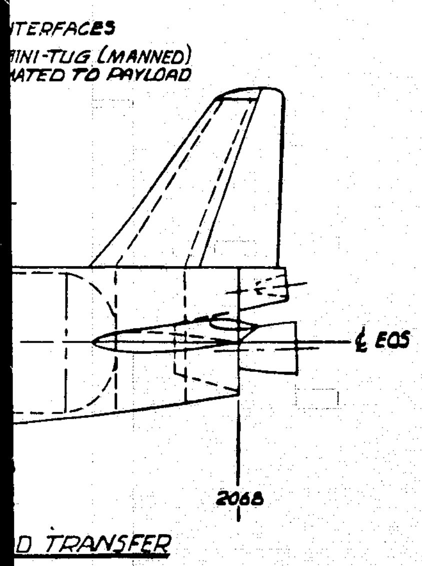

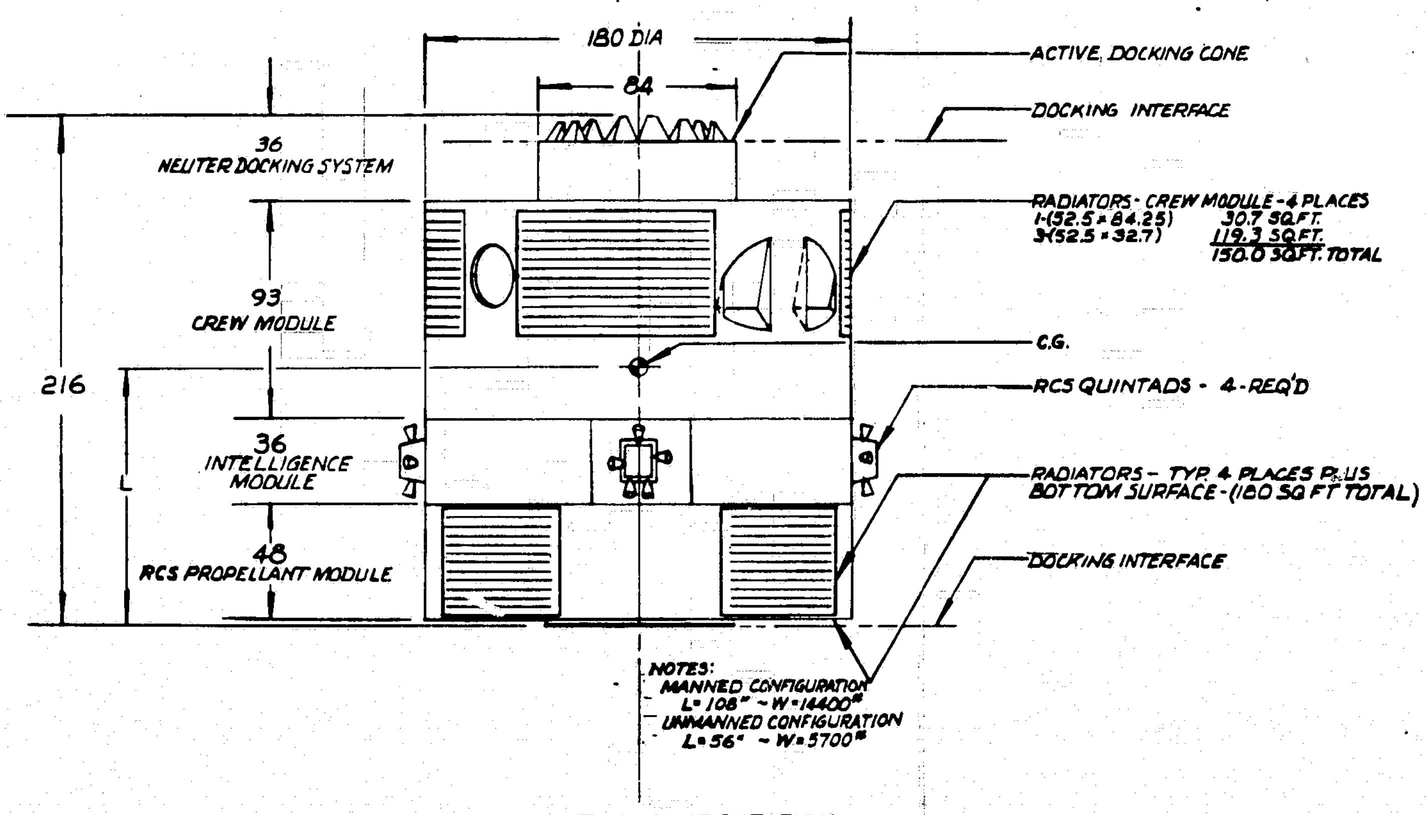

MINI-TUG CONFIGURATION
(MANNED)

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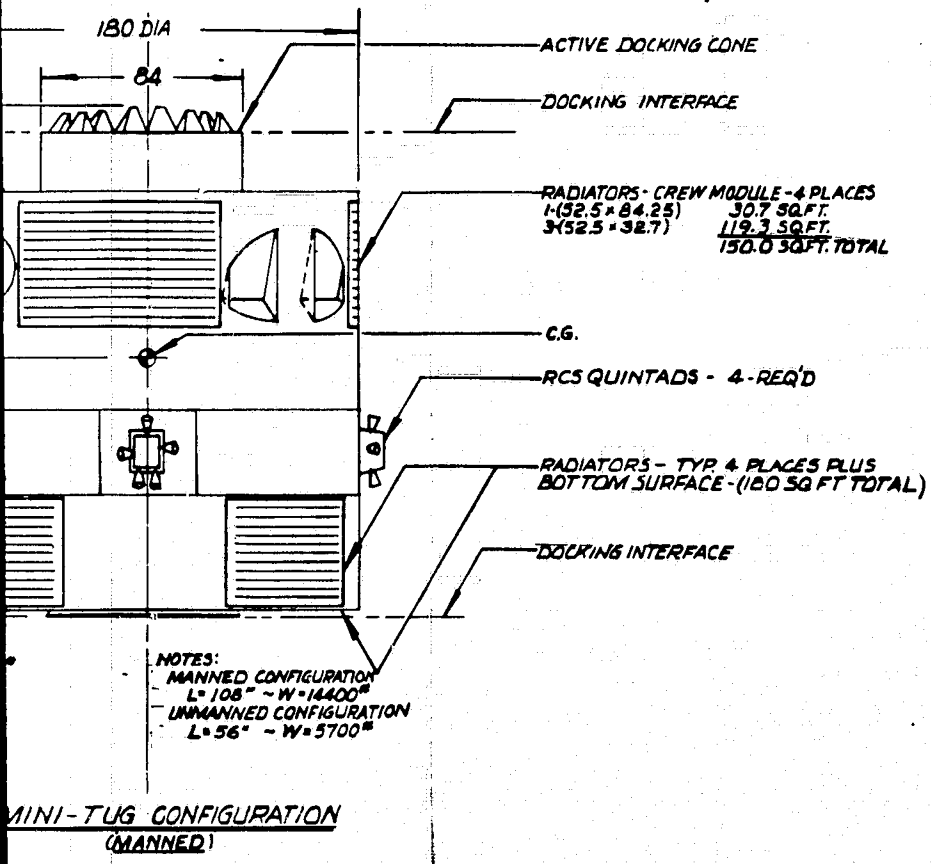
Figure 3-14. Minitug Con



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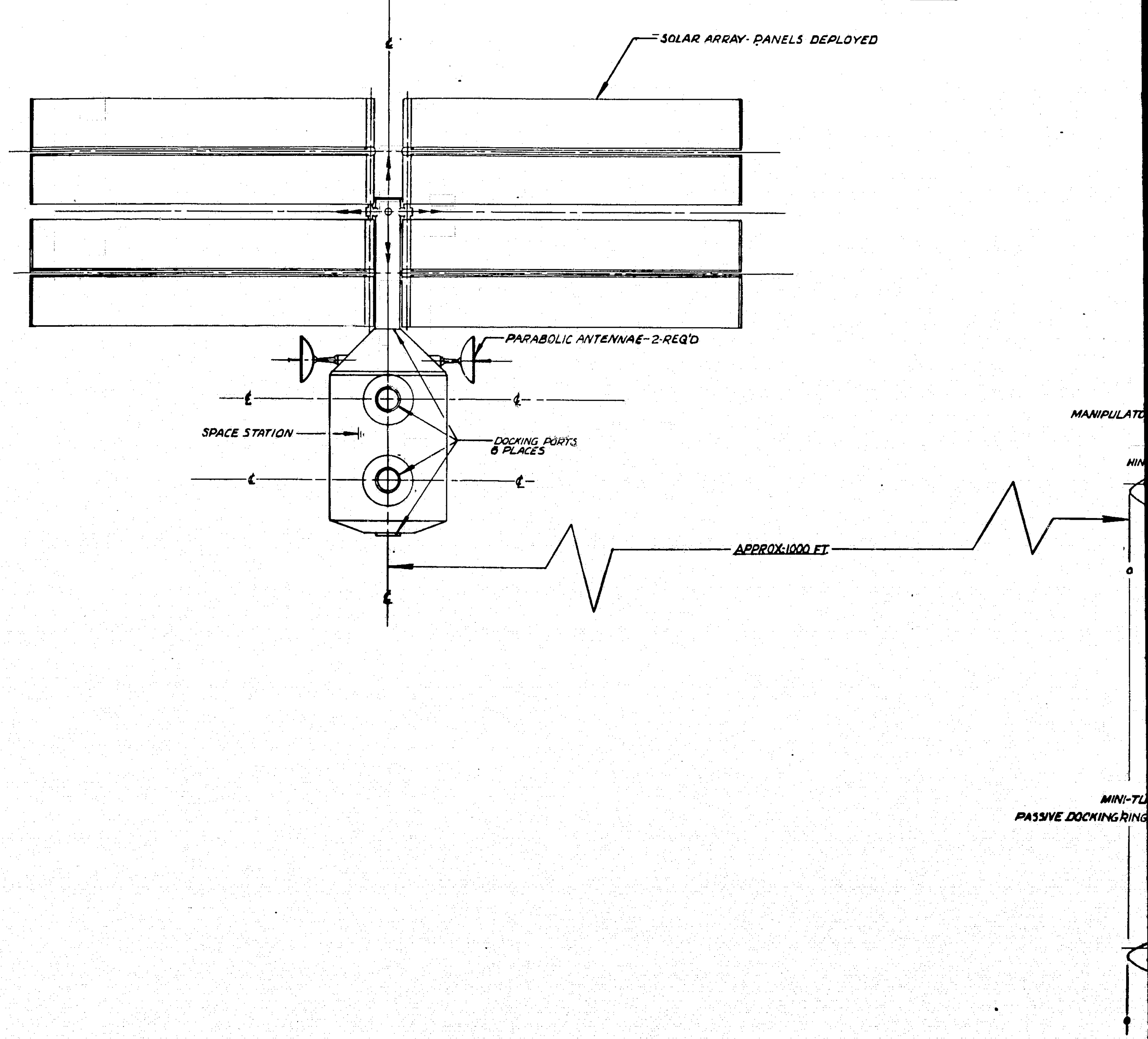
MINI-TUG CONFIGURATION SPACE TUG STUDY			2283-26 SHEET 1 OF 3

Figure 3-14. Minitug Configuration

STAND-OFF DOC.ING ~ CARGO TUG TRANSFE

(CONCEPT-3)



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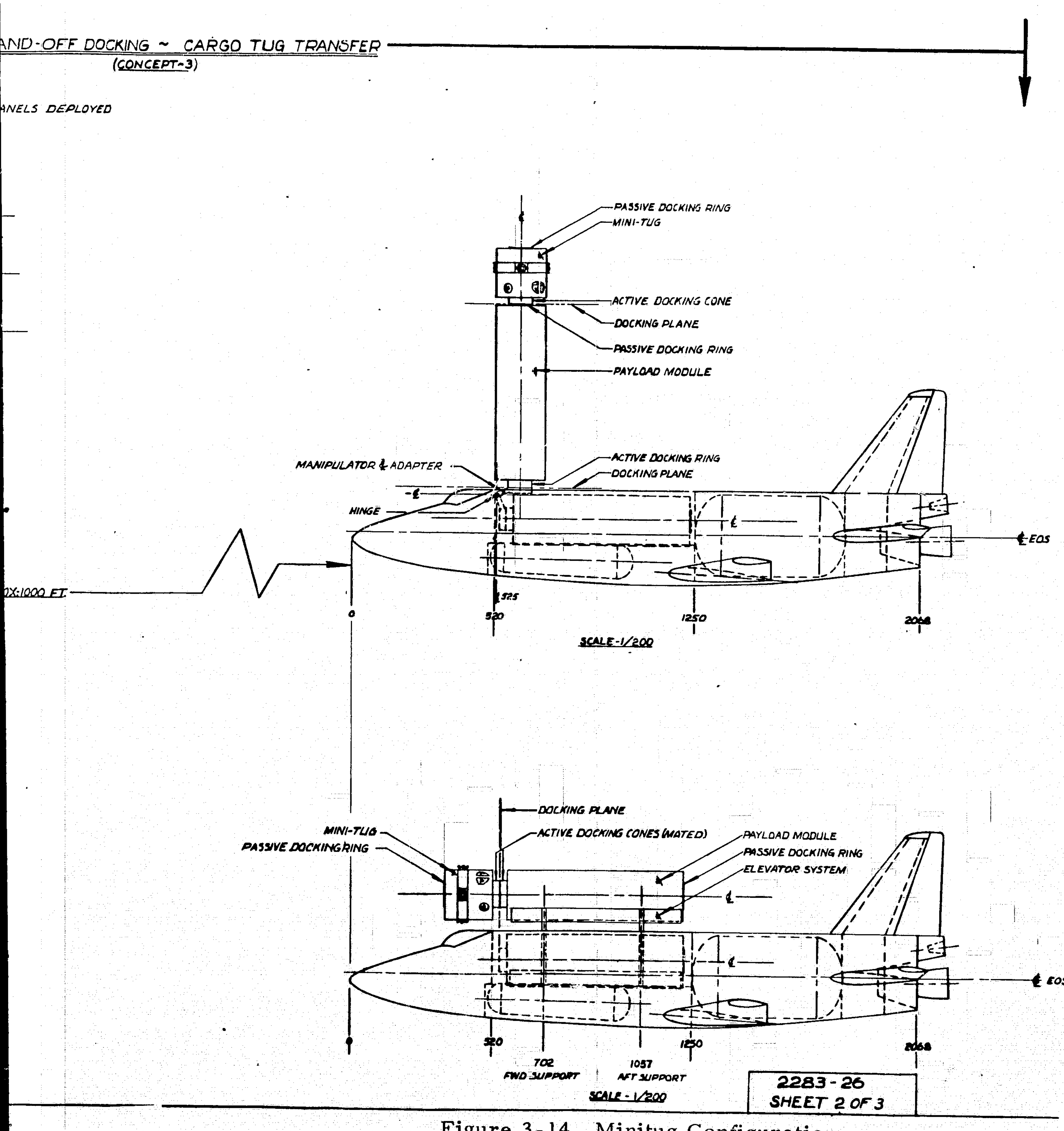


Figure 3-14. Minitug Configuration

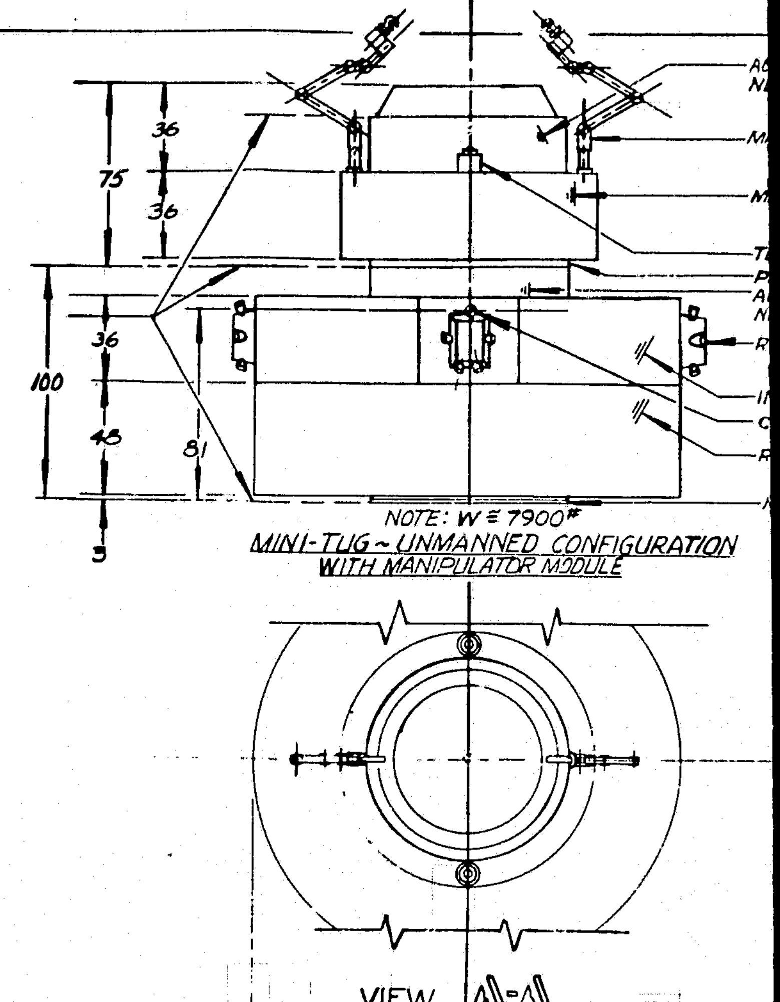


Figure 3-14. Minitug Configuration

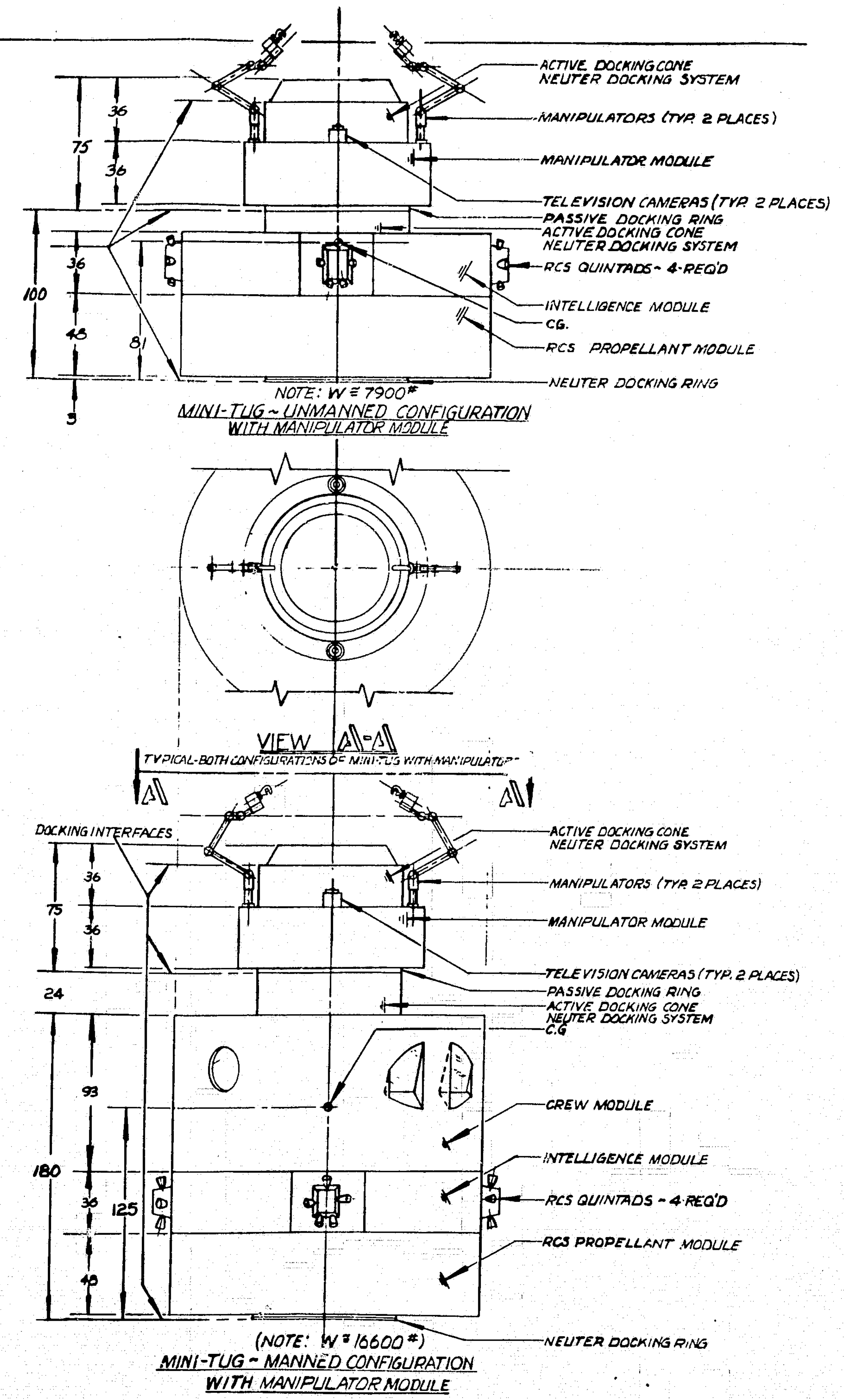


Figure 3-14. Minitug Configuration

3.5 DESIGN SENSITIVITY

3.5.1 SUMMARY DATA

Some choices among alternatives are unusually sensitive or significant in terms of their effect on design. Engines and propulsion parameters are examples. The type of engines, number, and design parameters such as area ratio, chamber pressure, and pump inlet characteristics have a substantial influence on the performance, weight, and the vehicle geometry, and these characteristics affect the propellant mass required to perform the design mission. Docking provisions and tank geometry also affect inert weights directly and in turn the vehicle geometry, such as overall length. Subsystems weights also have strong influence on concept weight and are drastically affected by the design and operational philosophy employed. For example, utilizing the tug as a ground-based concept rather than as a space-based system has the immediate potential of reducing the required level of redundancy in systems as a result of the good surveillance available on the ground between each mission. (The specific effects of variations in subsystems philosophy are explored in another volume of this report).

The estimated effects of variations in design and operations parameters on tug weight and geometry are summarized in Table 3-15. In this summary, it should be recognized that many of the variables shown are not truly independent. Examples are engines, tanks (adjacent to engines), and aft docking structure: these three variables are related and must be integrated to achieve a good design with minimum weight and length. However, for the purposes of this summary chart, an attempt has been made to isolate the individual variable effects insofar as possible to provide insight as to their influence on design. Weight and size increments shown are related to the baseline designs designs described previously and generally are referred to the single-stage recoverable Concept 1. In many cases, the exact value of the indicated weight change is not as important as the sense (plus vs minus weight change), in keeping with the uncertainty in the design details which should be expected in a Prephase A study. However, the direction of the indicated changes are believed to be reasonably certain. More specific weight relationships for the important area of base design where engines, docking, tanks, and structure must be considered together, are shown in Table 3-16.

Table 3-15. Tug Design Sensitivities Summary

Design Options*	Effect On*		Secondary Effects	Qualifying Remarks
	Weight	Size		
Propulsion Module				
2 Engines	-250	Slightly longer	Tends to favor 2 LOX tanks	Possible higher ISP, but less redundancy
1 Engine	-260	Slightly longer	Flts with multiple tanks, requires special docking	Highest performance, no redund.
Aerospike eng	-210	Shorter vehicle	Permits center docking	Lower ISP probable, devel risk
Retractable nozzle	-270	Shorter vehicle	Shorter launch stack in EOS	More complex engine
Various tankage arrang	-230	Variable-1, 2, 4 LOX tanks	Related to concept arrgt	Affected by load paths, docking, eng config
Separate aux propel	-150	Nil.	EPS, EC/LSS, RCS Interfaces	More complex logistics resupply
Grd based insulation		Nil	Outgassing, reentry venting	More complex insul design
Total length change	68#/ft	(ΔW_p - 2300#/ft)		Part of integration: Eng/tank/docking
Intelligence module				
4-Eng RCS cluster	-50	No change	Less vernier thrust	Reduced redundancy (if RCS on c.g.)
Mono N_2H_4 RCS	+450	None	Add separate tankage	Logistics
Storable RCS	+150	None	Add separate tankage	Exhaust contamination prob., logistics
Pressurized IM	+50	No change	Implies interior access	Permits manned access in space
Integrated IM funct	-400	Reduced Length (3 ft)	No separate IM, reduced propellants	Equip in PM close to propel. sources

Table 3-15. Tug Design Sensitivities Summary (Cont)

Design Options*	Effect On*		Secondary Effects	Qualifying Remarks
	Weight	Size		
Partly integ IM	-200	Reduced (avionics only)	Good packaging split	Electro-mech in PM; electronics in IM
Single-purpose, Non-autonomous IM	-400	None	Add separate tankage	Logistics
			Reduced propellants, less independence	Reduced flexibility-non-modular, requires ground support
Ground-based concept	-1300	Nil.	Permits less sys redund	Return to Earth each mission
Battery Power (36 hr)	+500	Nil	Simpler; affects PM, PCS, ECS/LSS interfaces; less radiators	Good for very short missions
Battery power (18 hr)		Nil		Good for expendable concept
Crew Module				
Horiz cyl config	+200	12 ft dia x 15 ft	More difficult to stack/integrate	Possibly better lunar shelter
2 man use/7 days	-300	None	Reduced expendables	Off-loaded 6-man CM
12 man/2 days rescue	NEGL	None	Added crew restraints	On-loaded 6-man CM

*Rel to baseline config.

Table 3-16. Engine-Docking-Tanks Configuration-Weight Sensitivity

	Base Docking	LO_2 Tanks	ΔWeight	ΔLength
(Baseline): (4) Reference	(yes)	(4)	(0)	(0)
2	yes	2	-260 lb (-118kg)	+2 ft (0.6M)
2	yes	1	-250 lb (-113kg)	+4 ft (1.2M)
2	no	2	-410 lb (-186kg)	+2 ft (0.6M)
2	no	1	-530 lb (-240kg)	+4 ft (1.2M)
1	no	4	-560 lb (-254kg)	+2 ft (0.6M)
1	no	1	-910 lb (-413kg)	+7 ft (2.1M)

3.5.2 PARAMETRIC DESIGN SENSITIVITY STUDY

Of primary interest in studying alternate vehicles is the effect of configuration variables on weights and geometry. A series of vehicles configured for the geosynchronous mission was compared in a parametric study to determine the weight and length increases due to changes in their basic design. To begin the study, a baseline vehicle was configured. All subsequent configurations were designed to the same groundrules and incorporated the various parameters in the study. The major parameters considered during this design synthesis process are summarized in Table 3-17. These key physical characteristics when selectively combined in the synthesis process provided a preliminary definition of vehicle length and the resultant length available for payload. They also permitted first order weights and center of gravity estimates and provided a basis for preliminary judgment relative to concept feasibility. The parameters considered various combinations of engines and tankage. One, two, and four engines of two types were incorporated. A conventional bell engine with a fixed and retractable nozzle as well as an aerospike, were considered. Single and multiple cylindrical, spherical, and elliptical tanks were included in the study.

To conduct a meaningful study, a simplified, lightweight baseline configuration was derived and deviations from this were then configured. The baseline vehicle was sized to deliver 10,000 pounds (4536 kg) to geosynchronous orbit and return to low earth orbit. This mission required a total of 70,000 pounds (31,751 kg) of propellant. The vehicle was made up of one LH_2 tank with elliptical bulkheads, a single spherical LOX tank and a single fixed nozzle conventional bell engine. This vehicle is 41 feet (12.4 meters) long and is shown as Concept A on Figure 3-15. Each of the subsequent vehicles shown on the figure incorporates a single LH_2 tank with elliptical bulkheads (1.4:1 ratio). They also are sized to accommodate 70,000 pounds (31,751 kg) of propellant. The weight shown next to each concept is the amount of propellant required by the particular configuration to complete the same mission as Concept A: 10,000 pounds (4536 kg) to geosynchronous orbit. The overall length of the vehicles is shown and is the length required to accommodate more or less propellant than the baseline vehicle. Since the tankage in each concept is sized for 70,000 pounds (31,751 kg) of propellant, the shaded area in each concept indicates this increase or decrease in length. The loaded and empty center of gravity locations for each concept also are indicated.

The concepts are divided into seven categories which compare engine number and LOX tank number and shapes. Categories I, II, and III utilized single engine concepts with single, multiple, and toroidal LOX tanks, respectively. Categories IV and V incorporated two and four engines, respectively, with single and multiple LOX tanks. A revised order of propellant tanks, LOX tank forward) was investigated in Category VI. The last category (VII) incorporated the aerospike engine with single LOX tanks. It

Table 3-17. Major Parameters Summary

ENGINE	PROPELLANT TANKS
Type	Number
Conventional bell	Single
Fixed skirt	Multiple
Retractable skirt	Arrangement
Aerospike	Shape
Number	Cylindrical
1, 2, 4	Spherical
	Elliptical

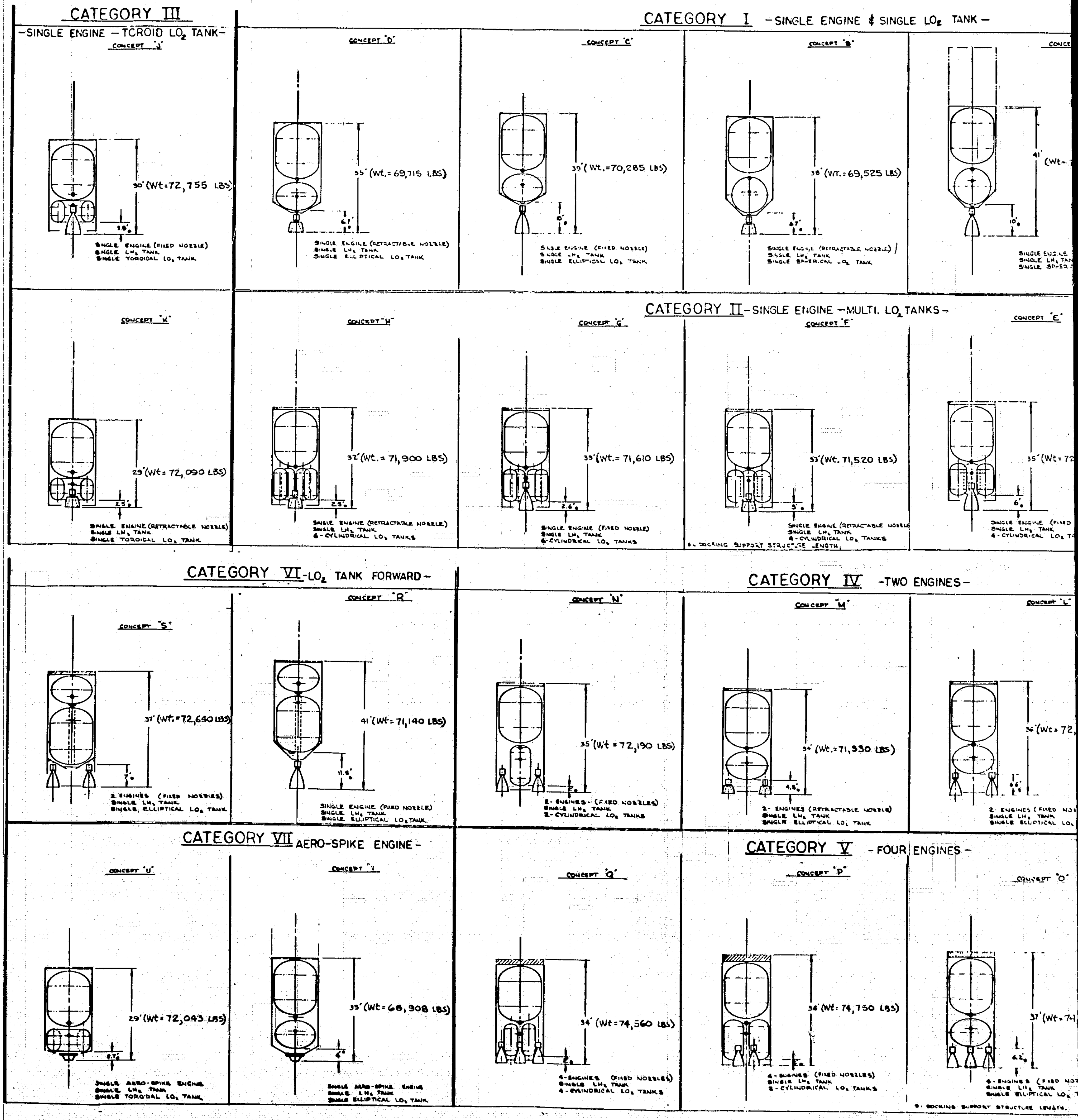


Figure 3-15. Parametric Design Configuration Study

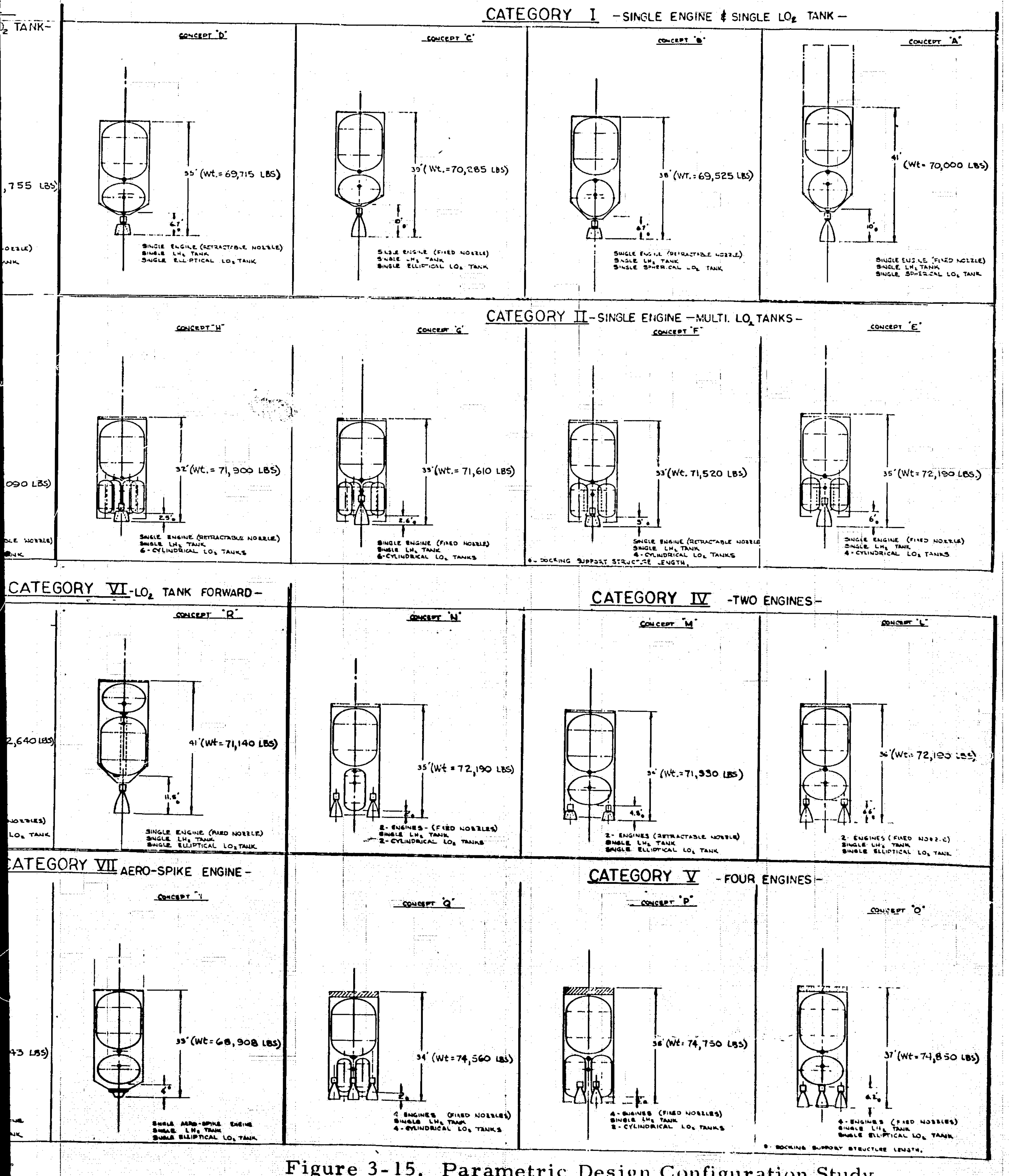


Figure 3-15. Parametric Design Configuration Study

can be seen, in general, that the major variation in vehicle length is primarily influenced by the LOX tank and engine options selected. The volume of LOX for a nominal vehicle is such that it permits consideration of multiple tanks for more efficient packaging. Clustered LOX tanks, at the expense of weight and system complexity, accommodate varying degrees of submersion of the engine support structures and the head end of the engine, as shown in Concepts E through H, N, P, and Q. Depending upon the number of clustered LOX tanks, propellant storage system and overall stage length reductions on the order of 6 percent and 22 percent, respectively, can be realized in comparison to a single LOX tank. The use of a retractable nozzle skirt on a conventional bell engine as shown on Concepts B, D, K and M permits propellant storage system and vehicle length reductions in the range of 15 percent and 29 percent, respectively, for single LOX tank concepts. The multiple LOX tank concepts shown as F and H with the retractable nozzle engine permits reductions in propellant storage system and vehicle length reductions in the range of 5 percent and 22 percent respectively.

A toroidal LOX tank in conjunction with a retractable nozzle engine (Concept K) provides a high degree of volumetric utilization and results in the shortest stage length for single engined concepts, 29 percent less than Concept A. The use of dual engines (Category IV) also provides stage length reductions in comparison to single engined arrangements, but does not appear to offer any advantage over a single engine with fixed nozzle and four LOX tanks (unless centerline docking is required). Dual engines result in the use of dual LOX tanks, from general packaging considerations.

Vehicles featuring four engines (Category V) can be configured with either one, two, or four LOX tanks. The greatest overall vehicle length reduction results from the use of four engines and four LOX tanks. However, it should be noted that this concept is only one foot (0.30 meter) shorter than a stage featuring four LOX tanks and a single fixed nozzle engine.

Concepts T and U illustrate configurations featuring an aerospike engine. The aerospike provides significant stage length reductions of 15 percent compared with the corresponding single engine fixed nozzle concept. In the case of the toroidal LOX tanks, however, the aerospike engine results in a length penalty (1/2 percent) compared to the retractable nozzle single engine concept. The concept does however, require less propellant.

All of the concepts on Figure 3-15 required more propellant than the baseline to complete the same geosynchronous mission with the exception of concepts B, D, and T, which required slightly less. Concept T required the least propellant, 68,908 pounds (31,256 kg) with the use of a single aerospike engine and a single elliptical LOX tank. The shortest overall vehicle

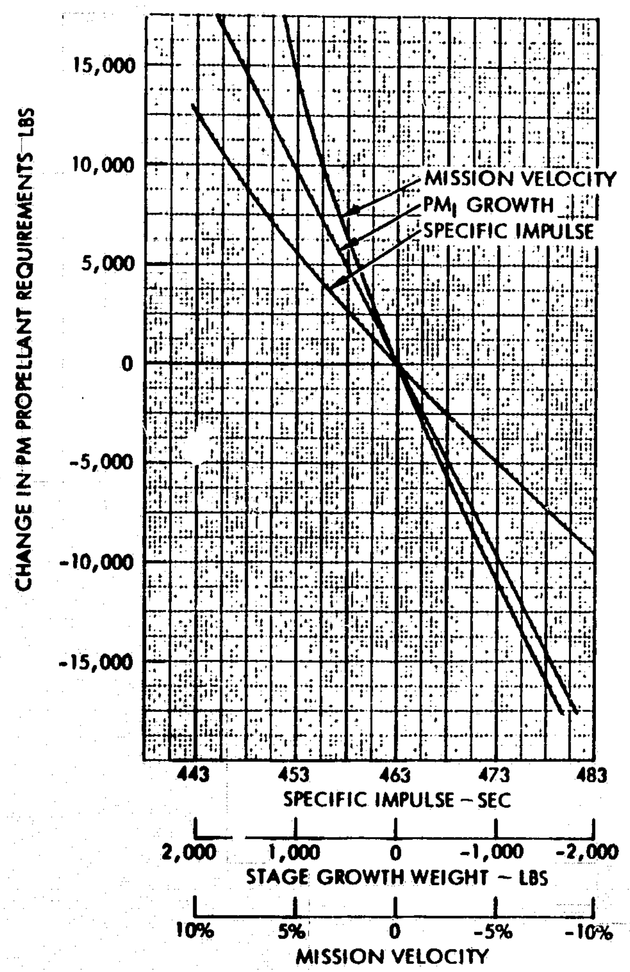
(29 feet - 8.8 meters) is Concept K which incorporated a toroidal LOX tank and a single retractable nozzle engine. The toroidal tank concepts probably would be more complex in terms of propellant delivery and tank manufacture. The cost of such a tank would also be higher than that of a more conventional tank. The inverted position LOX tank concepts, R and S, both exhibited larger propellant quantities and much higher center of gravities than the other concepts, but this is of importance only in the lunar lander application.

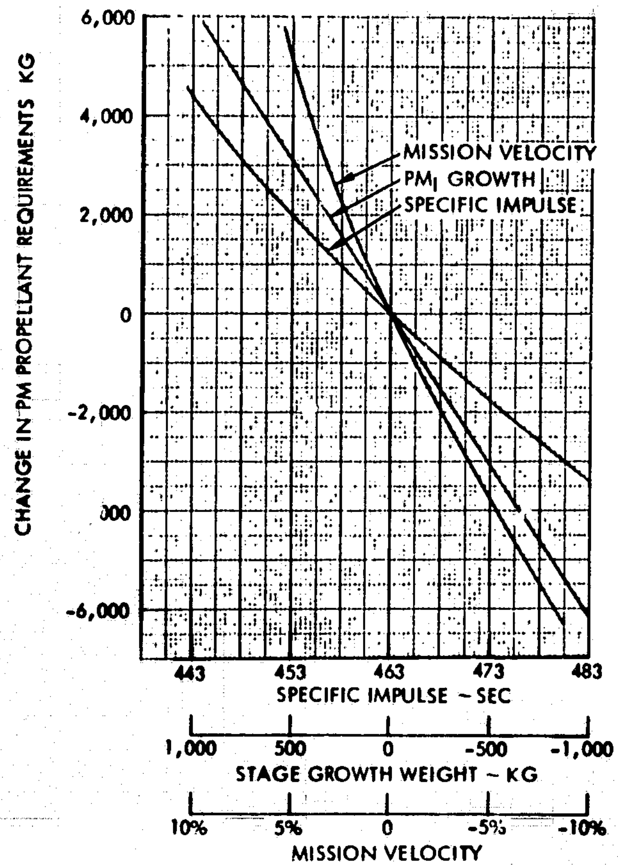
3.5.3 PROPELLANT SENSITIVITIES TO INERT WEIGHTS

In addition to these specific quantitative effects of system variables, the sensitivity of the three baseline tug concepts to such weight excursions from the nominal can be shown in terms of propellant weight required with changes in the primary variables of system inert weight, propulsion specific impulse change, and off-nominal delta-V. The following three plots, Figures 3-16, 3-17, and 3-18, show these sensitivities for the single-stage, tandem dual stage (in two staging modes), and the 1-1/2 stage expendable tank set baseline concepts.

In Figure 3-16, three primary system/operational variables are illustrated in terms of their effect on the design propellant quantity required for the baseline geosynchronous 10K payload placement mission. A nominal specific impulse, I_{SP}, of 463 seconds (see Propulsion Appendix of this report for details and background) has been used. A reduction of I_{SP} to 443 seconds would be appropriate if an earlier type of engine such as the existing RL-10 series were to be utilized. If the engine(s) is large enough (more than approximately 10K (44.5N) thrust) to qualify as a potentially higher chamber pressure design, I_{SP} up to as high as 470 seconds could be foreseen as a possibility. A drastic change in the operational groundrules such as ground basing and reduced autonomy could easily result in a subsystems philosophy which would reduce weight by 1000 pounds (454 kilograms). An increase in mission-peculiar provisions such as special kits or retrieval devices could easily add 1000 pounds to the basic weight. Likewise, changes in baseline delta-V could result from different allowances for reserves, or from a different flight profile.

In Figure 3-17, propellant sensitivities for the two-stage tandem Concept 5 are depicted. In this case, the change in specific impulse applies to both stages and the propellant change is for each of the two equal size stages. Likewise, the inert weight change or "stage growth weight" applies in equal amounts to each stage; in other words, the weight increment on the scale indicated is for each stage. The propellant weight sensitivity is seen to be less sensitive to changes in the trapeze mode.





100/100 N MI (185/185 KM) GEOSYNCHRONOUS MISSION
SINGLE-STAGE RECOVERABLE
10,000 LB (4536 KG) PAYLOAD
IM + PM + PL

Figure 3-16. Design Sensitivities for Concept 1

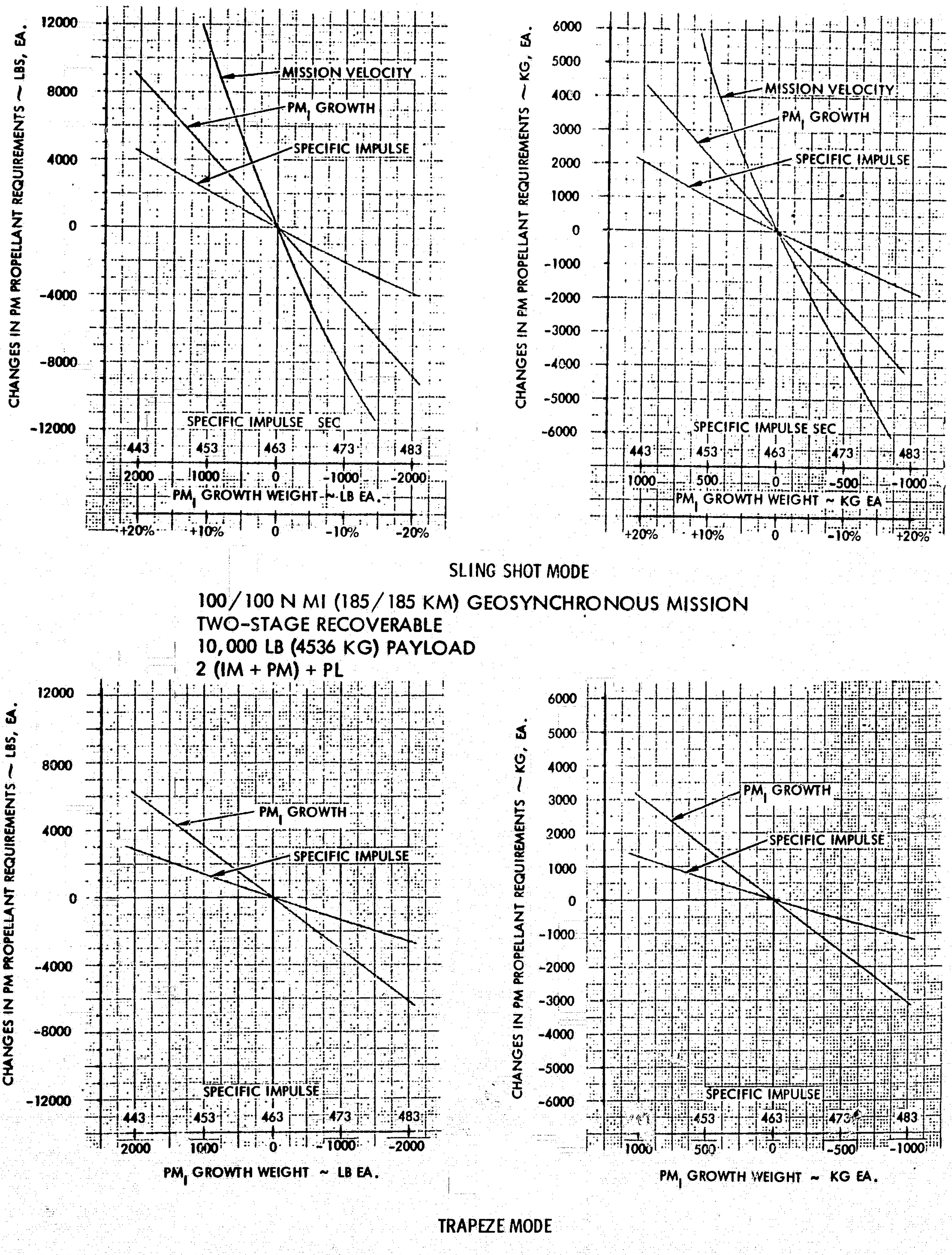
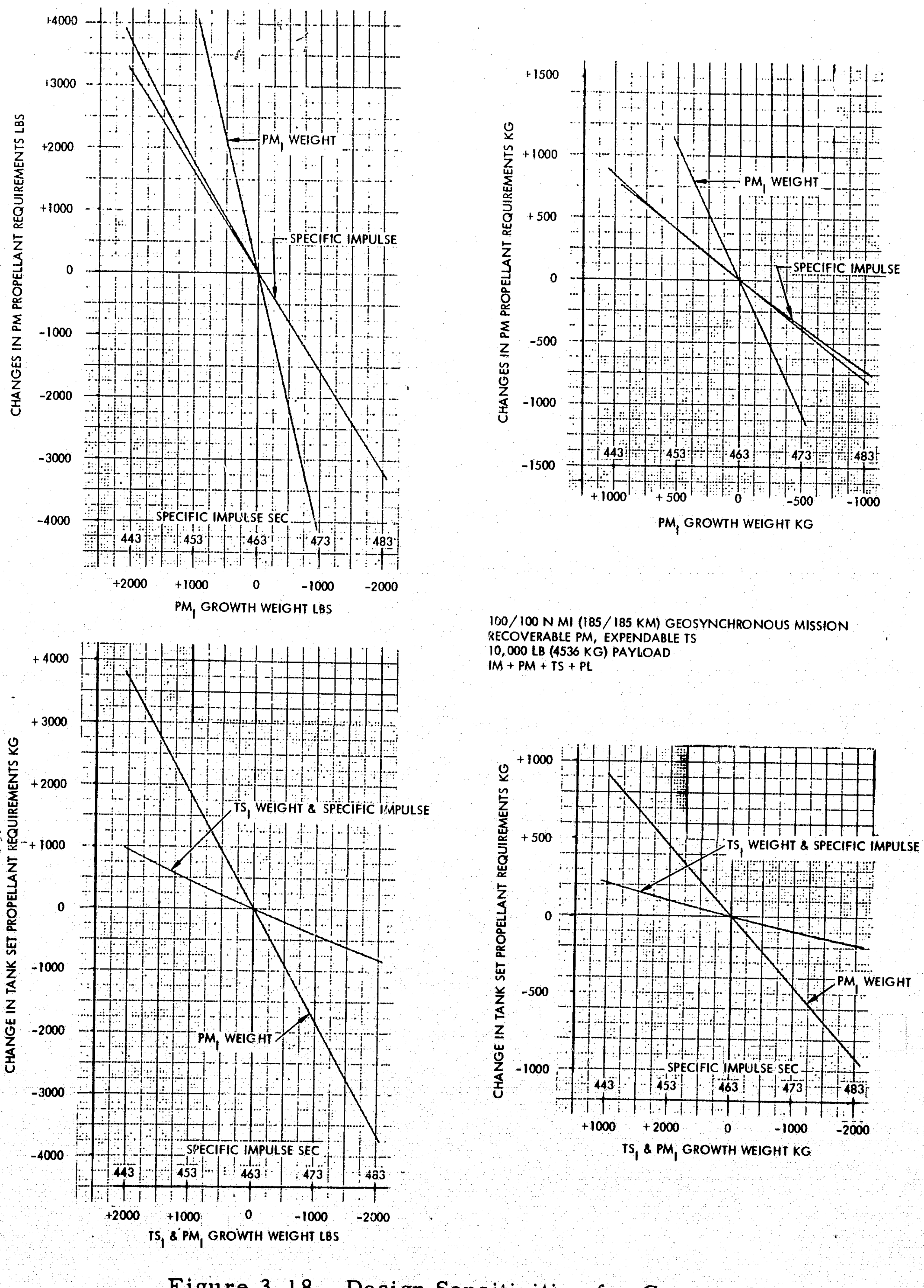


Figure 3-17. Design Sensitivities for Concept 5



100/100 N MI (185/185 KM) GEOSYNCHRONOUS MISSION
RECOVERABLE PM, EXPENDABLE TS
10,000 LB (4536 KG) PAYLOAD
IM + PM + TS + PL

Figure 3-18. Design Sensitivities for Concept 11

In Figure 3-18, the equivalent sensitivities for the 1-1/2 stage expendable tank set Concept 11 are shown. On the left, the changes are seen for the design propellant quantity in the small PM. On the right half of this figure are shown the effect on tank set required size with changes in PM specific impulse and with total stage (PM plus tank set) inert weight change.

The effect of propellant weight on system gross weight can be seen in Figure 3-19.

Propellant weight, in turn, can be related to vehicle length, an important parameter in respect to the EOS cargo bay length. The length of the combined IM/PM for three diameters, 12, 15, and 22 feet (3.7, 4.6 and 6.7 meters) is illustrated in Figure 3-20. Lengths for two alternate 15.0 feet (4.6 meters) vehicle arrangements are also shown. The allowable length of the spacecraft, without a crew module, and with a 1-floor or 2-floor crew module that will fit into an EOS cargo bay are illustrated by horizontal bands. The intersection of a horizontal band with a configuration length line yields the maximum propellant load the tug can be designed to carry and still fit into the EOS cargo bay in one piece.

Other factors indicating sensitivity of many significant items such as payload performance, systems performance, subsystem characteristics, and cost factors are described and illustrated throughout this report as design, operations, missions, modes, systems, and program variables are being discussed.

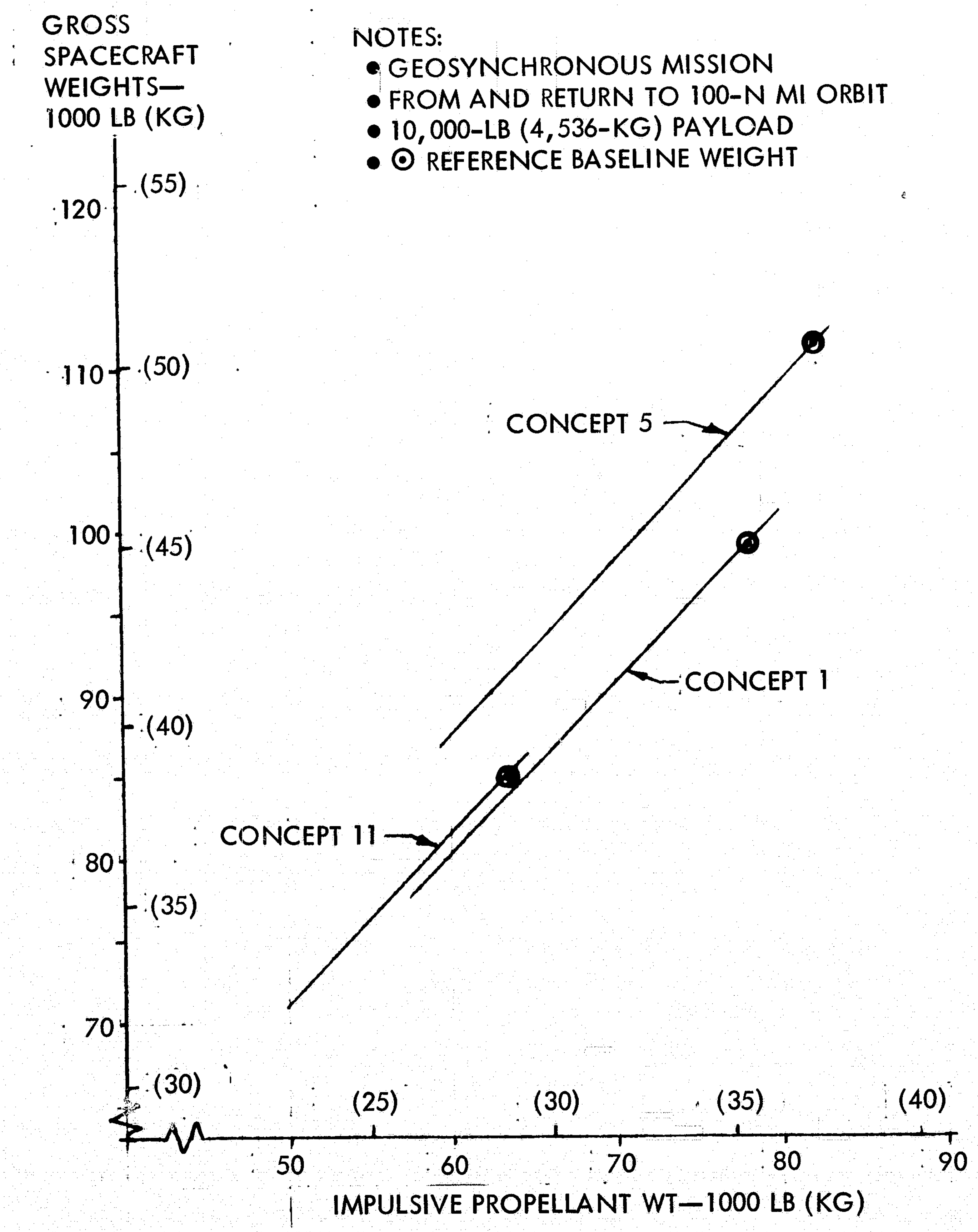


Figure 3-19. Gross Weight Versus Propellant Weight

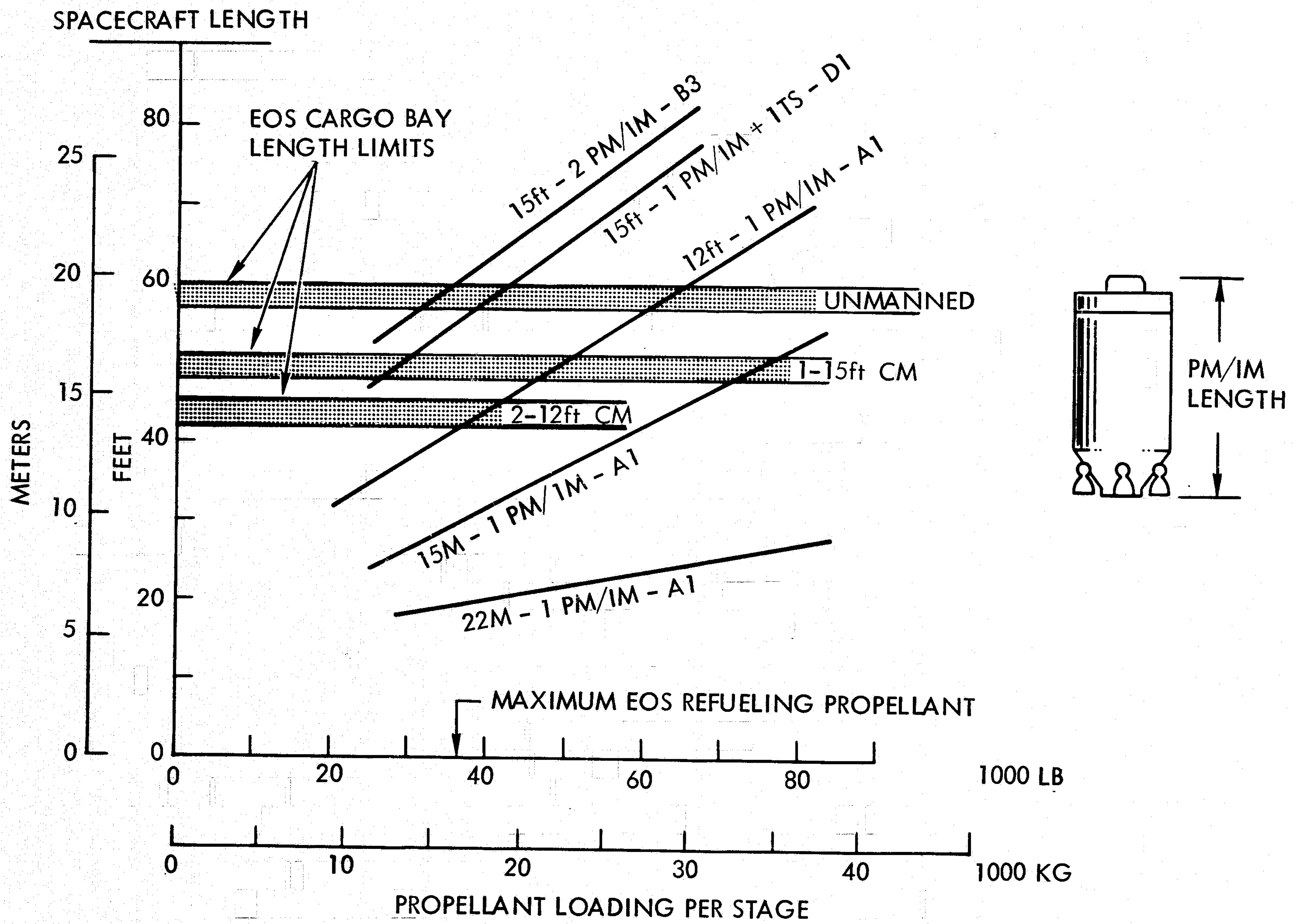


Figure 3-20. Spacecraft Length

3.6 ALTERNATE USAGE OF TUG MODULES

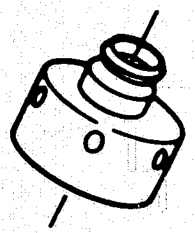
The PM designed for maximum performance in the geosynchronous 10,000-pound payload application and for flexible multi-purpose use in low earth orbit can be adapted later in a block change for use in lunar landing missions. The nature of the requirements for lunar landers dictate a block change from the earth orbital design configuration to accommodate landing gear, low c. g., engine throttling, engine deployment, CM modifications, cargo pods, etc. As an unmanned logistic cargo lander, this adapted PM can land large payloads in modules or pods symmetrically loaded onto the PM, or multiple PM, or multiple PM's can be attached to the cargo core module.

In addition to the several alternative mission applications for versions of tug which have been identified by NASA for the contract study, there are some interesting possibilities for use of these individual modules, Figure 3-21. The CM in the lunar configuration would be applicable as an interim mini-space station module launched to any reasonable orbit by the EOS. The CM could be used singly or multiples for more working and living space. The module also could be utilized as the passenger transport module within the EOS bay rather than a special one as currently studied in the EOS program. Likewise, the CM could be utilized on either the nuclear or the chemical cislunar shuttle, and could be the control station for an orbiting propellant facility.

The IM could be utilized to support EOSS experiment modules rather than having their own similar functions duplicated in each experiment module as at present. The IM could serve as the "brains" and stability center for an orbiting propellant facility, and portions or subsections of equipment could be used for similar functions on the cislunar shuttles or the orbital injection stage (replacement for EOS orbiter).

The CM hemipods could be bolted together and used as a space station supply module or "pantry". As hemicylinders, a pressurized version of the pods might be lowered to the surface and become lunar surface shelter modules.

The application of the tug as an interim space station, or modular element thereof, is considered so attractive potentially that this possibility is further explored, Figure 3-22. Regardless of whether a large space



CM

- 6-12 MAN MODULE IN EOS
- CM FOR CIS
- CM FOR ORBITING PROPELLANT FACILITY
- 15-FT DIAMETER MINI-SS MODULE(S)

IM

- IM FUNCTIONS FOR
  - EOSS EXPERIMENT MODULES
  - ORBITING PROPELLANT FACILITY
  - CIS OR OIS

CAM

- EOSS SUPPLY MODULE
- LUNAR SURFACE SHELTER

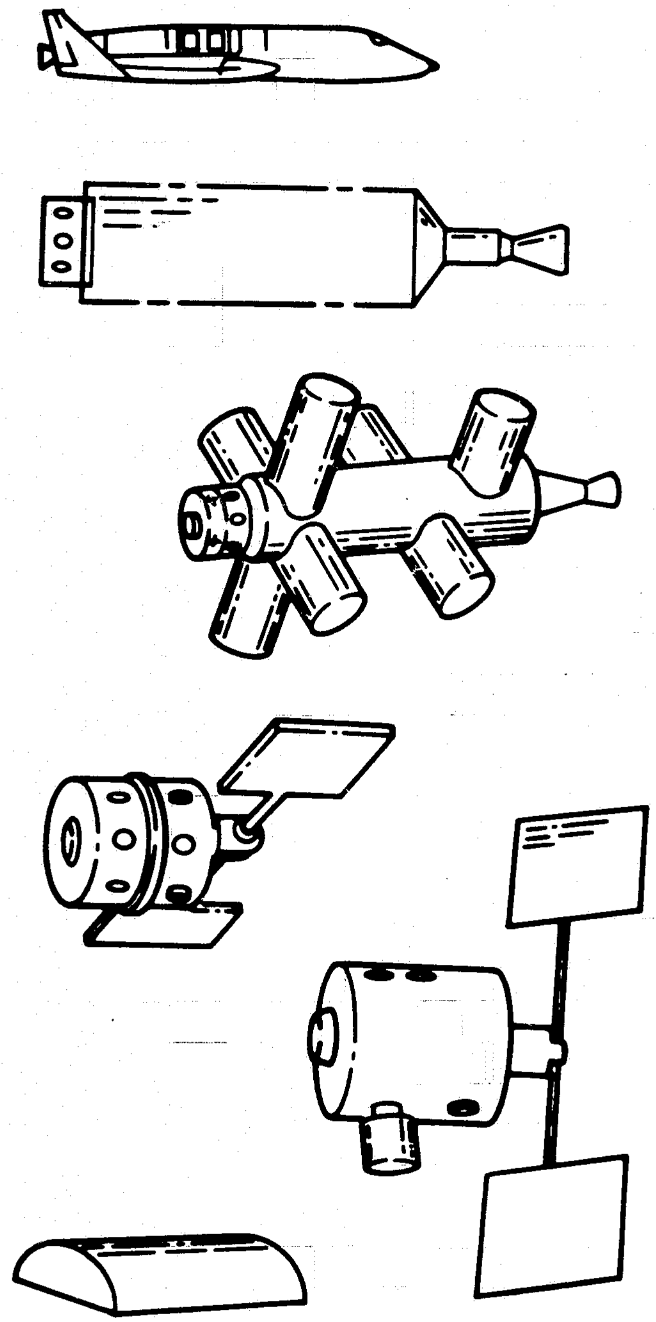


Figure 3-21. Potential Applications of Tug Modules

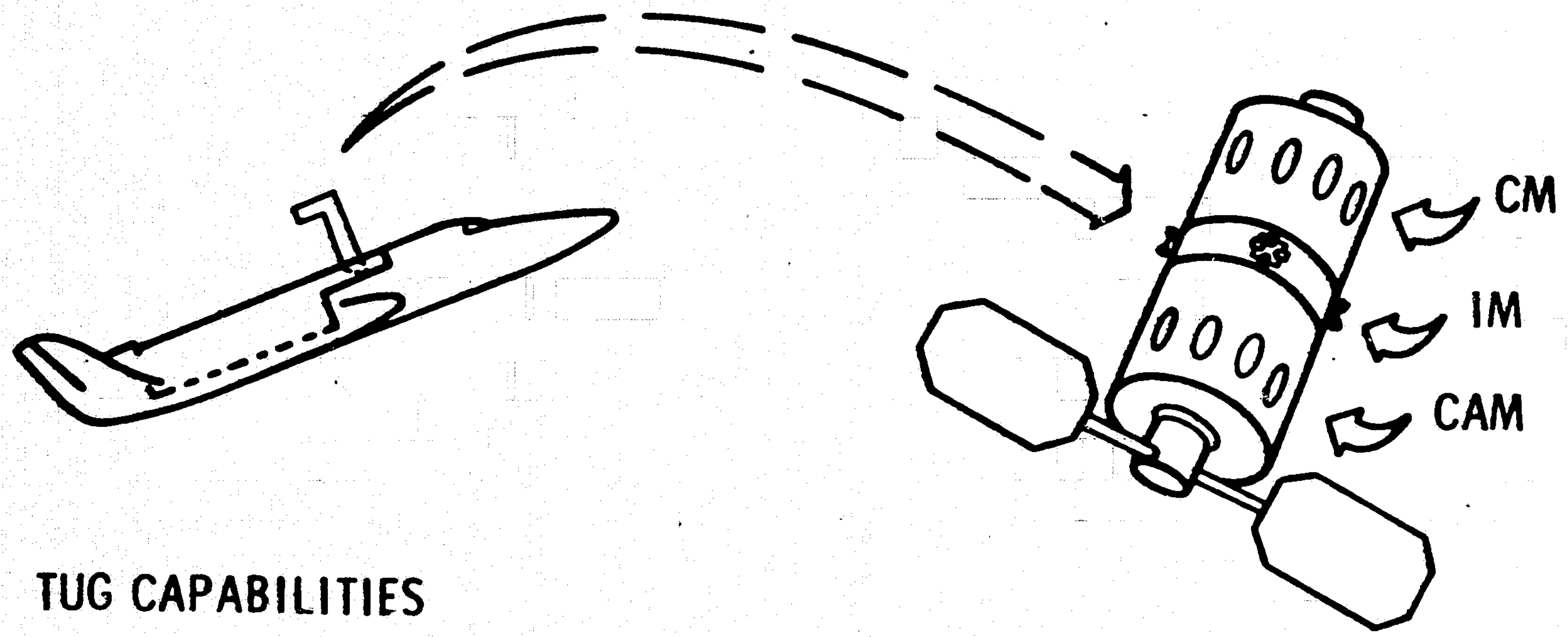
The application of the tug as an interim space station, or modular element thereof, is considered so attractive potentially that this possibility is further explored, Figure 3-22. Regardless of whether a large space station exists or not, it may be desirable to have the capability of conducting manned missions in any orbit, utilizing the EOS as the container of the small station or for delivery of the station to orbit. The crew module of the space tug will be designed for 4 men for 28 days on the lunar surface, and, therefore, can have at least this capability in earth orbit. The same shell also may be used as a module for housing experiments. The crew module when combined with the intelligence module and the necessary expendables can provide the capability for such a small station. Addition of solar cells might also be considered to reduce the amount of expendables.

3.6.1 POSSIBLE MINITUG VARIANT OF CONCEPT 1

One of the most attractive features of Concept 11, the 1-1/2-stage concept, is the availability of a small PM for many Earth orbital missions which do not involve large delta-V maneuvers. The small PM with 11K (500 Kg) propellants (see Figure 3-1) is adequate for a large number of these missions and its short length promotes maneuverability as well as minimizing length within the EOS cargo bay. This could be a distinct advantage when considering very bulky payloads, multiple payloads, or combined cargoes within the bay.

One way of achieving similar advantages of small size for many Earth orbital missions, yet retaining the large payload capability and vehicle simplicity of the single-stage Concept 1, is seen in Figure 3-23. An integrated IM (non-modular) concept could be utilized with all IM equipment in the lower or LOX tank portion of the PM, separable as a submodule from the LH_2 portion of the PM. Then, the LOX submodule could be separated and part of the LOX tankage converted to LH_2 tankage thus constituting a small PM. (The need for a separable IM appears to be eliminated when a small PM is available.) By adding a docking adapter, a small Minitug concept is achieved without severe compromise of the basic Concept 1 configuration.

Three LOX tanks would be converted to LH_2 as shown to provide nearly 20,000 pounds (9000 Kg) of propellants. However, this would result in an assymmetrical cg relative to the thrust vector. A better compromise would be to utilize only two LOX tanks converted to LH_2 with the other two LOX tanks offloaded for correct mixture ratio (6/1) to yield lateral cg symmetry. The resulting propellant capacity would be quite ample, about 13000 pounds (5900 Kg).



TUG CAPABILITIES

- 6 MEN/CREW MODULE
- COMPLETE ASTRIONICS
- ~ 45-DAY RESUPPLY PERIODS
- SOLAR CELL OR SOLAR CELL/CMG KIT
- ANY ORBIT EOS LAUNCH

Figure 3-22. Space Tug as Interim Space Station

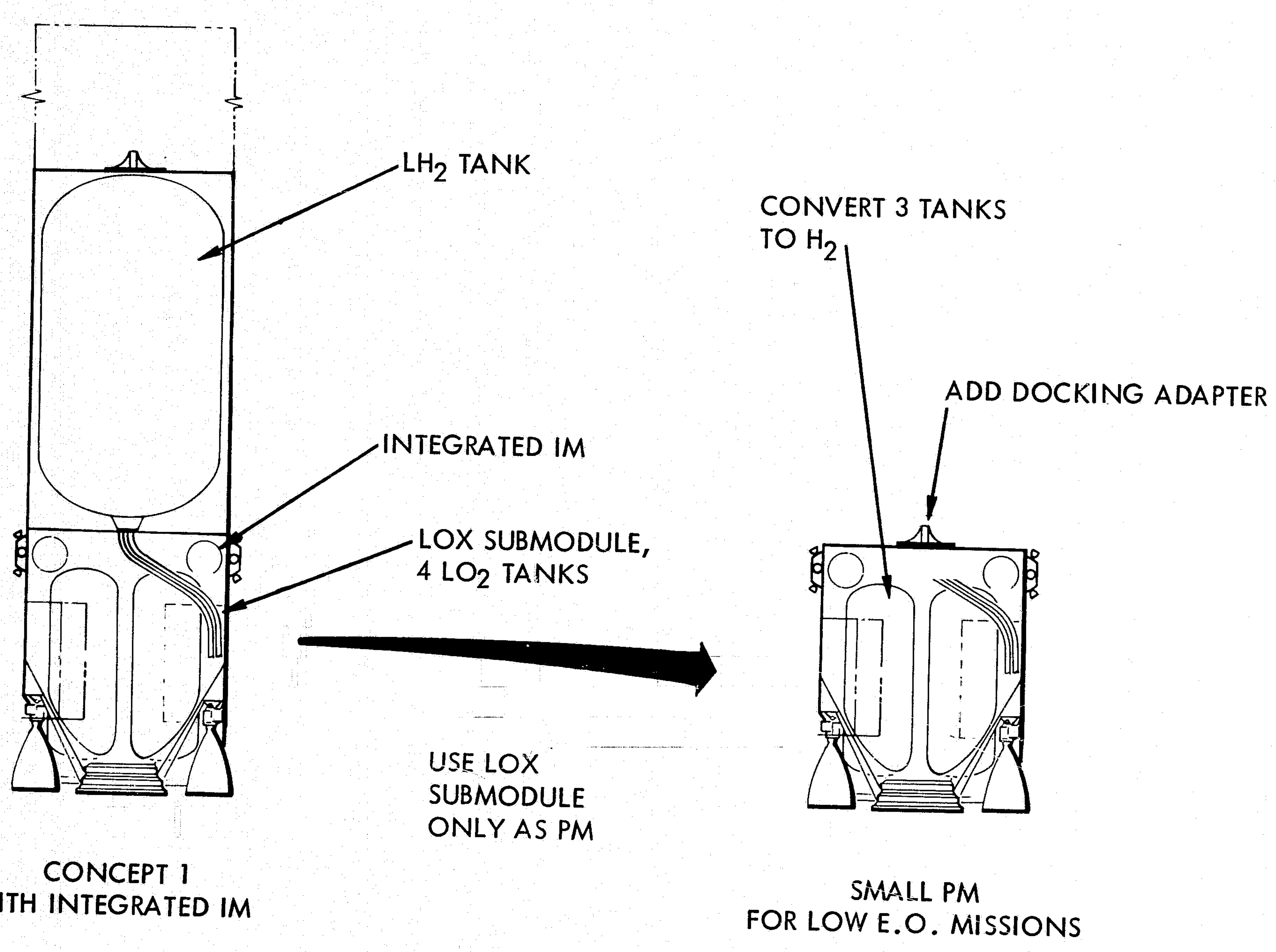


Figure 3-23. Possible Variant With Concept 1

4.0 RECOMMENDED FUTURE EFFORT

4.1 CONFIGURATIONS

As discussed previously concept configurations are extremely dependent upon assumptions and alternatives chosen during current and future studies. It becomes, therefore, extremely important to maintain a close vigilance relative to interfaces and changing requirements and to adjust the design concept approach accordingly. The concept configurations are very important in terms of operations, sizes, and weights, which in turn directly affect costs.

An example of the importance of configurations studies along with operations studies is in the effect of tug concept size. A single-stage recoverable tug might not be able to carry a certain long payload and be fitted into the EOS cargo bay. Thus, a second EOS launch of the payload with subsequent orbital assembly operations to mate payload and tug would be required. This would impact the operations costs substantially. An alternative set of assumptions (groundrules) might permit a reduction in tug size such that a single EOS launch is required when the payload is mated to tug. Operations and costs would thus be substantially lower for that mission.

In future tug study phases it is recommended that a concept design effort be given appropriate priority to insure that the groundrules and assumptions are properly reflected in tug sizing. It is also expected that still more promising avenues of design investigation will continue to evolve in future studies as a result of concurrent effort on other major interfacing systems, particularly the EOS.

4.2 SYSTEMS

The foregoing discussions have indicated the sensitivity of tug concepts to propulsion parameters and the need for a high degree of advanced propulsion technology. Current plans indicate that a major portion of this required advancement will be achieved in the development of the EOS program. However, two avenues of approach may need consideration in regard to tug. First, it is difficult at this time to predict with certainty the exact directions and dimensions of the propulsion technology which will be utilized by EOS. While it does appear that the necessary primary and secondary propulsion elements are already receiving some development attention, the ultimate directions which these elements will take for EOS are not necessarily certain at this time; therefore, their direct applicability to tug is not certain. Secondly, the

particular design requirements for EOS, while generally compatible with tug in respect to required state of the art (1972-74), may well force compromises in propulsion concepts which are not necessarily best or adequate for tug. Therefore, it will be necessary in future tug studies to maintain cognizance of EOS and tug similarities and differences so that a common program of propulsion development can evolve for the economic benefit of both programs.

A specific example of such a potentially common propulsion element, if properly coordinated in development, is the EOS Orbital Maneuvering System (OMS) engine. The basic technology and general size appear compatible with tug's main propulsion system but the OMS engine development for EOS would probably not include the turbomachinery cycle and expansion area ratio needed by tug. Therefore, a coordinated development may be in order so that the basic engine development can be utilized in both programs with a minimum of redesign.

In the event that the OMS engine technology does not develop in a direction favorable for tug application, a separate engine development effort directed toward tug may be required to ensure availability of engines with the high performance needed. Lacking this new development, it would be necessary to accept a compromise position by utilizing a modified version of the existing RL-10 engine. However, use of the RL-10, even with improvements to increase chamber pressure and area ratio to improve specific impulse and with turbo-machinery improvement to reduce NPSH required, probably would not result in performance equal to that expected with a new advanced engine development with inherent growth potential. Furthermore, it is believed that the costs involved with an RL-10 improvement program might well be a substantial fraction of those to be expected for an advanced engine development.

The other propulsion areas of study recommended deal with the long-term space use of cryogenic propellants with regard to zero-g slosh and feedout, thermodynamic venting and control, and propellant conditioning. Design techniques for practical long-life reusable high performance tank insulation systems, especially for ground-based operations, also require analysis and scale development testing.

Future studies in the area of optimum structural design will be required to develop the best philosophy for tug. It will be necessary first to establish more definitive design groundrules such as number of engines, type and number of tanks, and basing and docking philosophy, all of which affect the base structure design approach. The design effort will then have to be coordinated with the thermal design of the cryogenic propellant insulation system and will involve the conductive heat paths through structure, instrumentation, and fluid lines as well as the radiative high performance insulation and venting provisions.

Meteoroid protection is still another consideration. The potential utilization of boron-epoxy technology for the tug structure such as cryogenic tank supports and other inter-tank structure could offer substantial and important weight savings and requires special study effort to determine feasibility and availability for tug.

4.3 SUBSYSTEMS AND TECHNOLOGY

Subsystems study effort required is discussed under "Subsystems For Tug" in Volume 5 of this report. These and the preceding items discussed are also included under the Supporting Research and Technology Plan in Volume 6 of this report, in the "Development Plans for Tug" section.

Items in the subsystems areas which appear most in need of future study effort are summarized in the rollowing paragraphs.

Advanced Computers and Navigation Equipment

Promising laboratory developments in this field appear "just over the horizon" for tug applicability and could well be accelerated to insure availability for tug at substantial weight and power savings.

Optimum Communication Frequencies

A study of future integrated space operations could reveal that higher (than S-band) frequencies would be feasible and would result in appreciable telecommunications antennae and other equipment weight savings.

Rendezvous and Docking Aids

Problems of visibility in adverse space lighting conditions could be alleviated by development of improved visual aids.

Digital Data Buses

Presently known digital techniques have reduced the weight and volume required for wiring, but wire weight is still a significant item in the weight of a spacecraft such as tug, on the order of hundreds of pounds. Studies directed toward use of inconventional techniques such as, for example, flat or tubular conductors, might provide a needed weight saving.

Docking Structure

The docking structure as employed in the EOSS study imposes excessive weight penalty on tug which must undergo very large delta-V maneuvers. Therefore, another "standard" for mechanical docking interfaces which could be similar in size and weight to the Apollo probe and drogue system should be evaluated for possible universal use.

Landing Sensors

Current techniques for visualizing and sensing lunar surface features, roughness, obstacles, etc. are inadequate for several reasons. One is the obscuration resulting from the churning of surface dust by lander exhaust impingement. Visibility limitation in terms of glare and contrast is currently inadequate to permit landing in other than a narrow range of sun angles, and night landings are not permissible. Therefore, a study is needed to conceive and develop new techniques of sensors, graphic displays, enhanced TV, etc., and to evaluate means of reducing visual degradation caused by dust.

APPENDIX A

INTEGRATED STRUCTURES

by A. J. Richardson

APPENDIX A. INTEGRATED STRUCTURES

APPROACH

The structure of the vehicle consists essentially of the vehicle body, main propulsion thrust structure, the cryogenic propellant tankages, EOS attachment provisions, and the payload attachment provisions. Research at NR/SD on advanced propulsion module concepts requiring storage periods for cryogenic propellants has identified a strong interdependence between structure, thermal protection, and meteoroid protection to the point where it is essential to consider all three simultaneously to obtain accurate definitions of the most desirable solutions. The resulting assembly of these three types of components is identified as an integrated structure, and an integrated structures design (IS) method has been developed for study purposes. Occurrence of propellant boiloff during extended missions means that the mass fraction is not an accurate measure of propulsion module efficiency; therefore, in the IS approach the integrated structure concept evaluation is based on payload fraction (the ratio of payload capability to initial gross weight of the module). For example, an integrated structure that weighs 1000 pounds more than another but permits 2000 pounds less propellant boiloff during the mission is more efficient and desirable. This efficiency is measurable by denoting its payload capability which must consider boiloff of propellant.

A typical integrated structure concept is shown in Figure A-1. This concept is a direct evolution from current booster design, with modifications in the form of low thermal conductivity material for unpressurized structure, the use of multilayer insulation, and incorporation of an external atmosphere control barrier for the insulation, which also serves as a meteoroid bumper.

Application of the IS method includes:

1. Computation of critical loads in each major structural member followed by sizing of the member
2. Computation of optimum boiloff and the required insulation for the tanks and unpressurized structure
3. Computation of meteoroid protection (if required)
4. Computation of payload fraction capability.

Computer subroutines are available for steps 1, 2, and 3.

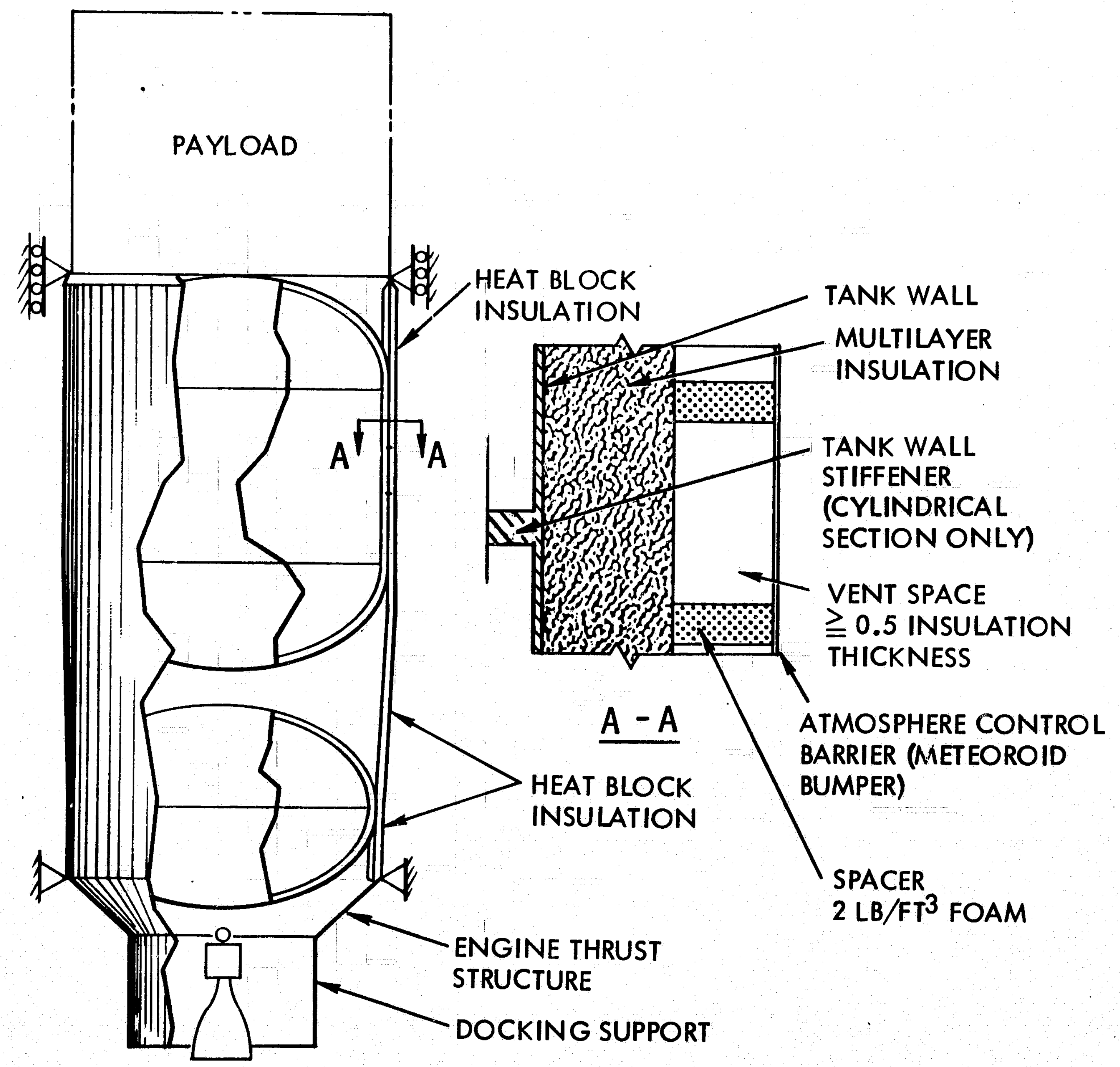


Figure A-1. Typical Integrated Structure for Cryogenic Propulsion Module

Structural loads are computed by conventional techniques and can be obtained for a variety of structural attachment combinations and for a variety of flight phases. Structural sizing and weights are obtained from existing theoretical data as a function of material and loading. Insulation sizing and boiloff are determined in the following steps:

1. Insulation length on LH_2 supports is computed which yields minimum insulation weight for a given heat leak

2. Insulation thickness for the LH_2 tank and supports is computed which maximizes the payload fraction

$$W_p/W_{ig} = \left[(W_{fb}/e^x - 1) - W_b\right]/W_{ig} \tag{1}$$

where

$x = \epsilon \Delta V/Tg$

W_p = propulsion module payload capability at ΔV

W_{ig} = initial gross weight of propulsion module

W_{fb} = weight of fuel burned

ϵ = propellant burn rate

ΔV = required velocity change

T = engine thrust

W_b = inert weight of propulsion module

3. Steps 1 and 2 are repeated for the LO_2 tank.

A meteoroid protection concept is adapted to the insulation-structure assembly and the delta weight computed. Both single and dual bumper concepts can be treated. The shielding required is related to the required probability of no failure by the Poisson relationship

$$P_{nf} = e^{-NAT}$$

where

P_{nf} = the required probability of no tank failure

N = the meteoroid flux of particles large enough to cause failure

A = ST surface area

T = ST exposure time

The resulting integrated structure is evaluated by the use of Equation 1.

STRUCTURE CONFIGURATION AND MATERIALS

The effects of RST structure configuration and choice of material for the unpressurized structure on ST structure weight were evaluated first, to identify the most promising configurations. A 30K propellant capacity PM was evaluated using the criteria of Table A-1. The mission phases and load factors considered are shown in Figure A-2.

The structure configurations considered were different combinations of integral and nonintegral LH_2 tank; ellipsoidal, toroidal, and three-tank LO_2 tank arrangements; truss and skin-stringer shell for the unpressurized structure. Material for the tanks was 2014-T6 aluminum, and for the unpressurized structure — boron-epoxy, glass/epoxy, or titanium alloy. Analysis results obtained are presented in Figures A-3 and A-4 and findings were as follows:

1. The computed integral tank structure weights were nominally 32 percent less than the nonintegral tank structure weights.

2. Total structure weights based on truss construction and on skin-stringer shell constructions for the unpressurized structure were essentially the same.

3. The structure weight is nominally 15 percent lower if a single LO_2 tank (ellipsoid or toroid) is used as opposed to multiple LO_2 tanks. This is due primarily to the extra tank support structure required for the multiple tank concept.

4. The choice of material for the unpressurized structure can alter structure weight by 25 to 40 percent with materials rated — boron-epoxy composite, glass-epoxy composite, and titanium alloy.

Table A-1. Input Data Summary (International Units)

	Configuration Study	Capability Study	Integ. vs Noninteg. Study
Propellants			
• Propellants	LH_2/LO_2	LH_2/LO_2	LH_2/LO_2
• Total weight, kg	13,600	13,600	27,200
• Ullage, percent	10/6	0	0
• Temperature, C	-251/-187	-251/-187	-251/-187
• Density, kg/m^3	69.6/1145	69.6/1145	69.6/1145
• Pressure, N/m^2	255,000/244,000	96,500/131,000	138,000/138,000
• Heat of vaporization, J/g	446/207	446/207	446/207
Mission duration, days	45	--	7, 25, 45
Factors of safety			
• Unpressurized structure	1.4	1.4	1.4
• Tanks	1.4	1.4	2.0
Meteoroid Protection			
• Probability of no failure	0.995	--	--
• Design period, days	450	--	--
Engine			
• Thrust, N	122,000	61,600	123,500
• Burn Rate, kg/sec	28.1	13.6	27.2
• Weight, kg	195	110	400
• Fuel Ratio	6:1	6:1	6:1
• Distance from mounting to c.g., cm	21.6	21.6	21.2
ΔV Requirement, mps	4,350	--	4,350
Payload			
• Weight, kg	11,800	4.5K(1)/11.8K(2)	4,540
• Length, cm	30.5	28.3(1)/30.5(2)	30.5
Instrument Module			
• Weight, kg	1,180	1,590(1)/1,225(2)	1,225
• Length, cm	23.6	14.2	14.2

(1) Lunar landing
(2) Launch and space firing

Table A-1. Input Data Summary (English Units) (Cont)

	Configuration Study	Capability Study	Integ vs Noninteg Study
Propellants			
• Propellants	LH_2/LO_2	LH_2/LO_2	LH_2/LO_2
• Total weight, lb	30,000	30,000	60,000
• Ullage, percent	10/6	0	0
• Temperature, F	-420/-284	-420/-284	-420/-284
• Density, pcf	4.35/71.5	4.35/71.5	4.35/71.5
• Pressure, psi	37/34	14/19	20/20
• Heat of vaporization, B/lb	192/92	192/92	192/92
Mission duration, days	45	--	7,25,45
Factors of safety			
• Unpressurized structure	1.4	1.4	1.4
• Tanks	1.4	1.4	2.0
Meteoroid protection			
• Probability of no failure	0.995	--	--
• Design Period, days	450	--	--
Engine			
• Thrust, lb	27,500	13,900	27,800
• Burn rate, lb/sec	62	30	60
• Weight, lb	430	242	880
• Fuel ratio	6:1	6:1	6:1
• Distance from mounting to c.g., inches	55	55	54
ΔV requirement, fps	14,300	--	14,300
Payload (for sizing)			
• Weight, lb	26,000	$10K^{(1)}/26K^{(2)}$	10K
• Length, inches	100	$93^{(1)}/160^{(2)}$	160
Instrument module			
• Weight, lb	2,600	$3.5K^{(1)}/2.7K^{(2)}$	2.7K
• Length, inches	60	36	36

(1) Lunar landing
(2) Launch and space firing

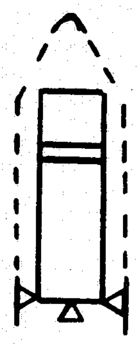
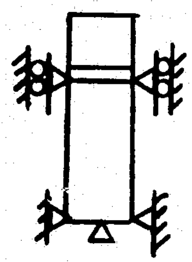
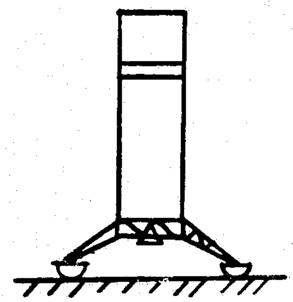
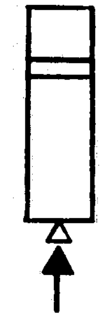
MISSION PHASE		SATURN LAUNCH	EOS LAUNCH	LUNAR LANDING	SPACE FIRING
SPACE TUG FLIGHT PHASE					
CONTROL		UNMANNED	UNMANNED	UNMANNED	UNMANNED
g LOAD	LONG.	6.0	4.0	8.0	0.72
	LAT.	2.5	2.5	2.5	0

Figure A-2. Space Tug Loading for Tug Structure Configuration Study

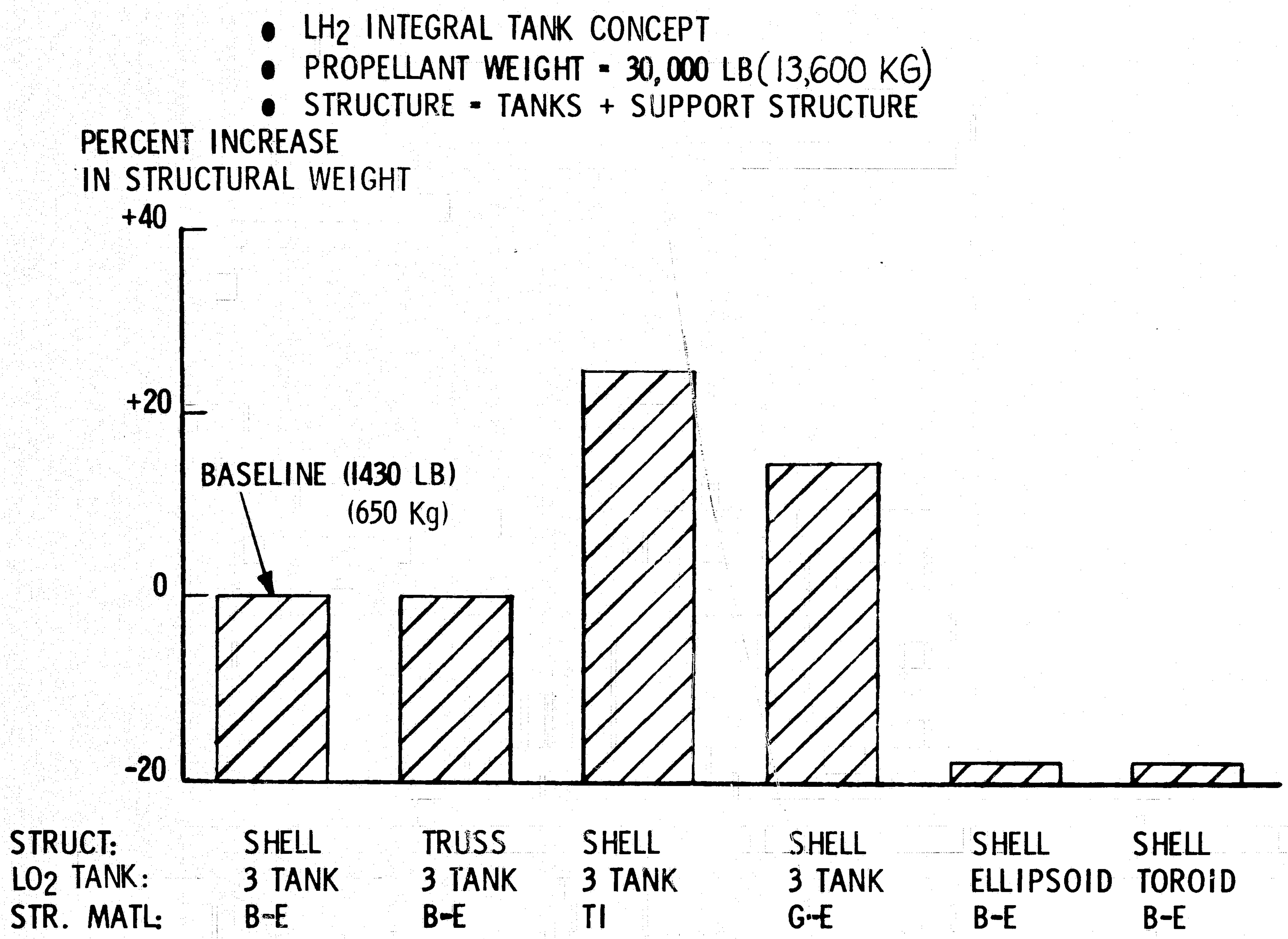


Figure A-3. Propulsion Module Structural Configuration Study Results

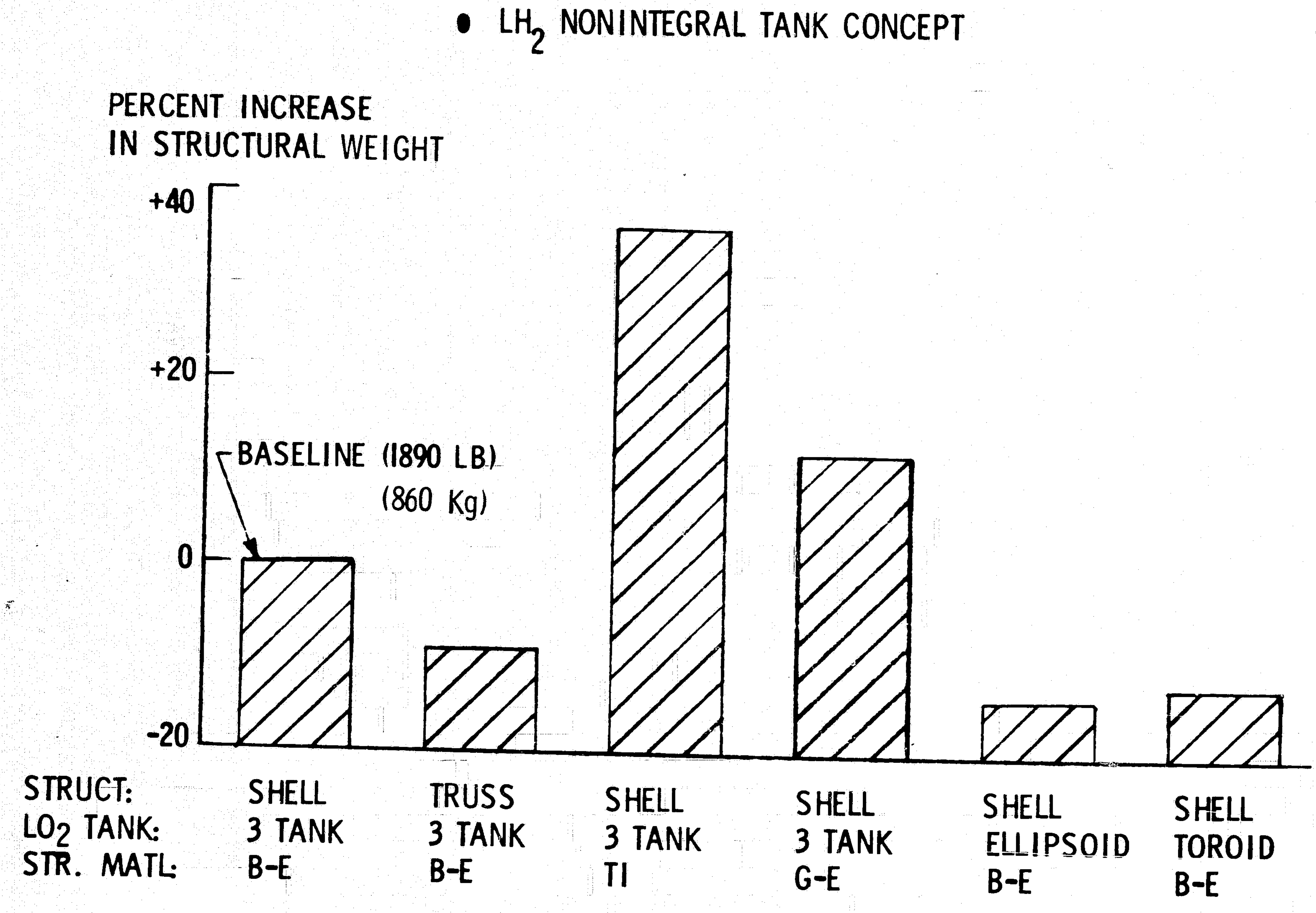


Figure A-4. Propulsion Module Structural Configuration Study Results

MISSION CAPABILITY EFFECTS ON STRUCTURE

The effects of mission capability on structure weight were investigated next, to establish the weight penalty associated with a versatile space tug as opposed to a single purpose space tug. The primary space tug mission was an EOS launch, with all propellants aboard, followed by insertion of payloads into geosynchronous orbit and/or insertion of payloads into space flight. It was considered desirable to extend mission capability to include launch by a Saturn-type booster and to utilize the space tug for lunar landings. Dry launch was considered to reduce total boost weight and to reduce critical structural loadings.

Four different mission capabilities were considered as described in Table A-2. Case I is baseline for it represents the minimum capability for the space tug. The PM was launched dry thereby reducing the loading on some structural members. Geosynchronous orbit insertion and space travel phases are within capability. Case II is the same as Case I except the PM is launched wet. Case III is the same as Case II except launch is via the INT-20. Case IV is the same as Case I except the tandem lunar landing phase is added. Phase load factors are in Figure A-5.

The PM structural concept evaluated (Figure A-6) was selected on the basis of the preceding analysis which had shown it to be efficient and inherently resistant to meteoroid damage. Details on the PM concept and the other components making up the space tug are provided in Table A-1.

For each mission capability case, the load intensity in the primary PM structure components was computed and the critical load established. Structure components were sized and weight computed. Results obtained are presented in Table A-2 and Figure A-7. The major findings were as follows:

1. Tank thickness required is minimum gage, and the shell structure components require near minimum gage for all mission phase loadings

2. Due to the above, there is only a small PM structure weight penalty associated with having full mission phase capability as opposed to the minimum capability

3. Changing from dry to wet launch increased the critical loading only in the LO_2 tank support and the inter-tank structure, with the result that the weight of the structure increased by only 8 percent.

Table A-2. Effect of Mission Capability on Propulsion Module Structural Loads

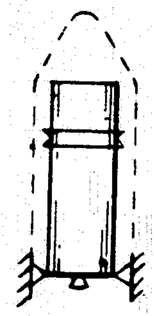
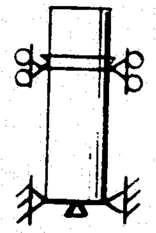
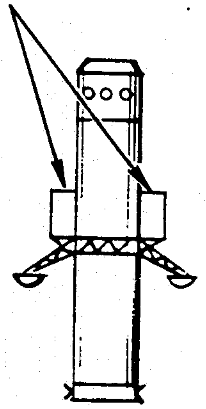
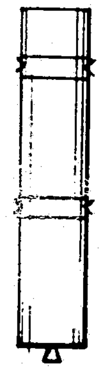
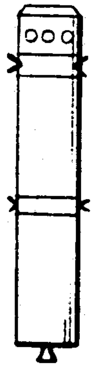
Mission Capability Case	Mission Capability: Mission Phases the Tug Subjected to (1)	Mission Capability: Launch	Booster	Structure Component: LH_2 Tank Shell: Critical Phase	LH_2 Tank Shell: Load(3) (lb/in) (N/cm)	LO_2 Tank Shell: Critical Phase	LO_2 Tank Shell: Load(3) (lb/in) (N/cm)	Forward Adapter: Critical Phase	Forward Adapter: Load(3) (lb/in) (N/cm)	Intertank Structure: Critical Phase	Intertank Structure: Load(3) (lb/in) (N/cm)	Thrust Structure: Critical Phase	Thrust Structure: Load(3) (lb/in) (N/cm)
I	2, 4G, 4S	Dry	EOS	(4)	+1190 (+2080)	(4)	+356 (+622)	2	-228.6 (-400)	2	-176.9 (-310)	4S	-98.8(2) (-173)
II	2, 4G, 4S	Wet	EOS		+1190 (+2080)		+356 (+622)	2	-228.6 (-400)	2	-190.8 (-334)	4S	-98.8(2) (-173)
III	1, 4G, 4S	Wet	Sat		+1190 (+2080)		+356 (+622)	1	-129.8 (-227)	1	-188.4 (-330)	4S	-98.8(2) (-173)
IV	2, 4G, 4S, 3T	Dry	EOS	(4)	+1190 (+2080	(4)	+356 (+622)	3T	-279.0 (-488)	3T	-500.0 (-875)	4S	-98.8(2) (-173)

(1) Mission Phases are: 1 INT 20 Boost, 2 EOS Boost, 3T Tandem lunar landing, 4G Geosynchronous orbit insertion, 4S Space firing (See Figure A-5)

(2) Load occurs on the lower module of two module configuration.

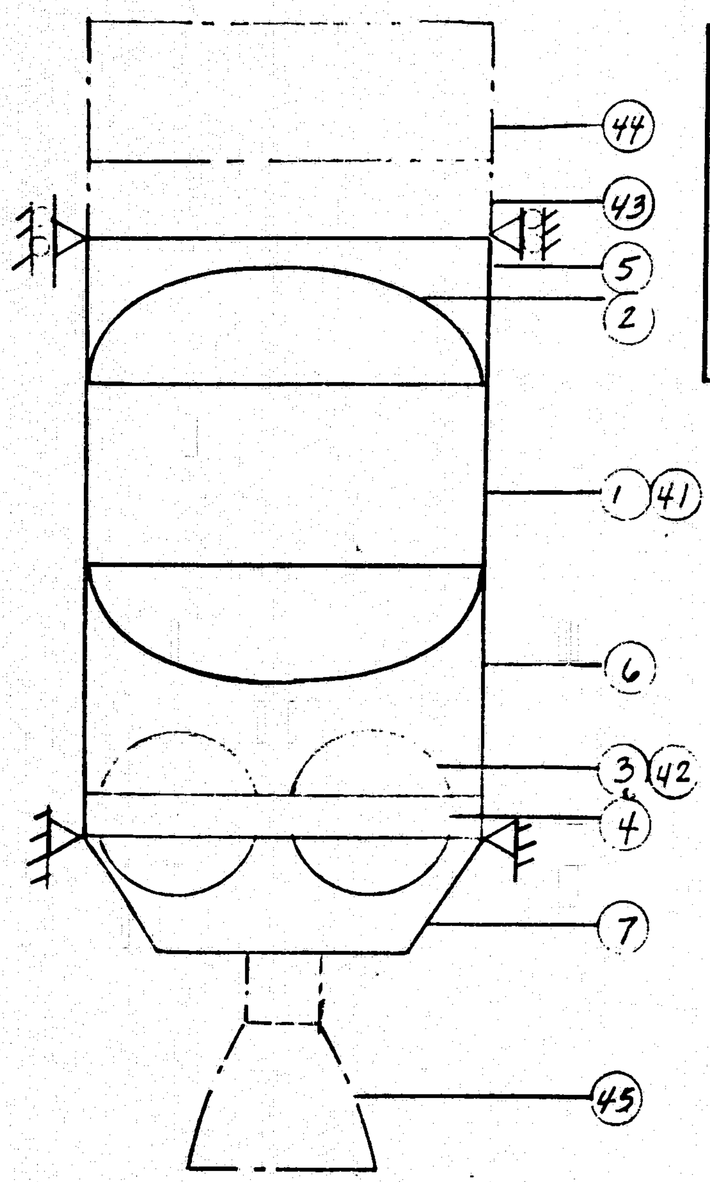
(3) Limit Loads

(4) Tank is sizes by internal pressure. LH_2 tank sidewall does not go into compression during any phase

MISSION PHASE		SATURN LAUNCH	EOS LAUNCH	LUNAR LANDING TANDEM PM BOTTOM LOADED (1)	TUG FIRING GEOSYNCH. ORBIT INSERTION	TUG FIRING SPACE TRAVEL
SPACE TUG CONFIGURATION						
CONTROL		UNMANNED	UNMANNED	MANNED	UNMANNED	MANNED
g LOAD	LONG	5	4	8	0.5	0.5
	LAT.	0.1	2.5	2.5	0	0

(1) LOWER PM IS JETTISONED BEFORE LOADING AND THE UPPER PM HAS 2/3 OF PROPELLANT AT LANDING

Figure A-5. Space Tug Loading for Mission Capability Study



	COMPONENT	CONSTRUCTION	MATERIAL
1	LH_2 TANK CYL	INTEG SKIN STRINGER	2014-T6
2	LH_2 TANK HEADS (2)	MONOCOQUE	2014-T6
3	LO_2 TANK HEADS (2)	MONOCOQUE	2014-T6
4	TANK SUPPORT		
5	FWD ADAPTER	SKIN STRINGER	BORON EPOXY
6	INTERTANK STRUCTURE	SKIN STRINGER	BORON EPOXY
7	THRUST STRUCTURE	SKIN STRINGER	BORON EPOXY

41 LH_2 PROPELLANT
42 LO_2 PROPELLANT
43 INSTRUMENT MODULE
44 PAYLOAD
45 ENGINE
46 DOCKING RING

Figure A-6. Space Tug Structure Concept for Mission Capability Study

CASE I, EOS DRY LAUNCH + GEOSYNC ORBIT + SPACE TRAVEL
CASE II, EOS WET LAUNCH + GEOSYNC ORBIT + SPACE TRAVEL
CASE III, SATURN V WET LAUNCH + GEOSYNC ORBIT + SPACE TRAVEL
CASE IV, EOS WET LAUNCH + GEOSYNC ORBIT + SPACE TRAVEL + LUNAR LANDING

PROPELLANT WEIGHT = 30,000 LB (13,600 Kg)

STRUCTURE = TANKS + SUPPORT STRUCTURE

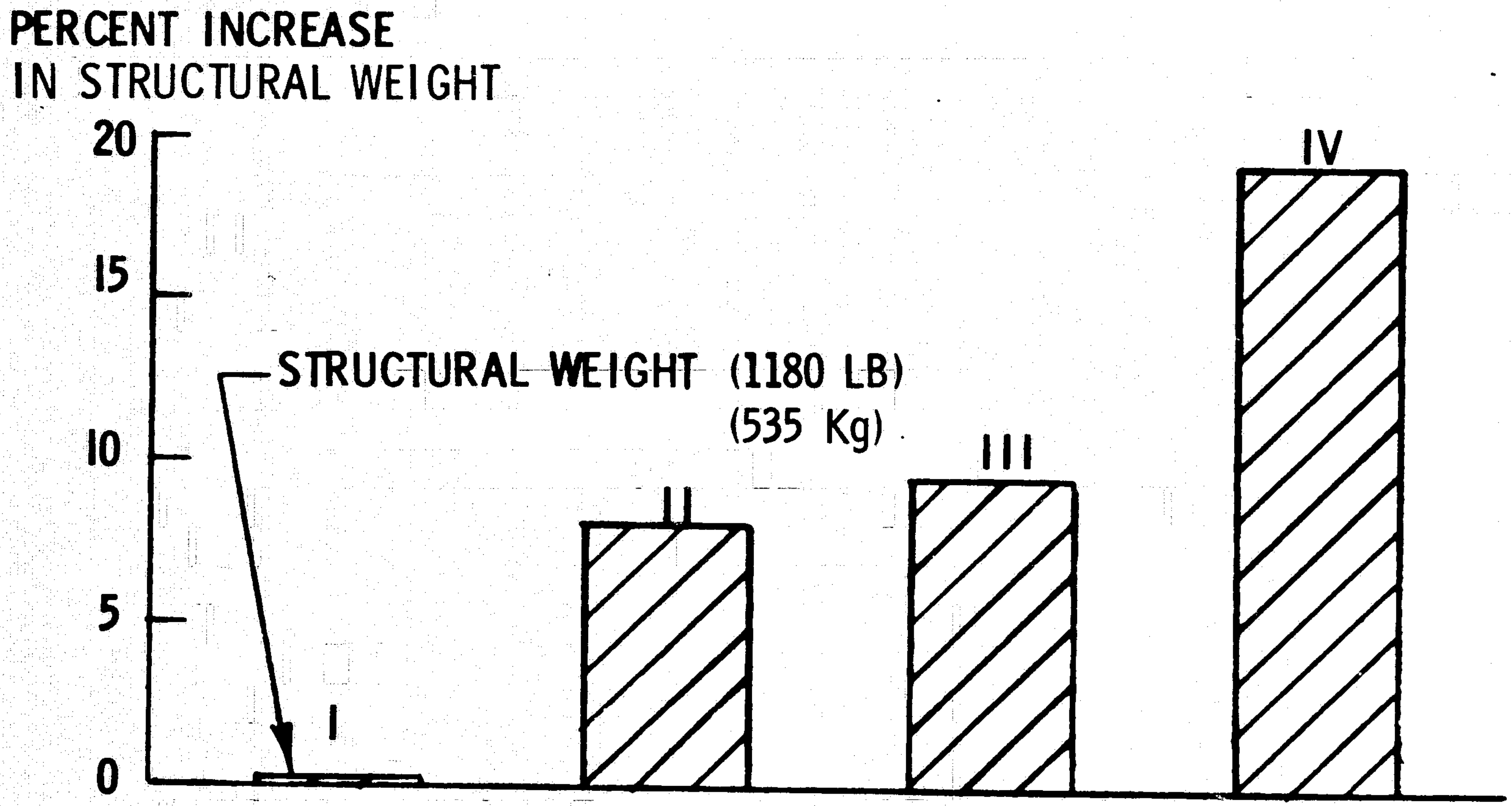


Figure A-7. Propulsion Module Mission Capability Study Results

4. Changing from EOS to INT-21 launch decreased the critical load in the forward adapter and the intertank structure due to the lower lateral acceleration used with the INT-21. Due to minimum gage limitations, there was no reduction in weight of the adapter or the intertank structure. However, the increase in axial acceleration requires additional strength in the LO_2 tank support resulting in an overall increase in structure weight for Case III relative to Case II.

5. Adding the lunar landing phase requirement in Case IV resulted in an increase in the critical loading on the forward adapter, intertank structure, and the LO_2 tank support, relative to the critical loadings for other cases. Here, the increase in axial and lateral accelerations overcame the reduction in propellant to two-thirds of capacity during landing. The ultimate load on the intertank structure due to lunar landing is well above that carried by a minimum gage boron epoxy shell; therefore, weight of this component went up to provide adequate strength.

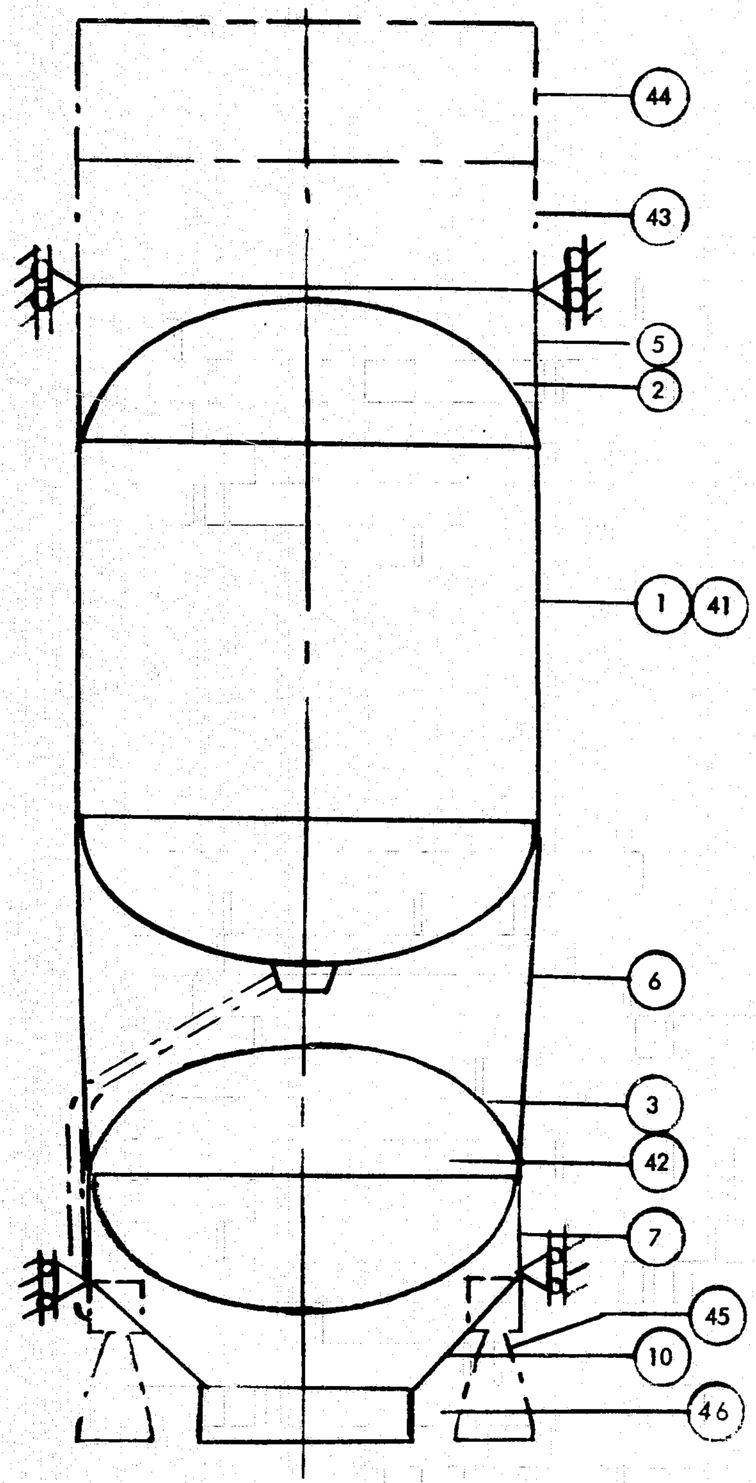
6. The findings for Case IV are based upon the assumption that for lunar landing a separate space tug landing platform would be employed which would introduce all loads on the PM at the same lower station where it is attached during launch, as shown on Figure A-5. With this landing platform the side payload attaches only to the platform for both the tandem and parallel space tug versions. If the landing gear and/or the side payload are attached to the PM at other points, the effect of lunar landing loads on the structure would be much greater and the PM structure weight would increase substantially over the values presented for Case IV in Table A-2 and Figure A-5.

7. The above findings were in regard to the 30,000 pound PM. Similar results were obtained for the box 60,000 pound PM except that loadings were substantially higher, and minimum gage limit did not apply as often.

INTEGRAL VERSUS NON-INTEGRAL PROPELLANT TANKS

The most basic characteristic of the space tug structure is the LH_2 and LO_2 propellant tank mounting and the choice made affects structure weight, structure cost, and propellant boiloff. In this study phase, integral and non-integral mounting concepts, representative of final baseline space tug concepts, were compared to resolve the tank mounting question.

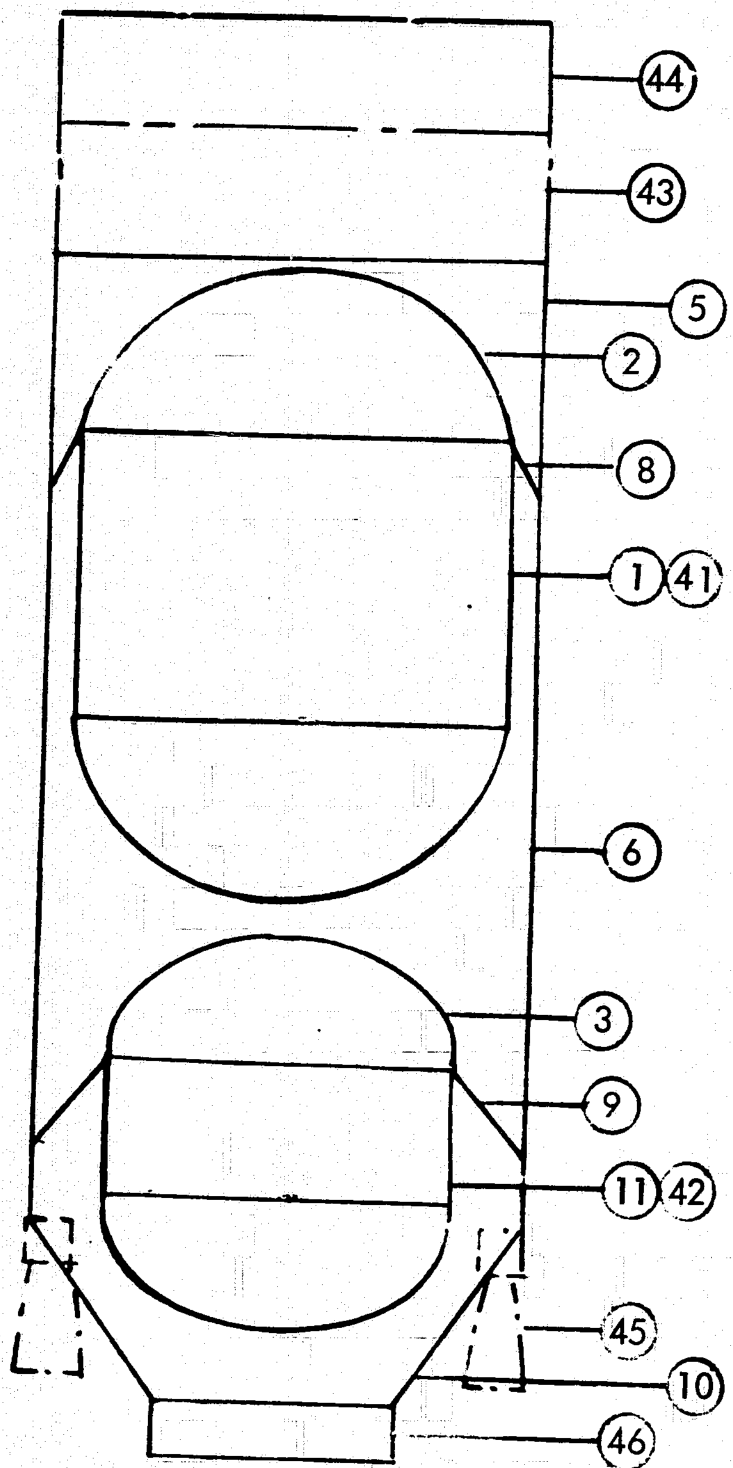
The concepts evaluated (Figures A-8 and A-9) were 60,000 pound propellant capacity. Both boron-epoxy and titanium alloy were considered



	COMPONENT	CONSTRUCTION	MATERIAL
1	LH_2 TANK CYLINDER	INTEG SKIN STRINGER	2014-T6
2	LH_2 TANK HEADS (2)	MONOCOQUE	2014-T6
3	LO_2 TANK HEADS (2)	MONOCOQUE	2014-T6
4	TANK SPACING		
5	FWD ADAPTER	SKIN STRINGER	BORON EPOXY, Ti
6	INTERTANK STR	SKIN STRINGER	BORON EPOXY, Ti
7	THRUST STR	SKIN STRINGER	BORON EPOXY, Ti
8	LH_2 TANK SUPT		
9	LO_2 TANK SUPT		
10	DOCKING RING SUPT	SKIN STRINGER	7075-T6

41 LH_2 PROPELLANT
42 LO_2 PROPELLANT
43 INSTRUMENT MODULE
44 PAYLOAD
45 ENGINE
46 DOCKING RING

Figure A-8. Space Tug Integral Structure Concept—60K Pounds Propellant



	COMPONENT	CONSTRUCTION	MATERIAL
1	LH_2 TANK CYLINDER	MONOCOQUE	2014-T6
2	LH_2 TANK HEADS (2)	MONOCOQUE	2014-T6
3	LO_2 TANK HEADS (2)	MONOCOQUE	2014-T6
4	TANK SPACING		
5	FWD ADAPTER	SKIN STRINGER	7075-T6
6	INTERTANK STR	SKIN STRINGER	7075-T6
7	THRUST STR	SKIN STRINGER	7075-T6
8	LH_2 TANK SUPT	SKIN STRINGER	BORON EPOXY, Ti
9	LO_2 TANK SUPT	SKIN STRINGER	BORON EPOXY, Ti
10	DOCKING RING SUPT	SKIN STRINGER	7075-T6
11	LO_2 TANK CYLINDER	MONOCOQUE	2014-T6

41 LH_2 PROPELLANT
42 LO_2 PROPELLANT
43 INSTRUMENT MODULE
44 PAYLOAD
45 * ENGINE
46 DOCKING RING

Figure A-9. Space Tug Nonintegral Structure Concept—60K Pounds Propellant

for the unpressurized structure to determine whether the results were material sensitive. The space tug mission capability was limited to EOS boost and space firing, and the space tug was inverted in the EOS (Figure A-9) to be compatible with the final baseline interface concept. EOS load factors were reduced and brought into agreement with load factors being used in the NR-SD shuttle study.

Analysis input data are presented in Table A-1. The factor of safety on the tanks was increased from 1.4 to 2.0 to compensate for the increased fracture resistance required for multiple pressurizations. Structural component sizes were computed along with the associated optimum boiloff and insulation thickness values (Tables A-3 and A-4). The module payload fraction values were then computed (Table A-5) leading to these findings:

1. The ultimate compression loadings in the unpressurized structural components were less than 250 pounds per inch (436 newtons per centimeter) for the load factors of Figure A-10, with the result that minimum gage construction was selected for these members.

2. A factor of safety of 2.0 was applied to the tanks to account for the multi-flight requirement. The increase from 1.4 to 2.0 did not increase tank weight because they are minimum gage limited. Pressure could be raised from 20 to 26 psi (137.5×10^6 to 179×10^6 newtons per square meter) without a change in tank thickness.

3. The integral concept has lower structure weight, higher insulation weight, and the same boiloff as the non-integral concept. The lower structure weight of the integral concept offsets the increased insulation, with the result that its payload fraction is approximately 2 percent higher.

4. The thermal analysis of the integral concept was conservative in treatment of the heat leak through the intertank structure (Component 6, Figure A-7), and a more rigorous treatment would show the payload fraction of the integral concept to be higher than 2 percent.

5. Considering its indicated performance advantage and its greater stability, the integral structure was recommended as prime candidate for space tug applications.

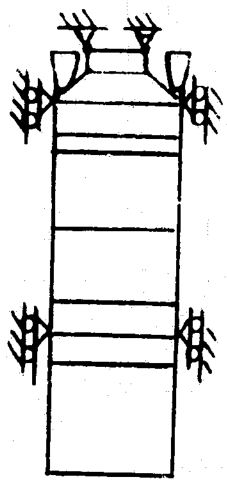
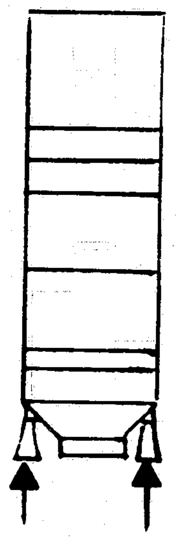
	EOS MANEUVERS		SPACE FIRING	
				
LOADS	BOOST(FULL)	LANDING (EMPTY)	FIRING (FULL)	FIRING (EMPTY)
LONG.	3.30	-1.30	0.43	1.77
LAT.	0.85	2.75	0	0

Figure A-10. Space Tug Loadings for Comparison of Integral Versus Nonintegral Structure

6. For a 7-day mission the payload fraction for the boron-epoxy concept was nominally 1 percent higher than that computed for the titanium concept, and its advantage increased to 3 percent for a 45-day mission. Based on previous analysis it is estimated that the performance of a concept using glass-epoxy for unpressurized structure would lie between that computed for the above two materials.

Table A-3. Thermal Protection Data Summary - 60K Integral Concept

Str Mat'l	Heat Leak Path	Max Insul Lgth inch (cm)	Min Heat Leak B/hr	Time = 7 Days B	Q_i	L_i	τ_i	W_i	Time = 25 Days B	Q_i	L_i	τ_i	N_i	Time = 45 Days B	Q_i	L_i	τ_i	W_i
Boron Epoxy	LH_2 Tank Sur	--	--	0.019	--	--	1.0 (2.54	--	0.038	--	--	2.0 (5.08)	--	0.050	--	--	3.0 (7.60)	--
	Fwd Adpt	58 (147)	--	0.019	--	58 (147.5)	1.0 (2.54)	--	0.038	--	58 (147.5)	2.0 (5.08)	--	0.050	--	58 (147.5)	3.0 (7.60)	--
	Int Str	63 (160)	--	0.019	--	63 (160)	1.0 (2.54)	--	0.038	--	63 (160)	2.0 (5.08)	--	0.050	--	63 (160)	3.0 (7.60)	--
	Σ							291 (132)					582 (264)					873 (396)
	LO_2 Tank Sur	--	--	0.007	--	--	0.5 (1.27)	--	0.018	--	--	1.2 (2.11)	--	0.015	--	--	2.0 (5.08)	--
	Int Str	63 (160)	--	0.007	--	63 (160)	0.5 (1.27)	--	0.018	--	63 (160)	1.2 (2.11)	--	0.015	--	63 (160)	2.0 (5.08)	--
	Thrust Str	40 (101.5)	--	0.007	--	40 (101.5)	0.5 (1.27)	--	0.018	--	40 (101.5)	1.2 (2.11)	--	0.015	--	40 (101.5)	2.0 (5.08)	--
	Σ							100 (45)					236 (107)					394 (179)
Titanium	LH_2 Tank Sur	--	--	0.021	--	--	1.25 (3.18)	--	0.050	--	--	2.5 (6.35)	--	0.076	--	--	3.75 (9.55)	--
	Fwd Adapter	58 (147)	--	0.021	--	58 (147.5)	1.25 (3.18)	--	0.050	--	58 (147.5)	2.5 (6.35)	--	0.076	--	58 (147.5)	3.75 (9.55)	--
	Int Str	63 (160)	--	0.021	--	63 (160)	1.25 (3.18)	--	0.050	--	63 (160)	2.5 (6.35)	--	0.076	--	63 (160)	3.75 (9.55)	--
	Σ							364 (165)					762 (346)					1090 (495)
	LO_2 Tank Sur	--	--	0.0072	--	--	0.62 (1.58)	--	0.016	--	--	1.5 (3.81)	--	0.024	--	--	2.5 (6.35)	--
	Int Str	63 (160)	--	0.0072	--	63 (160)	0.62 (1.58)	--	0.016	--	63 (160)	1.5 (3.81)	--	0.024	--	63 (160)	2.5 (6.35)	--
	Thrust Str	40 (101.5)	--	0.0072	--	40 (101.5)	0.62 (1.58)	--	0.016	--	40 (101.5)	1.5 (3.81)	--	0.024	--	40 (101.5)	2.5 (6.35)	--
	Σ							122 (56)					294 (134)					490 (222)

B = total propellant boiloff fraction
Q_i = heat leak through path i, B/hr (J/hr)
L_i = insulation length, inches (cm)
τ_i = insulation thickness, inches (cm)
W_i = insulation weight, lb (kg)

Table A-4. Thermal Protection Data Summary - 60K Non Integral Concept

Str Mat'l	Heat Leak Path	Max Insul Lgth inch (cm)	Min Heat Leak B/hr	Time = 7 Days B	Q_i	L_i	τ_i	W_i	Time = 25 Days B	Q_i	L_i	τ_i	W_i	Time = 45 Days B	Q_i	L_i	τ_i	W_i
Boron Epoxy	LH_2 Tank Sur	--	--	0.018	--	--	1.0 (2.54)	--	0.034	--	--	2.0 (5.08)	--	0.048	--	--	3.0 (7.63)	--
	LH_2 Tank Supt	2.5 (63.5)	--	0.018	--	2.5 (63.5)	1.0 (2.54)	--	0.034	--	25 (63.5)	2.0 (5.08	--	0.048	--	25 (63.5)	3.0 (7.63)	--
	Σ							194 (88)					386 (175)					580 (363)
	LO_2 Tank Sur	--	--	0.005	--	--	0.5 (1.27)	--	0.010	--	--	1.2 (3.05)	--	0.011	--	--	2.0 (5.08)	--
	LO_2 Tank Supt	40 (101.5)	--	0.005	--	40 (101.5)	0.5 (1.27)	--	0.010	--	40 (101.5)	1.2 (3.05)	--	0.011	--	40 (101.5)	2.0 (5.08)	--
	Σ							69 (31.3)					166 (75.5)					276 (125)
Titanium	LH_2 Tank Sur	--	--	0.017	--	--	1.25 (3.18)	--	0.041	--	--	2.5 (6.35)	--	0.065	--	--	3.75 (9.55)	--
	LH_2 Tank Supt	25 (63.5)	--	0.017	--	25 (63.5)	1.25 (3.18)	--	0.041	--	25 (63.5)	2.5 (6.35)	--	0.065	--	25 (63.5)	3.75 (9.55)	--
	Σ							242 (111)					485 (220)					725 (329)
	LO_2 Tank Sur	--	--	0.0056	--	--	0.62 (1.58)	--	0.011	--	--	1.5 (3.81)	--	0.015	--	-- 40 (101.5)	2.5 (6.35)	--
	LO_2 Tank Supt	40 (101.5)	--	0.0056	--	40 (101.5)	0.62 (1.58)	--	0.011	--	40 (101.5)	1.5 (3.81)	--	0.015	--	40 (101.5)	2.5 (6.35)	--
	Σ							86 (39)					207 (94)					345 (156.5)

B = total propellant boiloff fraction
Q_i = heat leak through path i, B/hr (J/hr)
L_i = insulation length, inches (cm)
τ_i = insulation thickness, inches (cm)
W_i = insulation weight, lb (kg)

Space Division
North American Rockwell

Table A-5. Final Results of Integral Versus Nonintegral Structure Comparison

Integ. Structure Concept	Unpress. Structure Material	Mission Duration (T) Days	Structure Weight Lb (kg)	Insulation Weight Lb (kg)	Boiloff Functions		Payload Fraction (W_p/W_{ig})
					LH_2	LO_2	
Integral	Boron Epoxy	7	1520 (690)	390 (177)	0.019	0.0070	0.538
		25	1520 (690)	820 (372)	0.038	0.0130	0.527
		45	1520 (690)	1270 (576)	0.050	0.0150	0.513
	Titanium	7	1720 (780)	470 (213)	0.021	0.0072	0.534
		25	1720 (780)	1020 (464)	0.050	0.0164	0.520
		45	1720 (780)	1580 (718)	0.076	0.0238	0.496
Non Integral	Boron Epoxy	7	2080 (945)	260 (118)	0.018	0.005	0.530
		25	2080 (945)	550 (250)	0.034	0.010	0.522
		45	2080 (945)	860 (391)	0.048	0.011	0.510
	Titanium	7	2200 (1000)	330 (150)	0.017	0.0056	0.523
		25	2200 (1000)	690 (313)	0.041	0.0113	0.514
		45	2200 (1000)	1070 (485)	0.065	0.0155	0.500

APPENDIX B

RELIABILITY

By

K. Relyea

APPENDIX B. RELIABILITY AND SAFETY ANALYSIS

OBJECTIVE AND APPROACH

The objective of the reliability and safety studies was to examine the design and operations of the space tug with a view to pointing out both ways and means of obtaining maximum mission success and crew safety as an aid to design. In addition, an objective was to determine and justify all single point failures in tug design and operation.

The approach taken was to follow closely the conceptual design effort and the subsystems and operations analyses as they were developed in the study. Mission success path logic diagrams were prepared for the various tug mission phases.

SUCCESS CRITERIA

To perform a reliability and safety analysis of the space tug the following success criteria, based on the statement of work, were identified.

Mission Success - The probability of safe continuation at any point in a mission (S.O.W. 3.1.1)

Safe Continuation - The ability to pursue actively all space tug functions and objectives while retaining adequate safety (S.O.W. 3.1.1)

Crew Safety - The probability of crew survival at any point in a mission (S.O.W. 3.1.1)

Crew Survival - All crewmen returned to a haven without loss of life (S.O.W. 3.1.1)

Safety - The joint probability of tug return to a haven, no damage to other space vehicles, and crew safety

Single Failure Points - The number of single systems whose failure will lead directly to (1) loss of life, (2) loss of hardware, or (3) unscheduled mission termination (S.O.W. 4.2.4)

Mission success is an excellent measure of space tug capability. All success paths which allow safe mission continuation in support of a balanced, low cost Integrated Program Plan are included: earth orbital, lunar, and planetary (S.O.W. 1.1). As a probability, mission success allows a realistic appraisal of the frequency of occurrence of safe continuation of many vehicles throughout the projected program operational phase. Application to any single mission or mission phase is made to indicate the degree of performance repeatability. Sequential application indicates the conditional probability of success from any early mission phase to mission completion.

Crew safety limits the objective to crew survival, but increases the number of sequential logic paths to success. All successful abort paths are considered as well as all mission success paths. From the launch pad, crew safety is entirely mission oriented. From intermediate mission phases, crew safety covers the remainder of the mission and mission aborts.

In order to characterize economical operation as well as to provide a complete definition of safety, the considerations of preventing damage to other vehicles by the tug and the return of tug hardware to a safe haven must be added to crew safety objectives. Any hardware loss represents an additional expense for replacement, retrieval, repair, and time to proceed with the next mission. Large interfacing vehicles (e.g., EOS, and EO space station) must be protected from damage by a space tug because of the cost amplification. The same probabilistic sequential treatment as before can be made.

Single failure points must be identified for possible crew and mission loss without warning. Recognition is made of the stochastic limitation wherein single vehicles are characterized as being either successful or not on any given mission, i.e., any single failure point can occur on any mission. Reduction of single failure points to zero is the only sure way to prevent their occurrence, but a low number coupled with preventive analysis and design for each is an indicator of engineering confidence in no failure.

REDUNDANCY PHILOSOPHY

Enhancement of success through redundancy is a necessary part of design. All facets of the design problem are involved since the redundant equipment or operations consume space, time, and resources. Redundancy is an accepted means for success enhancement by provision of more operating modes for completing any particular success criterion. The

increased complexity and performance loss must be traded against the success gain. Extended development for success enhancement of a non-redundant design is usually excessively long because of the high success goals. Redundancy decisions prior to detailed design must be based on general mission requirements and/or similarities of hardware to be used with that used on earlier programs. The several kinds of redundancy considered for the tug are:

On-line - Completely functioning as part of normal operation and still capable of mission performance following other failure with no need for detection and switching. At least one on-line redundancy should be required for all time-critical failures.

Standby - Capable of mission performance, functioning quiescently before main element failure, but requiring some positive switching operation before being inserted in the mission-performing system. Detection of main element failure is required. Insertion means depends on reaction time, weight penalty, and operation capability; e. g., switch or patch panel.

Spare - Capable of mission performance, but stored separately from the mission-performing system in any convenient location and inserted into the mission system following other failure. Detection of failure is required, and insertion time must be available. Location of spares could be on board the Space Tug or any convenient vehicle, at the LSB, and on the earth.

The space tug mission requires consideration of all three types of redundancy. Mission critical equipment utilizes on-line redundancy to preclude any cessation of function. Where reaction time allows or function prevents on-line continuous attachment, standby redundancy may be used, but provides a mixed blessing: it increases long-duration reliability, but requires detection and positive action before failed function is restored. Completion of at least ten uses or three years in space (the tug mission) can be accomplished by replacement of failed parts both in flight and on the ground. Commercial airline practice of a graduated scale of detail in inspection will be followed:

Sortie inspection - External visual check, checklist, and automatic checkout routines before and after every sortie with no disassembly.* The objectives are in-space failure detection and servicing.

Operator inspection - Crew and telemetry monitoring of mission performance for the purpose of failure detection.

*On-orbit servicing is expected to be minimal until a substantial space base facility is available.

Periodic inspection - Planned replacement of limited life components and inspection for extent of wear. Limited disassembly in space or on the ground.

Overhaul - On the ground complete disassembly, inspection, and replacement of worn parts. Initiated within the ten-sortie sequence by space tug failure or damage which is incapable of repair in space.

The urgency of repair is a function of the kind of failure. For failures which do not restrict mission operation, repair is not immediately required, if at all. Mission continuation with partial failure is possible and may be desirable when the time to the next sortie is short, the expense is high, technicians are unavailable, or spares are unavailable. For failures which affect safety, immediate repair is in order. This is particularly true when the next sortie is one which has severe performance requirements. Since the lunar landing sortie limits the extent of several redundancies, a minimum number of failures could be tolerated when the lunar landing is expected as the next sortie. In fact, special review should be made before each attempted landing with failed equipment. Where possible, the tug with failed equipment should be returned for use in earth orbit and replaced by a fresh tug in lunar orbit.

Several of the above statements have implied criteria by which the proper level of redundancy can be judged. The complex interactions of the many space tug objectives require more precise consideration of criteria and desirable development of each.

Higher Mission Success - Increasing redundancy increases mission success by adding success paths.

Higher safety - Increasing redundancy increases safety by adding success paths.

Fewer single failure points - Providing redundant equipment or operations eliminates single failure points and gives the operator a choice of continuing or aborting the mission. Complete elimination of single failure points may not be practical, e.g., propellant tanks, high pressure piping, and lunar landing gear legs. (See later section on single failures.) A decrease in single failure points is specifically desired.

Lower weight - On-line and standby redundancies add weight according to the equipment size (Figure B-1). Several strategems must be employed to prevent redundant weight from doubling or tripling. For example, weight savings (while retaining an abort

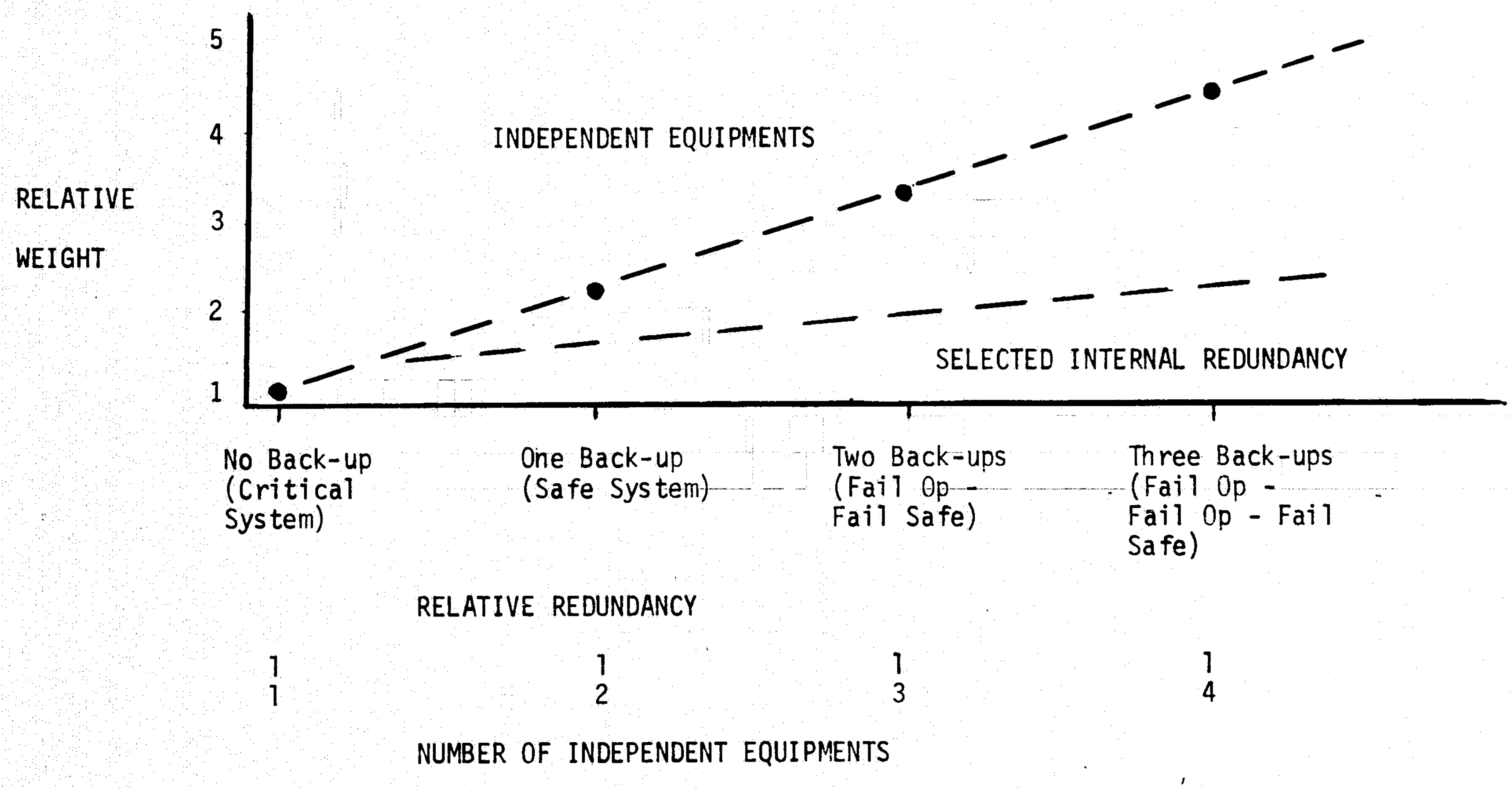


Figure B-1. Weight Trends of Redundant Equipment

capability) can be achieved by adding complexity in a special way: decrease size of redundant equipment. More pieces are needed but less weight is required for the amount of redundancy desired. Figure B-2 demonstrates this feature for general design. When either one of two subsystems could be operational, the relative weight to the minimum is 2. Decreasing the subsystem size to one-half normal and adding one redundant subsystem produces a configuration with three subsystems, but a relative weight of only 1-1/2. Continuing to decrease subsystem size (i.e., increasing the number of pieces required for full performance) has less effect on weight. Equal weight for equal performance was assumed. Detailed subsystem analyses will shift the minimum practical weight toward the left, i.e., some weight is added for increased complexity as well as performance.

Lower cost - Increased number of pieces for redundancy tends to increase cost. However, decreased development expense or decreased manufacturing cost due to lower individual piece reliability will be used as offsetting factors. (S.O.W. 4.0).

Less complexity - More pieces usually means more complexity. This reflects adversely on inspection time, number of measurements, and expectation of operational difficulty. On-line and standby redundancies are particularly susceptible to equipment complexity problems and spares increase operational complexity by requiring more in-space replacements. The S.O.W. has specifically noted this as undesirable, but complexity can be acceptable with design for easy replacement (S.O.W. 3.1.1).

Current redundancy considerations have been couched in terms of positive design requirements: e.g., "fail operational - fail operational - fail safe" for major electronic components. This is interpreted to mean that normal mission operation can continue following one and two failures and that safe return to the ground is still possible following the third major component failure in any particular subsystem. Further, the early EOS requirement of "fail operational - fail safe" for major mechanical components means normal mission operation following one failure and safe return following a second failure.

It is noted that the direct specification of such redundancy has been deleted in current EOS study. Specification of arbitrary, blanket, predesign redundancy requirements for all space tug subsystems is too restrictive at this time and emphasizes mechanical acquiescence rather than conceptual innovation. The major effort would become the arrangement of multiple subsystems or the specification of entirely new equipment. The latter would require extensive development for low weight rather than for

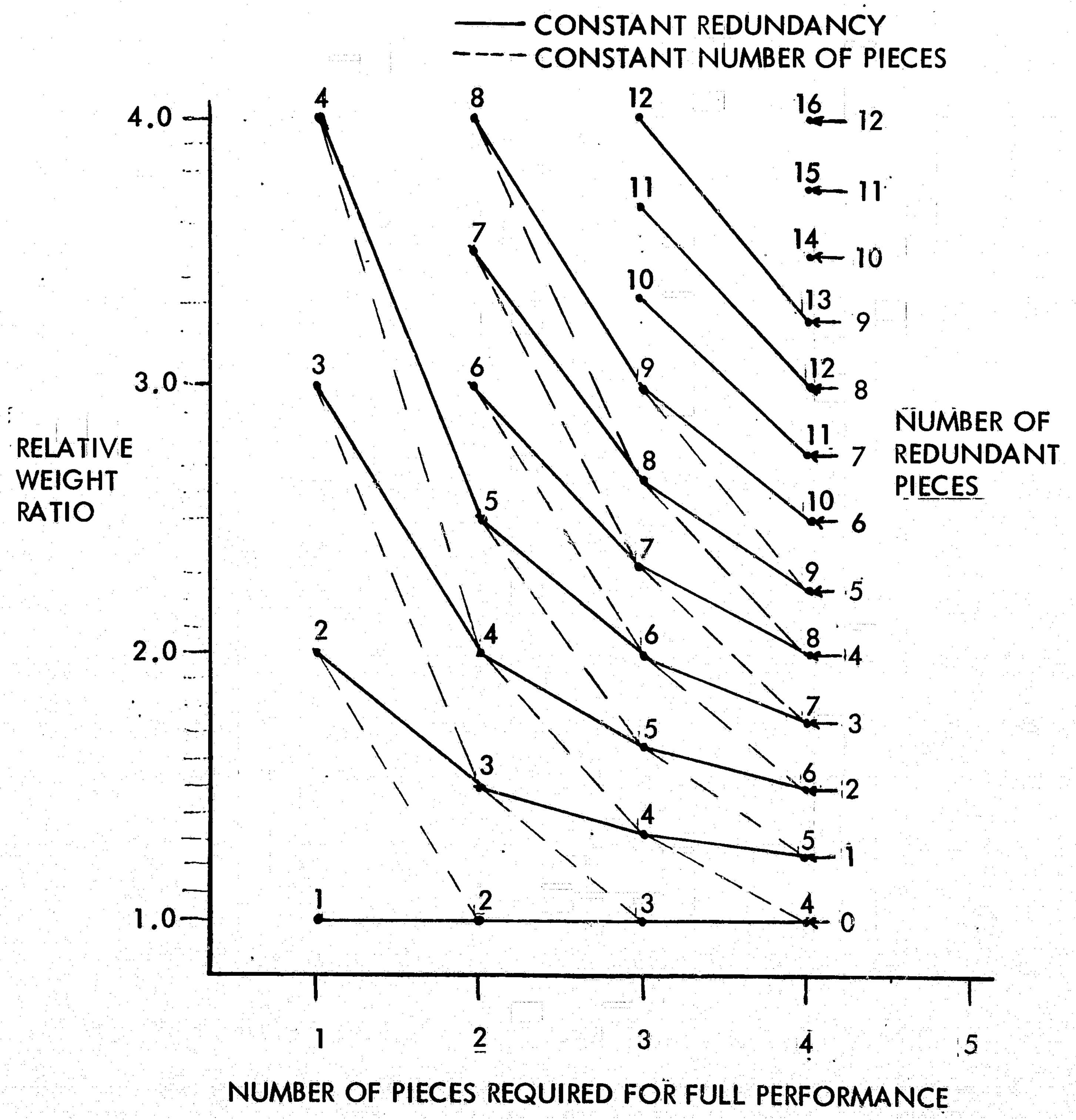


Figure B-2. Saving Weight by Increasing Complexity

the high reliability which prompted the redundancy. Alternatively, individual subsystem characteristics for accomplishing each mission phase must be considered in the redundancy selection in order to provide an optimum mix of criteria levels; e.g., non-critical subsystems require less redundancy than critical subsystems. In addition, the assembly level at which redundancy is to be applied varies with the subsystem. Electronics subsystems could obtain a large degree of cross connection flexibility by utilizing detailed component redundancy, while main engines are best made redundant in the larger size. Certainly, a separate standby rescue tug is redundancy on a very large scale.

Space tug mission characteristics give special meaning to the redundancy nomenclature. "Fail operational" really means that the space tug remains in space and continues to perform successive missions after a failure. Succeeding sorties may have many failed components without the necessity of abort. The space tug must be capable of performing any mission until a final "fail safe" failure which initiates abort. The essence of "fail operational" redundancy is long life, and the essence of "fail safe" redundancy is crew survival and hardware return.

Frequent return of the space tug to the ground would assist in reducing redundancy. The mission could be selected in advance of ground launch, and the optimum vehicle configuration could be assembled on the ground. Sophisticated and positive checkout could be achieved. Better individual mission performance would result at the expense of flight readiness, additional ground launches, and some mission flexibility. However, frequent return of the tug to the ground could be against the criterion of lower cost. When should the space-based tug be returned to the ground?

After every sortie? - No. In-space automatic checkout is sufficient.

After a fixed number of sorties? - Yes. Ten (10).

After a fixed time? - Yes. Three (3) years in space.

After any "fail operational" failure? - No. Enough redundancy has been included for continued operation by definition.

After a "fail safe" failure? - Only when repair or replacement cannot be performed in space.

Never? - No. Airline practice of increasing the overhaul time limit as additional experience is gained will be followed.

In summary, the salient features of Space Tug redundancy are listed below and functionally illustrated in Figure B-3.

1. Minimum feasible single failure points - (goal of no single failure points).
2. No credible combinations of failures which result in crew loss.
3. Tradeoff long life with multiple standbys.
4. Utilize onboard standby and on-line redundancy in preference to replacing failed equipment in space.
5. Provide measurements for failure detection when corrective crew action is required and is possible.
6. Modularize component assemblies for on-ground replacement.
7. Provide alternate safe continuation modes.
8. Recommend design margins.
9. Identify additional study.

Redundancy should be considered when the following benefits can be achieved.

Higher mission success

Higher safety

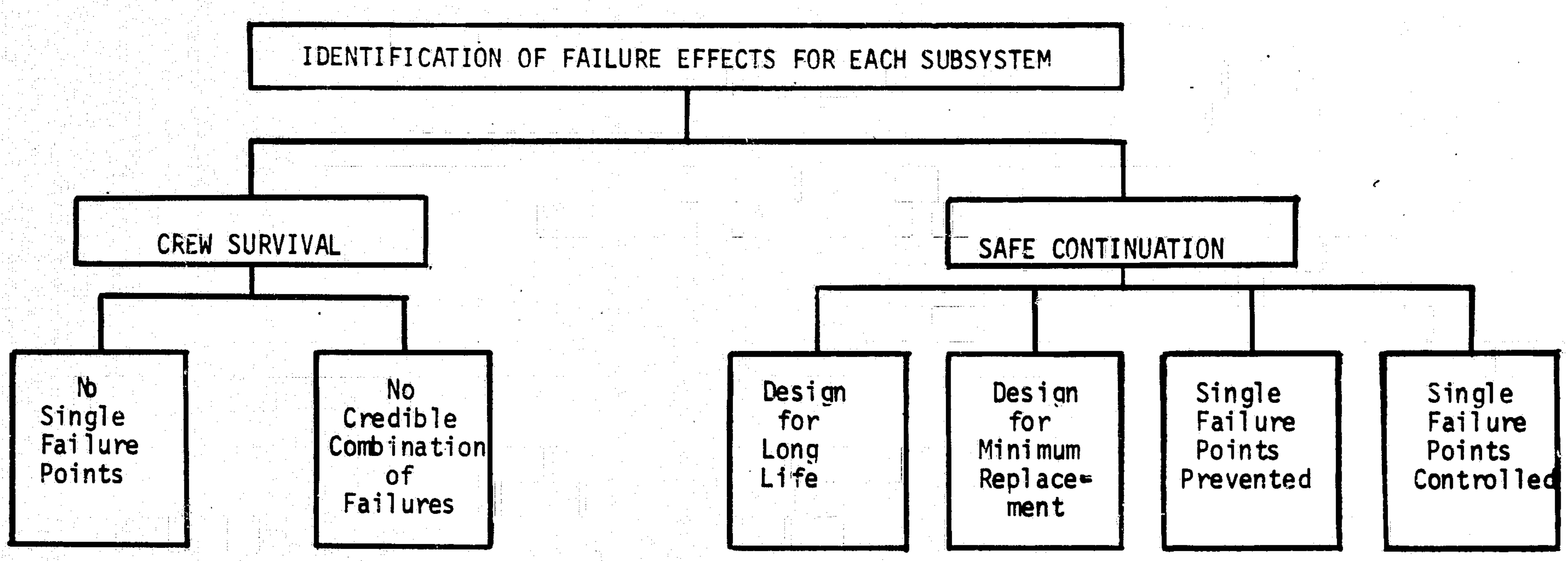
Fewer single failure points

Lower weight

Lower cost

Less complexity

Fewer in-space replacements and repairs



No Single Failure Points

- Selected Component Redundancy
- Abort Provisions
- Alternate Safe Operating Modes
- Rescue Provisions

No Credible Combination of Failures

- Preliminary Crew Safety Analysis
- Crew Safety Analysis Evaluation
- Relative Credibility Assessment
- Incredible Design

Design for Long Life

- Adequate Design Margins
- Detailed Environment Specification
- Multiple Redundancy
- Modular Spares

Design for Minimum Replacement

- On-line Redundancy
- Failure Detection
- Simple Repair
- On-board Standby
- Modular Spares

Single Failure Points Prevented

- Selected Component Redundancy
- Alternate Safe Continuation Modes

Single Failure Points Controlled

- Adequate Design Margins

Figure B-3. Redundancy Allocation

SUCCESS PATH LOGIC

Figures B-4 through B-9 are the success path logic diagrams for unmanned synchronous and manned lunar landing missions. The criteria of mission success and safety were used for the unmanned missions, and mission success and crew safety for the manned. These diagrams were based on the criticality matrices from a section later in this report. Each mission phase is shown separately for clarity. Arrangement of each diagram in the correct mission phase sequence will synthesize the entire mission. The extent of the retrieval missions, rescue missions, and aborts depends on the mission phase in which the deciding failure occurs; e. g., rescue time from lunar orbit coast is very brief compared to rescue from the lunar surface by a lunar orbiting tug.

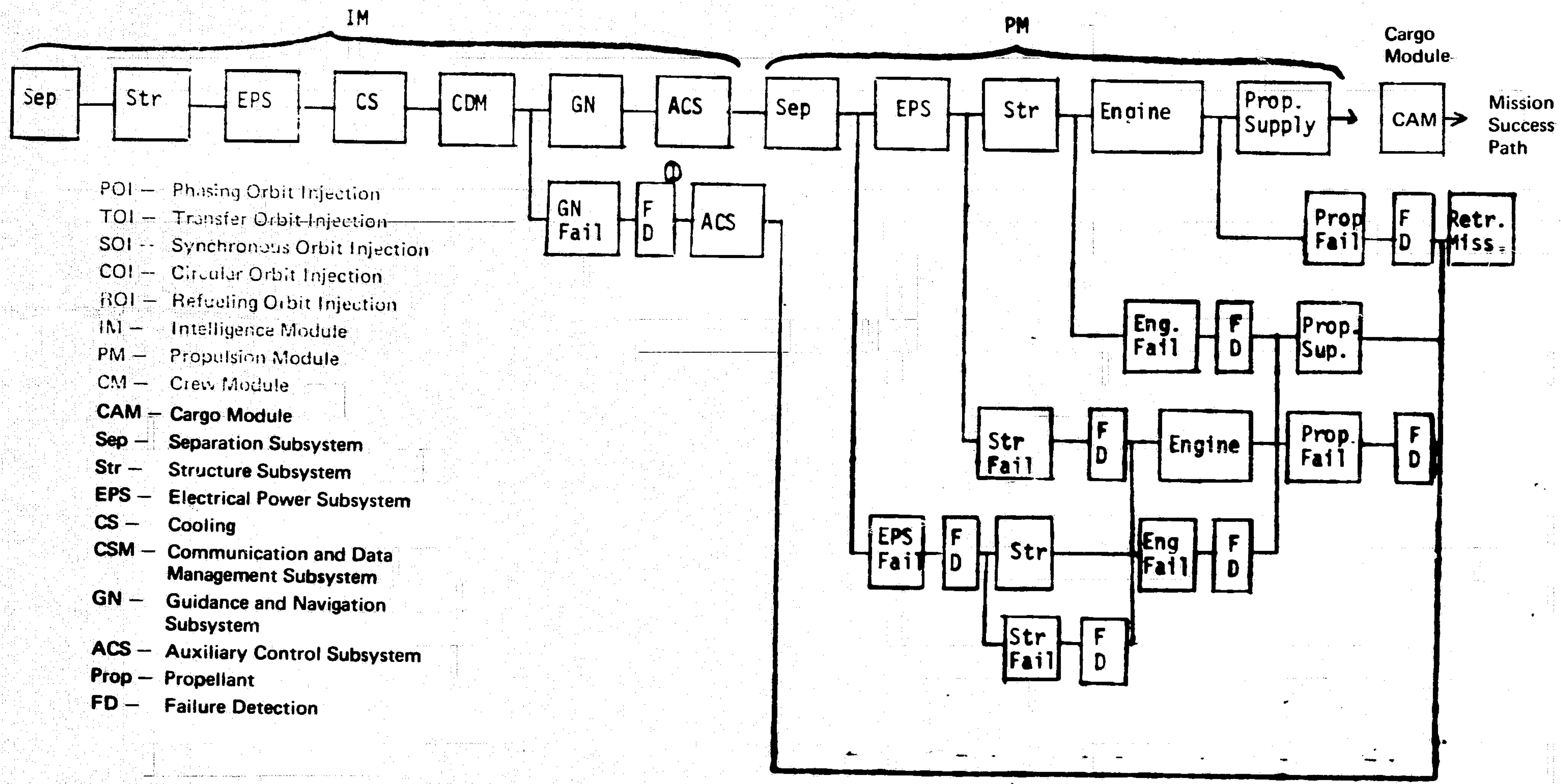
All mission phases for the unmanned mission are shown in Figure B-4. The mission success path comprises the successful operation of all subsystems in both the IM and the PM. The alternate paths include those subsystems which allow the disabled vehicle to maintain environmental and attitude control until a retrieval vehicle arrives. Most of the PM and the GN (guidance and navigation subsystem) can fail without vehicle loss. Separation subsystems must not separate inadvertently because the PM has no attitude control, and subsequent docking would be difficult, if not impossible.* Structural failure is considered as severe damage but not complete separation, and explosions are not shown. In fact, since this is a success path diagram, no paths lead to loss of safety.

The same type of diagram applies to each phase of the manned lunar landing mission. Not only are the number of subsystems increased but also the number and variety of alternate paths. In general, stable orbits and the lunar surface are places from which rescue can be made; and the PM need not be operative for crew safety. Also, the CM has some backup means for GN and CMD (communication and data management subsystem) if only the crew observation and hardwire control lines. This redundancy is primarily continued operation with either GN and either CDM. When both GN's fail, a resuce mission is required. Both CDM's cannot fail, because attitude control would be lost. Four outcomes are noted, mission success, abort intact, abort without the cargo module, and rescue mission. Before braking and landing, all subsystem success is the general criteria for mission success. From within the braking and landing phase to the end of the mission, mission continuation becomes the dominant abort criterion; so "abort intact" becomes part of mission success. Loss of the CAM and the rescue requirement are always mission success failures but crew safety successes. Tables B-1 and B-2 present abort considerations and sample states.

*Note: This applies when IM-PM interface is not ground-assembled.

For Mission Phases

POI	PO Coast
TOI	TO Coast
SOI	Payload Ops.
COI	



POI — Phasing Orbit Injection
TOI — Transfer Orbit Injection
SOI — Synchronous Orbit Injection
COI — Circular Orbit Injection
ROI — Refueling Orbit Injection
IM — Intelligence Module
PM — Propulsion Module
CM — Crew Module
CAM — Cargo Module
Sep — Separation Subsystem
Str — Structure Subsystem
EPS — Electrical Power Subsystem
CS — Cooling
CSM — Communication and Data Management Subsystem
GN — Guidance and Navigation Subsystem
ACS — Auxiliary Control Subsystem
Prop — Propellant
FD — Failure Detection

Figure B-4. Safety Logic for Unmanned Geosynchronous Missions

POI — Phasing Orbit Injection
TOI — Transfer Orbit Injection
SOI — Synchronous Orbit Injection
COI — Circular Orbit Injection
ROI — Refueling Orbit Injection
IM — Intelligence Module
PM — Propulsion Module
CM — Crew Module
CAM — Cargo Module
LG — Landing Gear
Sep — Separation Subsystem
Str — Structure Subsystem
EPS — Electrical Power Subsystem
CS — Cooling
CDM — Communication and Data Management Subsystem
GN — Guidance and Navigation Subsystem
ACS — Auxiliary Control Subsystem
Prop — Propellant
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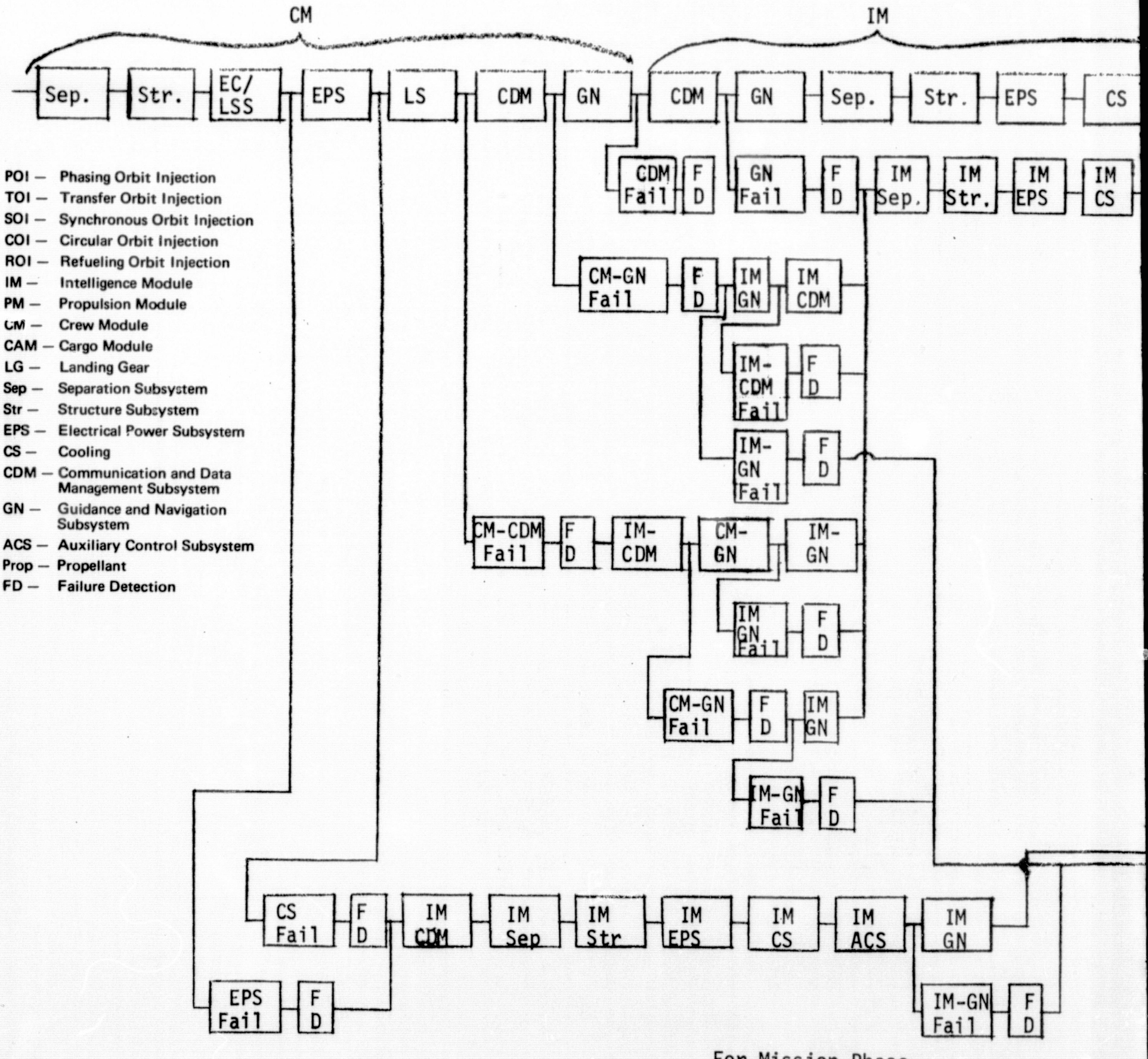
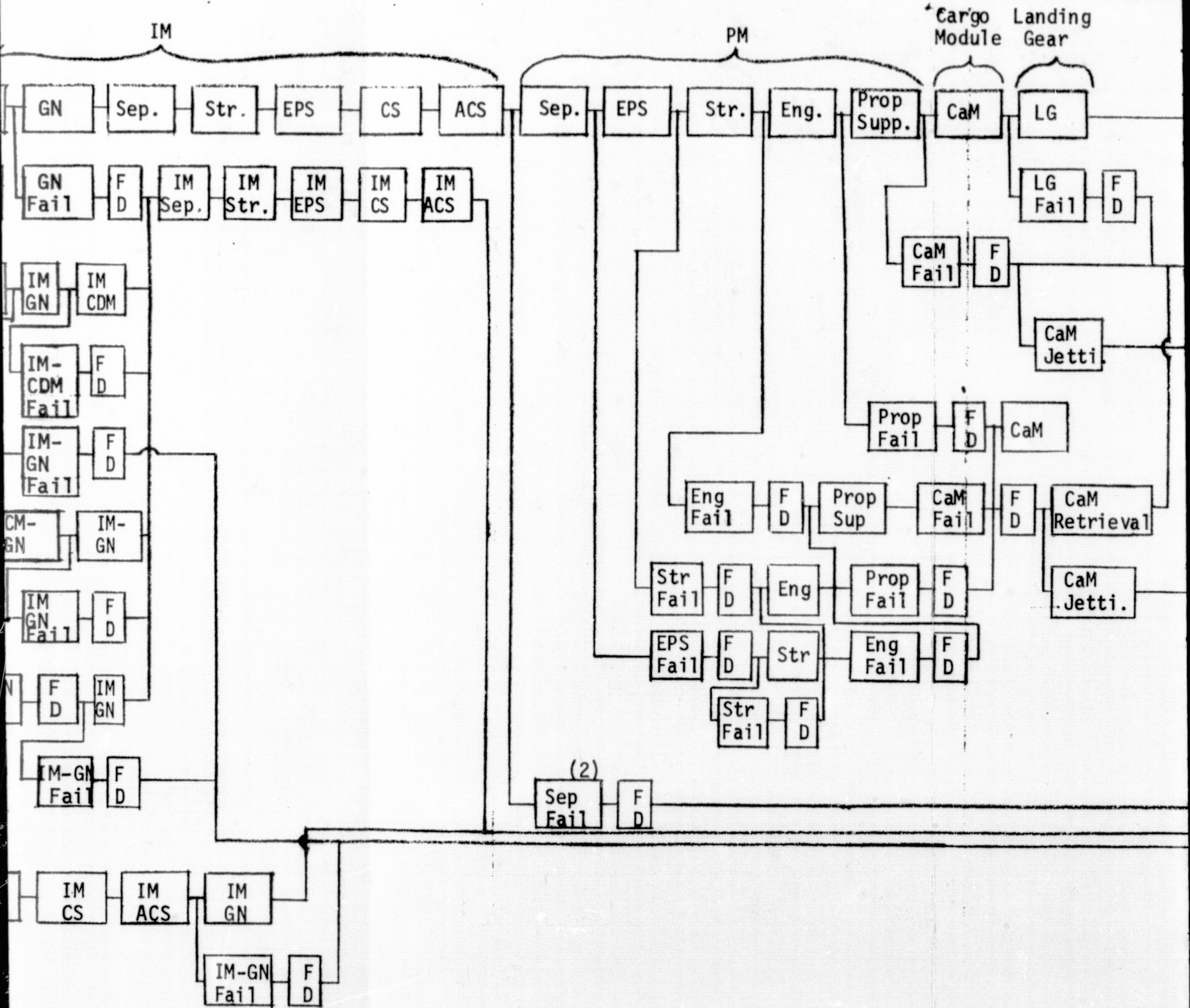


For Mission Phase
Lunar Orbit Coast - Prior to Lunar E
Lunar Orbit Coast - Following Lunar

(1) Abort intact from the second Lunar Orbit Coast following lunar surface excursion is containe
(2) For the configuration with Cargo Module attached to the Propulsion Module near the engines a

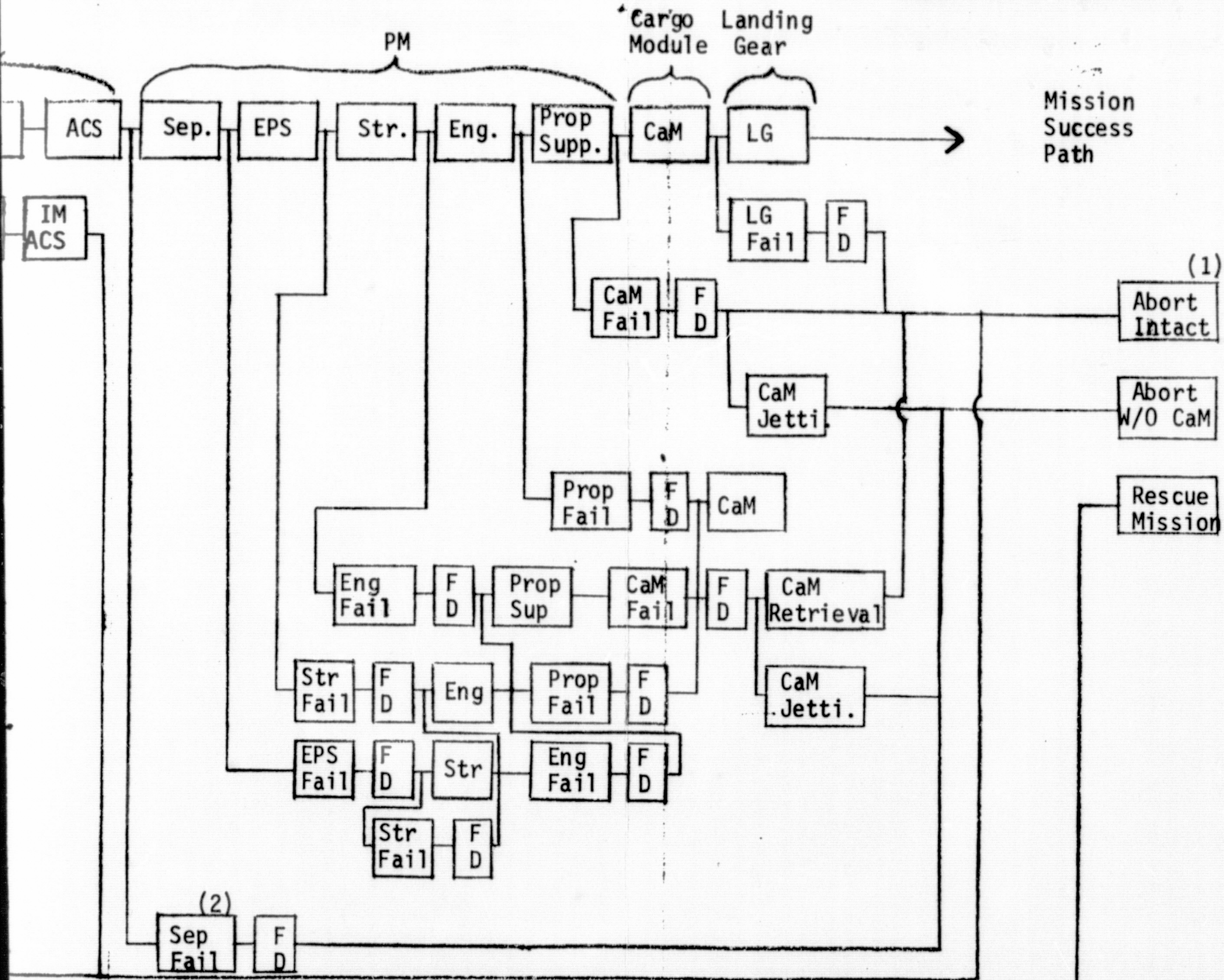
FOLDOUT FRAME

FOLDO



For Mission Phase
Lunar Orbit Coast - Prior to Lunar Excursion (Any failure is an abort or rescue return to the LOSS).
Lunar Orbit Coast - Following Lunar Excursion (Mission continuation should prevail whenever possible).
g lunar surface excursion is contained in Mission Success.
Propulsion Module near the engines and the Crew Module attached only to the Intelligence Module at the forw

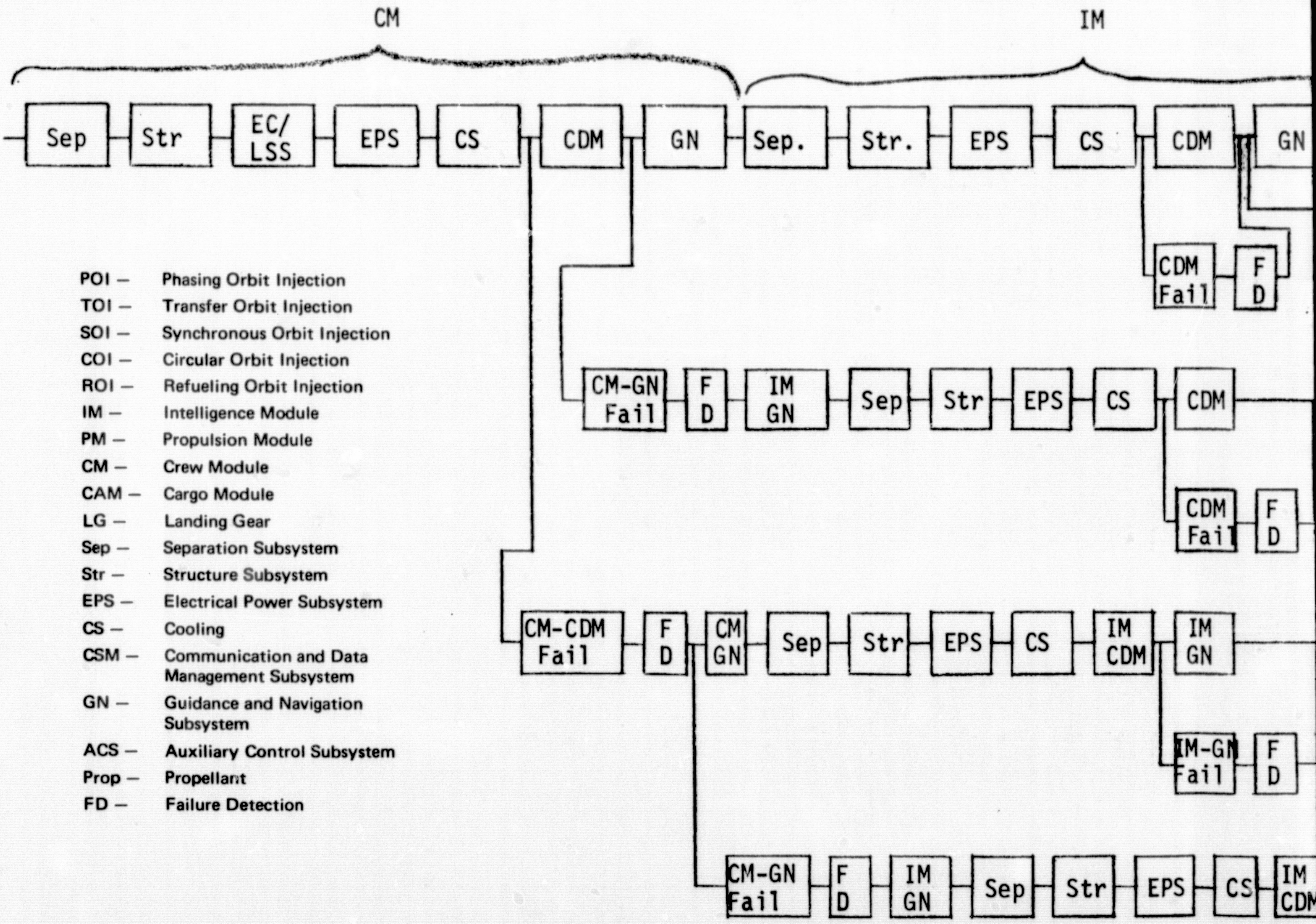
Figure B-5. Crew Safety Logic for Manned Lunar L
NASA Mode A or B



Excursion (Any failure is an abort or rescue return to the LOSS).
Excursion (Mission continuation should prevail whenever possible).
d in Mission Success.
nd the Crew Module attached only to the Intelligence Module at the forward end.

Figure B-5. Crew Safety Logic for Manned Lunar Landing Mission—NASA Mode A or B

For Mission Phases
De-orbit (Any failure in this phase is cause for abort, i.e., return to lunar orbit.)
Braking & Landing. (Mission continuation is the predominant operating mode except for cargo
Lift-off thru POI. (Mission continuation is the predominant operating mode.) (Does *not* in



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GN – Guidance and Navigation Subsystem
ACS – Auxiliary Control Subsystem
Prop – Propellant
FD – Failure Detection

FOLDOUT FRAME /

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abort, i.e., return to lunar orbit.)
predominant operating mode except for cargo loss). (Does not include Landing Gear Failure).
predominant operating mode.) (Does not include Landing Gear Failure).

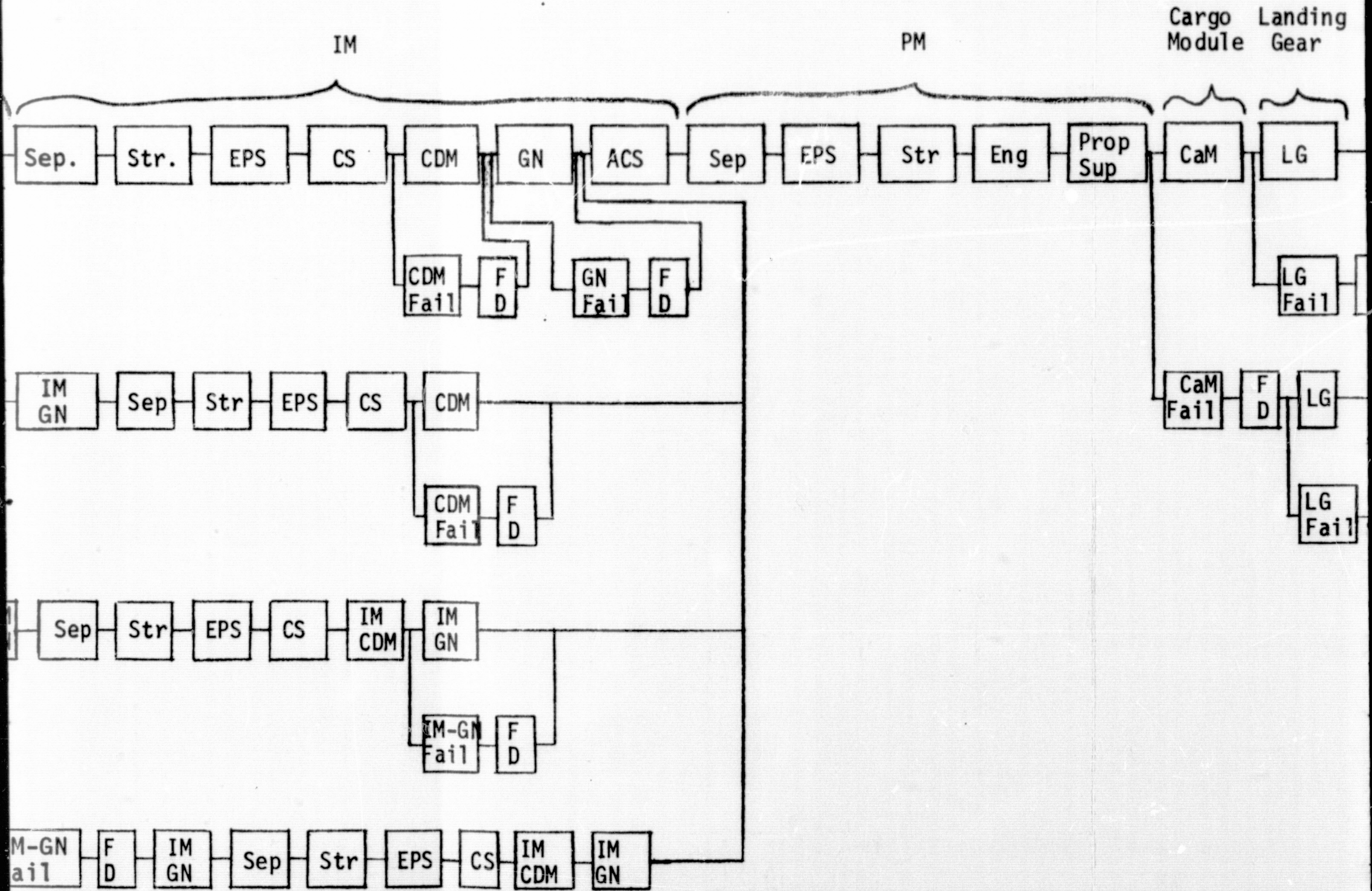


Figure B-6. Crew Safety Logic for Manned Lunar
NASA Mode A or B

t.)
for cargo loss). (Does not include Landing Gear Failure).
es not include Landing Gear Failure).

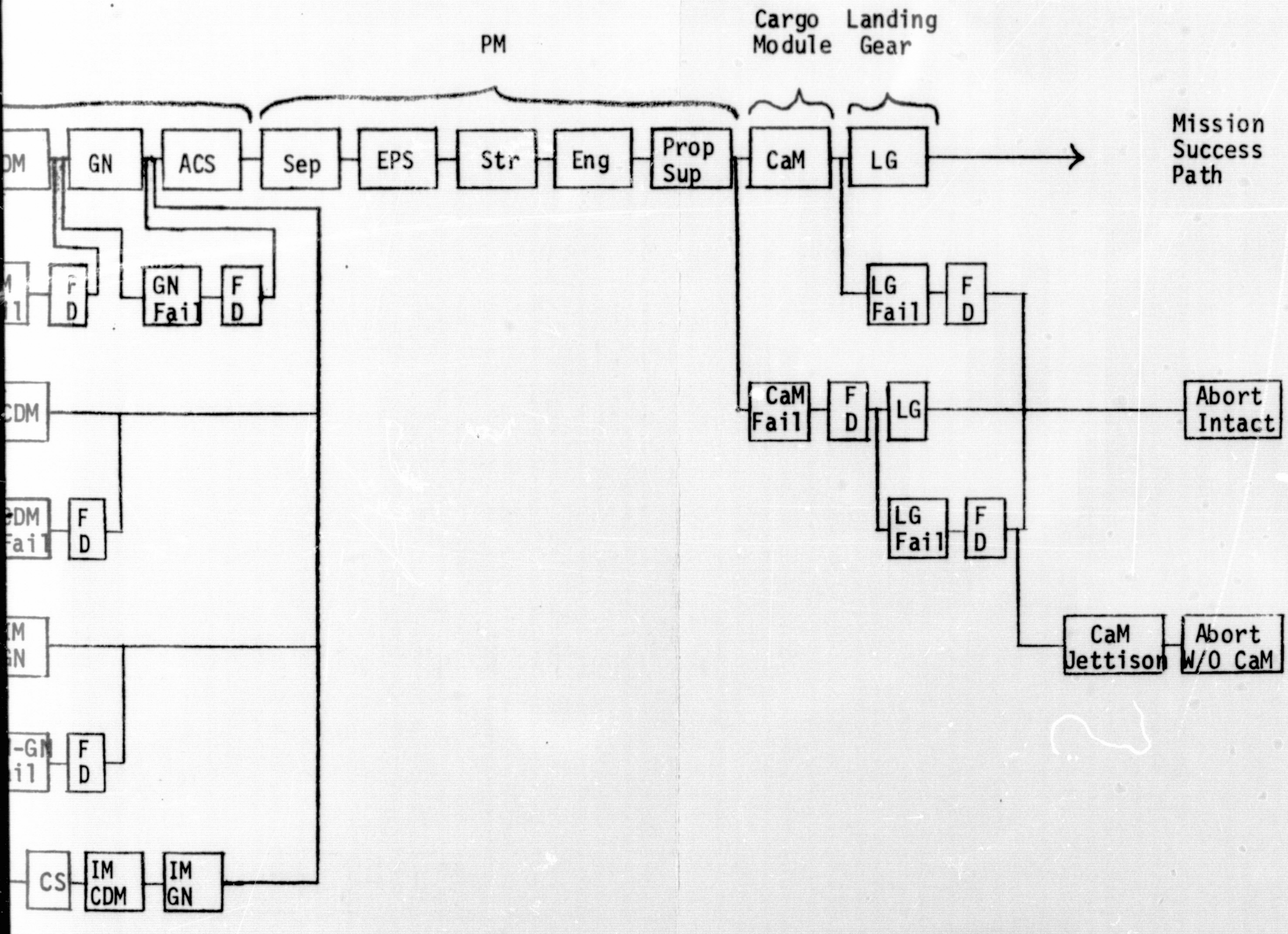
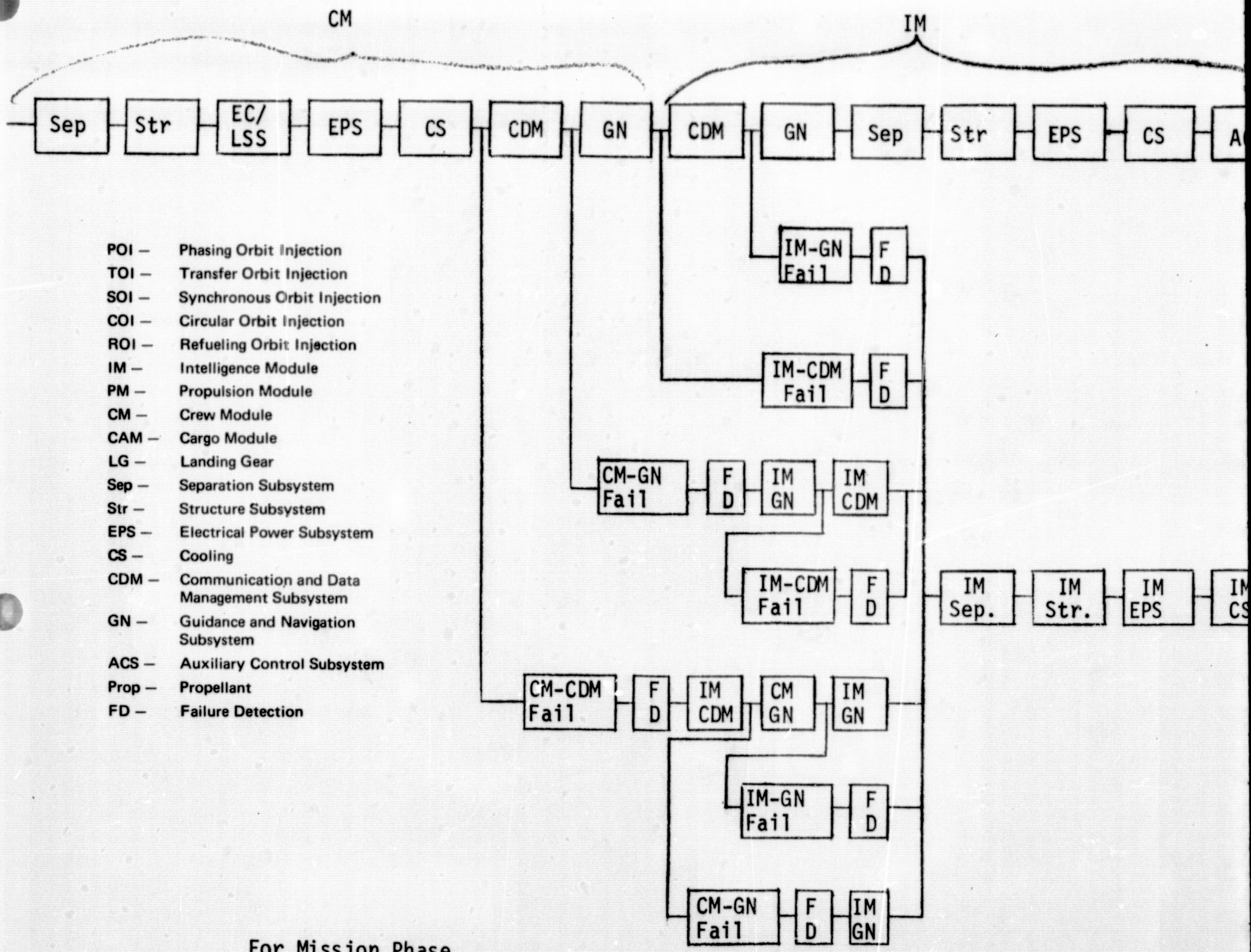


Figure B-6. Crew Safety Logic for Manned Lunar Landing Mission—NASA Mode A or B

FOLDOUT FRAME 3

UT FRAME 2



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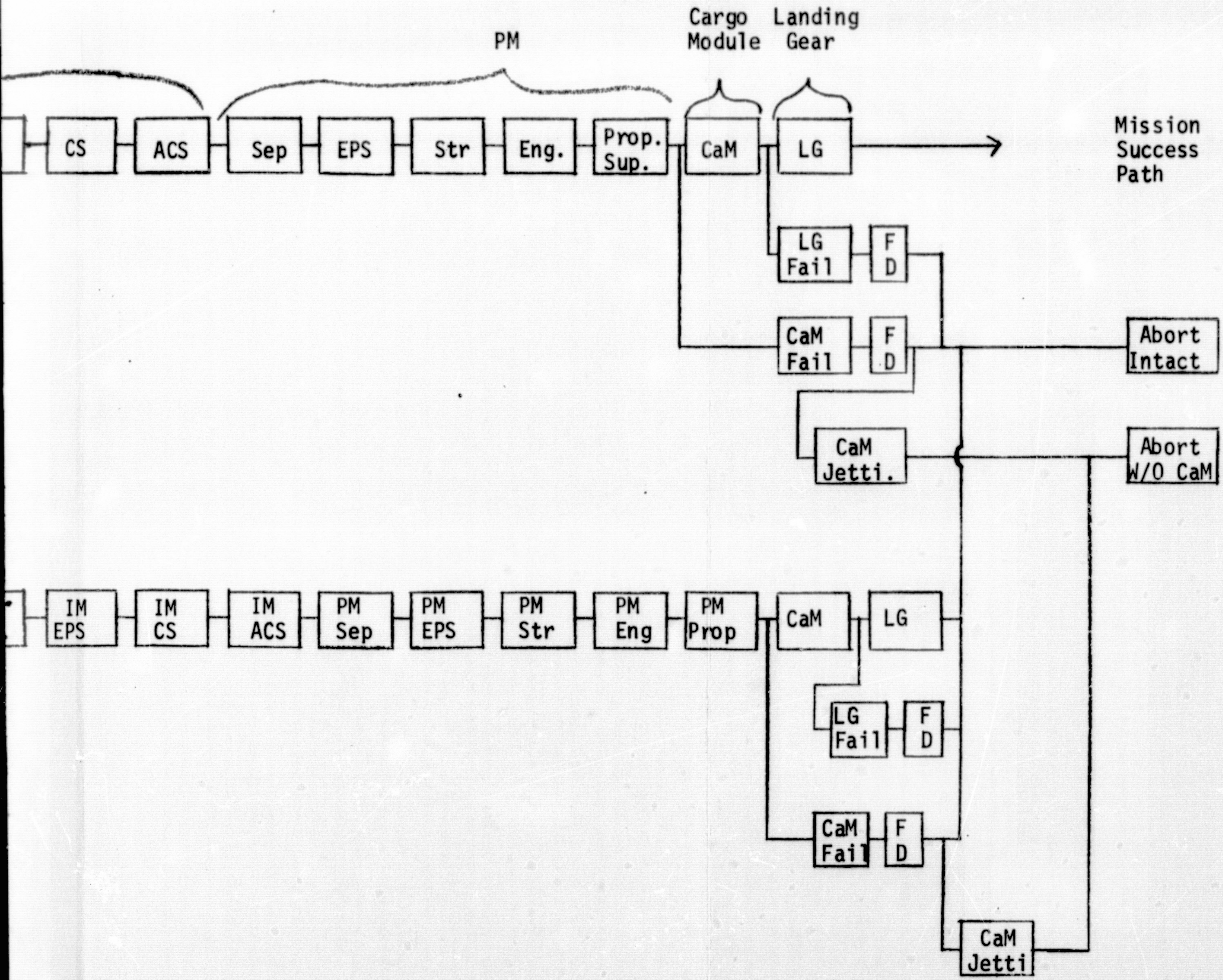
CDM — Communication and Data Management Subsystem
GN — Guidance and Navigation Subsystem
ACS — Auxiliary Control Subsystem
Prop — Propellant
FD — Failure Detection

For Mission Phase
Deorbit Coast

FOLDOU



FOLDOUT FRAME 2

FOLDOUT FRAME

Figure B-7. Crew Safety Logic for Manned Lunar Landing Mission—NASA Mode A or B

POI — Phasing Orbit Injection
TOI — Transfer Orbit Injection
SOI — Synchronous Orbit Injection
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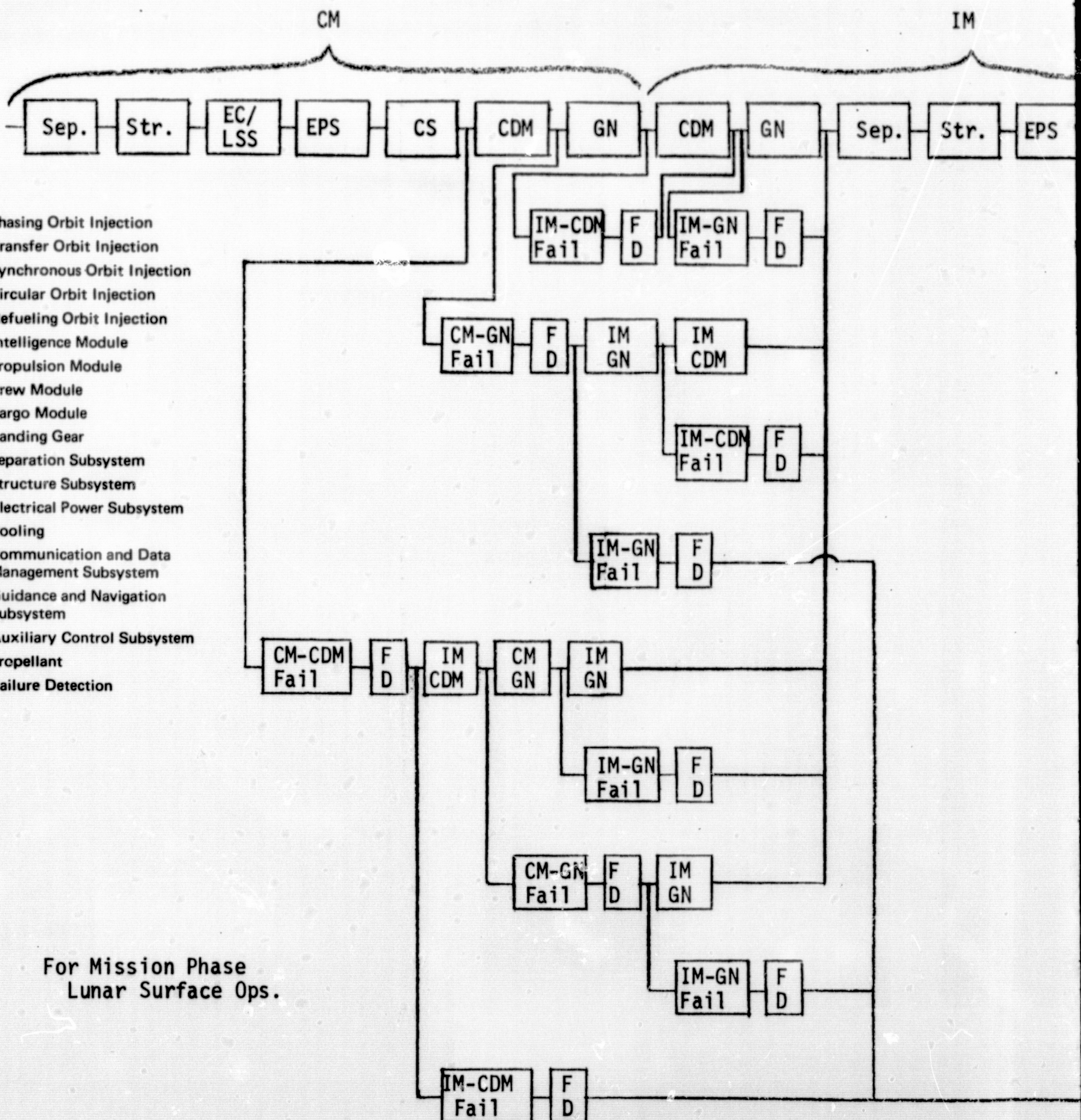
FD — Failure Detection



For Mission Phase
Lunar Surface Ops.

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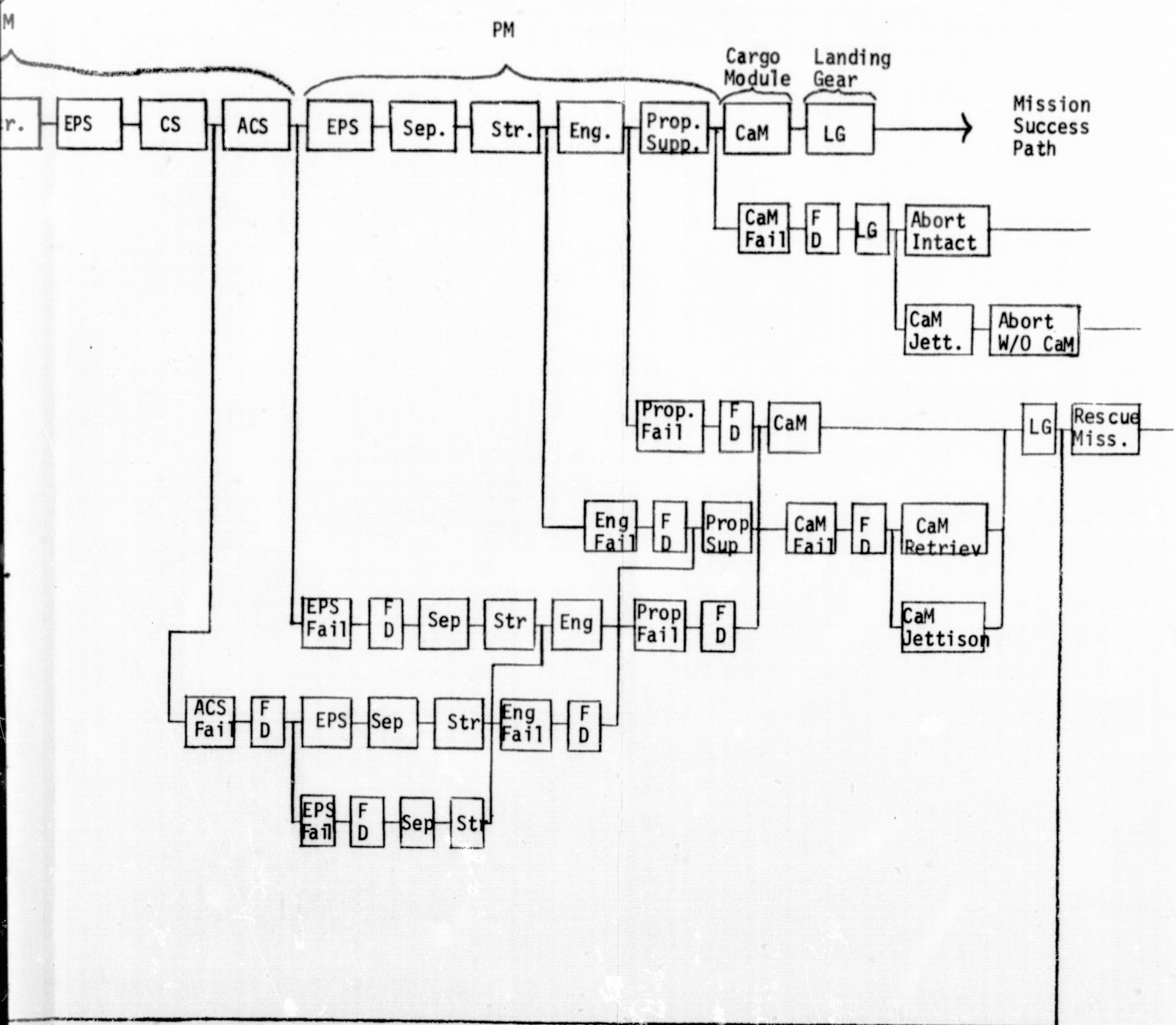
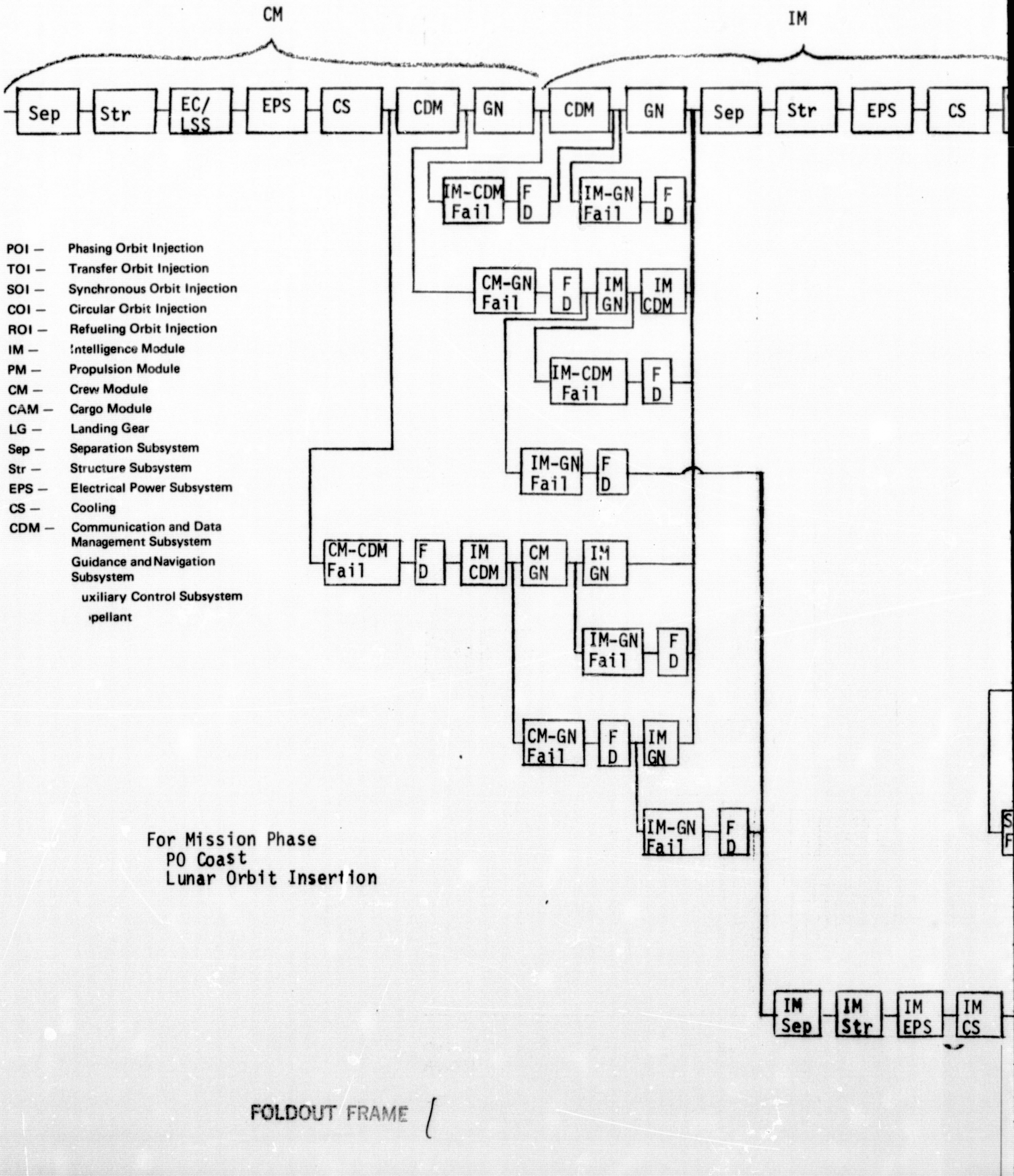


Figure B-8. Crew Safety Logic for Manned Lunar Landing Missions — NASA Mode A or B



POI — Phasing Orbit Injection
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COI — Circular Orbit Injection
ROI — Refueling Orbit Injection
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LG — Landing Gear
Sep — Separation Subsystem
Str — Structure Subsystem
EPS — Electrical Power Subsystem
CS — Cooling
CDM — Communication and Data Management Subsystem
Guidance and Navigation Subsystem
uxiliary Control Subsystem
ıpellant

For Mission Phase
PO Coast
Lunar Orbit Insertion

FOLDOUT FRAME 1

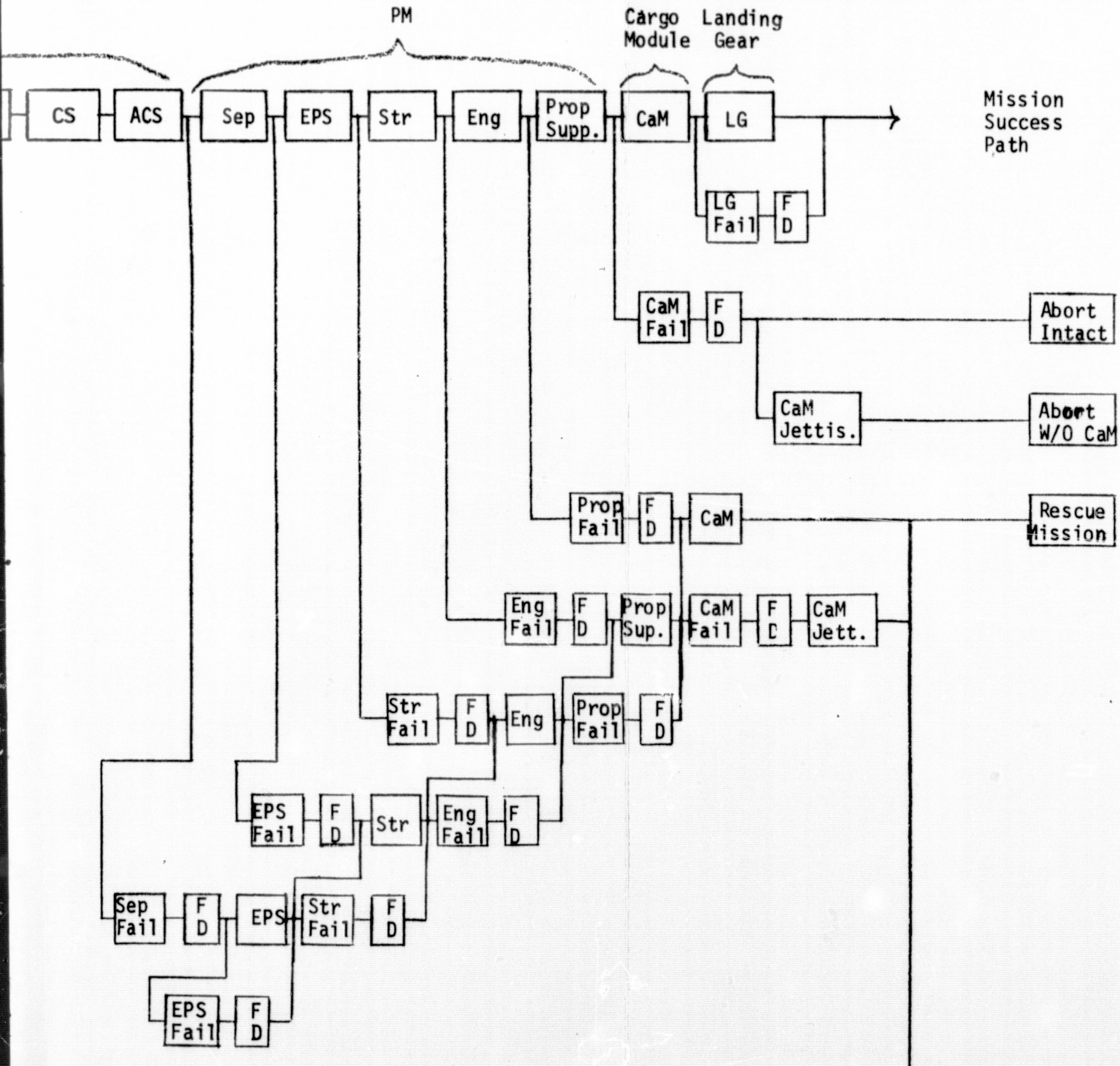


Figure B-9. Crew Safety Logic for Manned Lunar Landing Missions — NASA Mode A or B

FOLDOUT FRAME 2

Table B-1. Lunar Landing Abort Considerations

Abort Considerations	Explanation
Abort	Dependent on predictions of future incapability based on current assessment of mission hardware, crew, and environmental states Initiated by Crew Action only
Mission Features	Most critical times when rescue is extremely difficult Deorbit, deorbit coast, braking, and landing Liftoff to phasing orbit injection Rescue must be possible during lunar orbit coast, lunar surface operations, phasing orbit coast, and lunar orbit injection CM, IM, PM, and landing gear must provide emergency functional success for crew safety during critical mission phases Lunar surface requires least mission hardware operation
Hardware Features	Multiple, internal redundancy required in CM, IM, and PM for acceptable crew safety Landing gear must either have redundancy or large design margins Inadvertent separation must be avoided
Abort Criteria	Impossible to complete mission Mission hardware performance degradation below acceptable operational levels to the "safe" level Degradation of crew capabilities Environment excursion beyond planned levels Rescue when crew safety with mission hardware is less than crew safety with rescue hardware

Space Division
North American Rockwell

Table B-2. Sample Abort States

Sample Abort States	Description
Impossible mission completion	Leakage rate of pressurant, propellant, or crew oxygen sufficient to exhaust the supply before planned return time
	Rugged lurain
Mission hardware performance	Loss of visual contact with lunar surface
	Loss of cabin pressure
	Reduction of electrical power to emergency level
	Two PM engine failures (when the two remaining and RCS are sufficient for emergency landing after cargo jettison)
Degradation of crew capabilities	Illness of any crewman
	Any bleeding
	Accident on lunar surface
Environmental excursion beyond planned levels	Solar radiation
	Lunar surface temperature
	Meteoroid shower
Rescue	Degraded RCS capability so that docking risk is increased
	Emergency level of any critical function on lunar surface

HAZARD IDENTIFICATION

Every subsystem is required for safety success at some time during the projected tug missions. Therefore, failure of that subsystem function constitutes a hazard to the continued safety of hardware and crew. A hazard is a personnel and hardware state in which significant loss is possible. Selection of hazards for analysis emphasis and preventive design involves an understanding of the mission objectives as well as the potential hardware failure modes. At this time in the tug studies, experience on similar subsystems is substituted for a detailed hardware analysis. In general, each hazard is a system characteristic which could be slighted during the analysis of each subsystem. Several general hazards have been identified.

Loss of Attitude Control

This function is required for mission success during all mission phases and for safety during all flight phases. Recovery of the tug is considered unlikely if attitude control is lost in flight because of the resulting uncontrolled motion and the docking difficulty.

Damage to Adjacent Vehicles

Large losses are possible from collision with other space program vehicles; e.g., earth orbit space station, lunar orbit space station, lunar surface base, propellant facility, and expensive, autonomous experiment modules. Use of the tug as a shuttle among vehicles provides frequent exposure to severe damage. Subsystems directly involved are primarily in the IM; i.e., ACS (for translation as well as attitude), CDM (for translation of commands to the ACS), and GN (for trajectory determination). The PM initiates the high relative velocity maneuvers and must provide translation sufficient to avoid another vehicle when properly commanded. When available, the CM can also provide some redundancy for GN and command. The special case when the tug is confined in the shuttle is an interface problem during earth launch and return; e.g., mounting failure with resultant destructive movement or vapor explosion in the cargo bay.

Explosion

Explosion of mixed gases can be made very remote by proper design. The IM and PM use both hydrogen and oxygen but also are of relatively open construction. Both modules are steeped in the vacuum environment; i.e., leakage can be immediately dissipated. The causes of high pressure tank and line explosion are more difficult to control by design and must be checked by test. Design margin can be provided in non-frangible material and each tank and line can be proof pressure tested; but the possibility still exists of

a high pressure rupture. Physical isolation of redundant pressure containers is a further safeguard against explosions that cannot be allowed to occur.

Fire

Fire also falls in the category of a hazard which cannot be allowed in the tug. Material compatibility is the key to this problem and special attention to flammability limits of each detailed material in its environment is necessary. In addition, the starting means (high temperature source) must be eliminated by hermetic sealing or solid state circuitry for switches, routing of high tension wires in vacuum, and monitoring cabin atmosphere composition. The means of fire avoidance are available and must be applied in detailed design.

Inadvertent Separation.

This hazard exists because of specification of functions in separate modules. Concentration of the ACS in the IM only prohibits PM or CM from existing independently in orbit because of tumbling. Providing electrical power generation in the IM only eliminates independent PM or CM operation. Interconnectors must be designed with margin and internal redundancy to prevent inadvertent separation. (This problem is negligible with ground assembly of modules.)

External Environment Excursion.

Manned space tug missions are short enough for accommodation by reliable predictions of sun spot activity. Solar wind shelter for men at either end of the excursion are necessary; e.g., lunar orbit and earth orbit space stations. Equipment design can utilize margin for noise control and provide added reliability for other environments. Special shielding should be avoided as providing no functional improvement for most of the tug lifetime. Meteoroid frequency is characterized by model distributions with respect to size, density, and velocity. The essential feature of these models is their dependence on exposure time only, not location: a meteoroid is as apt to strike at one point in earth orbit as at any other. However, some consideration of time in orbit and module criticality provide clues to shielding distribution. For example, the criterion of crew safety would allow the following distribution.

1. IM - Provide shielding sufficient for full in-space time resistance because subsystem operations are required for recovery. (Relatively minor weight penalty.)

2. PM - No shielding required. Shielding would severely penalize performance. Retrieval is possible with the PM completely disabled. Inspection before mission usage would reveal damage.

3. CM - Provide shielding sufficient for manned occupation in space. This would occur on missions only - not on quiescent storage in space between missions. Retrieval is possible as long as the IM is operative. Inspection before mission usage would reveal damage.

4. Cargo Module - No shielding required. Short-time mission operations reduce the chance of puncture with cargo inside. Inspection following storage would reveal damage.

ALTERNATE OPERATIONAL PROCEDURES

At this time, several alternate operational procedures for reducing hazards can be suggested for further study.

1. Loss of attitude control - Provide cold gas jets on the PM directly from the hydrogen tank for IM-RCS backup, and provide gas-operated, body-mounted gyros in the PM for emergency stabilization control.

2. Damage to adjacent vehicles - Provide dual control for all docking maneuvers, primarily from the larger vehicle with alternate from the tug.

3. Explosion - Replenish the tug with propellants in orbit, outside of the shuttle cargo bay, or keep the cargo bay ventilated at all times.

4. Fire - No current alternatives.

5. Inadvertent separation - Assemble IM and other modules on the ground for specific missions. Maintain them in orbit in their original configurations. Interfacing provisions would be light in weight and fixed securely.

6. External environment excursion - Allow special tug orientation (e.g., PM toward Sun) during high solar activity, thus shielding IM and CM.

RESCUE MISSION CAPABILITY

Rescue capability improves safety and possibility of mission success. Safe return of the crew is mandatory, and return of disabled hardware is required by economy, even though the mission is interrupted. NASA objectives demand, at the minimum, safe operation. Mission success is

also increased by the capability of repair in order to attempt the mission again.

From a reliability and safety point of view, the space tug is an excellent rescue vehicle. The redundancies and durations required for normal operation are more than adequate for a rescue mission. Mission continuation is the prime abort criterion, since loss of the rescue mission means crew loss in any case. For a single sortie of rescue operations, the probability of success is much higher than for the cumulative, ten sorties required over the three years of expected space tug life.

In addition, the incidence of rescue will be rare, and the joint probability of a need for rescue and a failure to rescue is very small. A rescue mission should occur infrequently—once in a hundred manned flights. Failure during rescue could have a smaller incidence frequency—one in a thousand rescue attempts. Joint occurrence of both failures is only one in a hundred thousand. At 10 manned flights per year, failure to rescue when needed would then occur 10,000 years apart.

CREW SAFETY SENSITIVITY TO FLIGHT SCHEDULE AND SPACECRAFT TURNAROUND TIME

Most parts of the tug will have a long enough life that the overall failure model can be predicted as exponential; i.e., a relatively constant failure rate. This type of distribution has been experienced within the expected life of components common to many spacecraft. Even though any particular tug could fail on any flight, over the entire population of tugs the failures will probably be evenly distributed with time of flight. This is a result of good design; i.e., wear-out of any part has been compensated for by increasing marginal or preventive maintenance. The multitude of improbable failures are extremely difficult to fix analytically, expensive to design around, and impractical to find by test. Since absolute verification of the distribution is infinitely expensive, the true distribution is never known. From a practical standpoint, as long as the failure rate is lower than some reasonable value, the ease of constant failure rate utilization conservatively justifies its use. A continuously varying failure rate (i.e., high for early and later times with an intermediate minimum) is probably more accurate, but difficult to handle analytically and impossible to verify. Life-limited items will influence the later flights but not greatly because the design and maintenance will compensate.

Each in-space turnaround must include adequate time for failure detection. Since any failure can occur at any time, detection is a prime requirement for continued safety. Redundancy for safe return must be assured before every flight. Detailed and redundant instrumentation is the key to successful detection, and requires time for measurement data

reduction, and data analysis. Before a flight, any failure can be "corrected" (i.e., prevented from resulting in loss of crew) by individual fault repair or by return of the entire tug to earth for overhaul. Scheduling time for operational status determination between each flight is a necessity.

Extremely short turn-around times cause human factors problems. Crew performance under increasing stress becomes less accurate; more errors occur. This could result not only in undetected failures but also in induced failures. Turn-around times which allow routine maintenance with ground verification are required.

Utilization of the space tug for many flights early in the allowable 3-year life increases the overall probability of success by eliminating long, quiescent, in-space storage degradation. If every failure were detectable before a flight, the crew safety would be relatively independent of turn-around time, since the probability of failure during a single flight is relatively constant. However, measurement quantity would be extensive and impractical for some minute kinds of failure. Even though in-space storage is relatively reliable (no atmospheric contamination), other factors are continuously degrading the potential performance; e.g., solar and trapped particle radiation, internal contaminant corrosion, and outgassing into vacuum. Projected flight schedules could allow an average of approximately 3-1/2 months storage between each flight and 3 years in space is still a long time by present standards for manned vehicles.

The trend of success with schedule and turn-around time is shown in Figure B-10. Short time degradation is a function of crew competence in performing maintenance under stress. Long time degradation is a more gradual function of equipment failure after long, in-space storage. Some intermediate turn-around time is best for success.

The maximum probability of 10 successful flights depends on the detailed design of the space tug for rapid failure detection and long life design.

SINGLE POINTS OF FAILURE ANALYSIS

Module Criticality

Criticality matrices were developed for the criteria of mission success, safety, and crew safety for several typical missions. Each matrix identified the subsystem requirement in each mission phase against the applicable criterion. Retrieval of a disabled, unmanned tug and rescue of a disabled, manned tug were considered possible from any stable orbit and the lunar surface by another tug in low earth or lunar orbit.

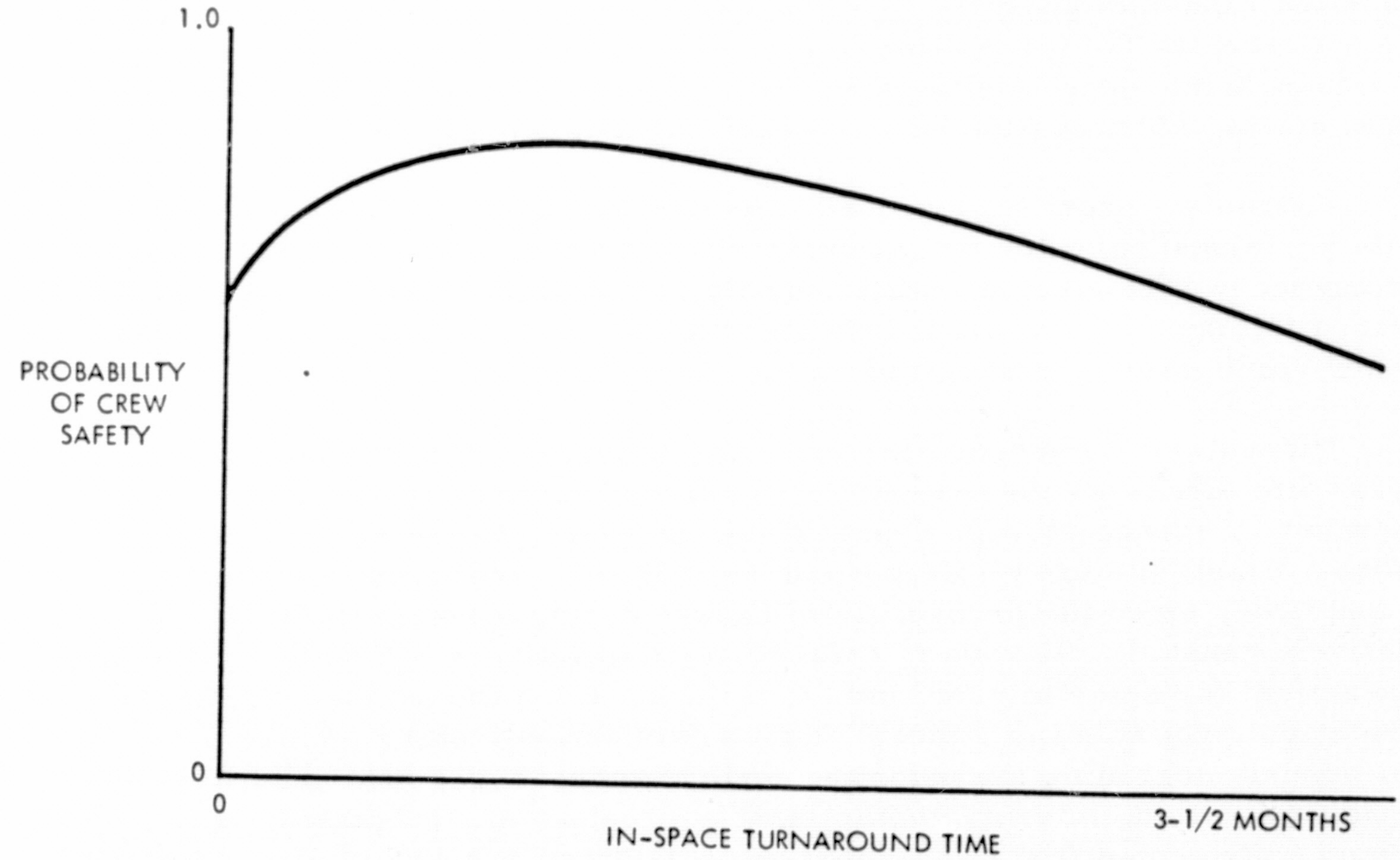


Figure B-10. Crew Safety Sensitivity to Turn-around Time.

It is logical and evident from Table B-3 that all modules are necessary for mission success; i.e., are mission critical. Different contributions are made to safety because of the additional modes for return without accomplishing the mission. Only the IM is required for all success criteria in all mission phases of all missions. The CM is next in criticality by requiring success in all manned missions. Propulsion need not be successful to return either crew or hardware from orbit or lunar surface because of the rescue and retrieval tug. Catastrophic failures must be considered and designed around so that the probability of occurrence is acceptably small. Lunar landing legs are critical on the lunar surface not only for landing but also for support during the lunar stay. Although landing implies a severe shock, the continued support during temperature extremes is not easy,. especially when reuse is a requirement. A properly designed cargo module (1) need never be crew safety critical since this module passively carries cargo without the need for active environmental control and (2) can be jettisoned in emergency.

Subsystem Criticality

Individual subsystems within each module have been evaluated for necessary functions contributing to mission success, safety (unmanned) and crew safety (manned). Tables B-3 through B-10 present subsystem

Table B-3. Module Criticality

Module	Unmanned Space Missions		Unmanned Lunar Landing		Manned Space Missions		Manned Lunar Landing	
	Mission Success	Hardware Safety	Mission Success	Hardware Safety	Mission Success	Crew Safety	Mission Success	Crew Safety
Intelligence module (IM)	X	X	X	X	X	X	X	X
Crew module (CM)	NA	NA	NA	NA	X	X	X	X
Propulsion module (PM)	X		X	X	X		X	X
Lunar landing legs	NA	NA	X	X	NA	NA	X	X
Cargo module	X	X	X	X	X		X	

Notes

IM required for all success criteria in all mission phases of all missions.

Selected subsystems within the CM required for safety.

PM required for safety on lunar landing, suborbital flight only.

Lunar landing legs are critical on lunar surface.

Cargo module is never crew safety critical.

Table B-4. Geosynchronous Mission—Unmanned for Single Stage Operations

Mission Phase	Intelligence Module							Propulsion Module					
	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply
MISSION SUCCESS (Accomplishing the payload ops and returning all hardware to low earth circulated orbit.)												
1. Phasing orbit injection	X[1]	X	X	X	X	X	X	X[3]	X	X[1]	X	X[2]
2. Phasing orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
3. Transfer orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
4. Transfer orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
5. Synchronous orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
5a. Payload operations (undefined)	X	X	X	X	X	X	X	X	X	X	X	X
6. Transfer orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
7. Transfer orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
8. Phasing orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
9. Phasing orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
10. Circular orbit injection	X	X	X	X	X	X	X	X	X	X	X	X

Table B-4. Geosynchronous Mission—Unmanned for Single Stage Operations (Cont)

Mission Phase	Intelligence Module							Propulsion Module				
	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply
	HARDWARE SAFETY						(Returning all hardware to low earth circular orbit.)					
1. Phasing orbit injection	X(4)	X	X	X(5)	X	(6)	X		X(4)			
2. Phasing orbit coast	X	X	X	X	X		X		X			
3. Transfer orbit injection	X	X	X	X	X		X		X			
4. Transfer orbit coast	X	X	X	X	X		X		X			
5. Synchronous orbit injection	X	X	X	X	X		X		X			
5a. Payload operations (undefined)	X	X	X	X	X		X		X			
6. Transfer orbit injection	X	X	X	X	X		X		X			
7. Transfer orbit coast	X	X	X	X	X		X		X			
8. Phasing orbit injection	X	X	X	X	X		X		X			
9. Phasing orbit coast	X	X	X	X	X		X		X			
10. Circular orbit injection	X	X	X	X	X		X		X			

(1) Inadvertent separation of either payload or PM from IM.
(2) Propellant supply performs the function of retaining propellants throughout the mission.
(3) Electrical distribution system only.
(4) Inadvertent separation of PM from IM.
(5) Cooling system required for EPS operation.
(6) Recovery vehicle can use its own GN.

Code:
- Sep - Separation subsystem
- Str - Structure subsystem
- EPS - Electrical power subsystem
- CS - Cooling subsystem
- ACS - Auxiliary control subsystem

Table B-5. Sun-Synchronous Mission—Unmanned for Mode 1
(Roundtrip From EOSS)

Mission Phase	Intelligence Module							Propulsion Module				
	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply
MISSION SUCCESS												
1. Phasing orbit injection (including undocking from EOSS)	X[1]	X	X	X	X	X	X	X[3]	X[1]	X	X	X[2]
2. Phasing orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
3. Transfer orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
4. Transfer orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
5. Synchronous orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
5a. Payload operations	X	X	X	X	X	X	X	X	X	X	X	X
6. Transfer orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
7. Transfer orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
8. Phasing orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
9. Phasing orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
10. Circular orbit injection (including docking to EOSS)	X	X	X	X	X	X	X	X	X	X	X	X
HARDWARE SAFETY												
1. Phasing orbit injection (including undocking from EOSS)	X[4]	X	X	X[5]	X	(6)	X		X[4]			
2. Phasing orbit coast	X	X	X	X	X		X		X			
3. Transfer orbit injection	X	X	X	X	X		X		X			

Space Division
North American Rockwell

Table B-5. Sun-Synchronous Mission—Unmanned for Mode 1
(Roundtrip from EOSS) (Cont)

Mission Phase	Intelligence Module							Propulsion Module				
	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply
HARDWARE SAFETY (CONT)												
4. Transfer orbit coast	X	X	X	X	X		X		X			
5. Synchronous orbit injection	X	X	X	X	X		X		X			
5a. Payload operations (undefined)	X	X	X	X	X		X		X			
6. Transfer orbit injection	X	X	X	X	X		X		X			
7. Transfer orbit coast	X	X	X	X	X		X		X			
8. Phasing orbit coast	X	X	X	X	X		X		X			
9. Phasing orbit coast	X	X	X	X	X		X		X			
10. Circular orbit injection (including docking to EOSS)	X	X	X	X	X		X		X			

(1) Inadvertent separation of either payload or PM from IM.
(2) Propellant supply performs the function of retaining propellants throughout the mission.
(3) Electrical distribution system only.
(4) Inadvertent separation of PM from IM.
(5) Cooling system required for EPS operation.
(6) Recovery vehicle can use its own GN.

Code:

Sep - Separation subsystem
Str - Structure subsystem
EPS - Electrical power subsystem
CS - Cooling subsystem
ACS - Auxiliary control subsystem

Table B-6. Sun-Synchronous Mission—Unmanned for Mode 2 (Roundtrip From EOSS With Refueling Stop at 100 n mi and 101.7 Degrees Inclination)

Mission Phase	Intelligence Module							Propulsion Module				
	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply
MISSION SUCCESS												
1. Phasing orbit injection (including undocking from EOSS)	X(1)	X	X	X	X	X	X	X	X(1)	X	X	X
2. Phasing orbit coast	X	X	X	X	X		X		X	X		
3. Transfer orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
4. Transfer orbit coast	X	X	X	X	X	X	X	X	X	X	X	
5. Synchronous orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
5a. Payload operations	X	X	X	X	X	X	X	X	X	X	X	
6. Transfer orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
7. Transfer orbit coast	X	X	X	X	X	X	X	X	X	X	X	
8. Refueling orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
9. Refuel	X	X	X	X	X	X	X	X	X	X	X	
10. Phasing orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
11. Phasing orbit coast	X	X	X	X	X	X	X	X	X	X	X	
12. Transfer orbit injection	X	X	X	X	X	X	X	X	X	X	X	X

Table B-6. Sun-Synchronous Mission—Unmanned for Mode 2 (Roundtrip From EOSS With Refueling Stop at 100 n mi and 101.7 Degrees Inclination) (Cont)

Mission Phase	Intelligence Module							Propulsion Module				
	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply
MISSION SUCCESS (CONT)												
13. Transfer orbit coast	X	X	X	X	X	X	X	X	X	X	X	
14. Circular orbit injection (including docking with EOSS)	X	X	X	X	X	X	X	X	X	X	X	X
HARDWARE SAFETY												
1. Phasing orbit injection (including undocking from EOSS)	X(1)	X	X	X(3)	X	(4)	X		X(1)			
2. Phasing orbit coast	X	X	X	X	X		X		X			
3. Transfer orbit injection	X	X	X	X	X		X		X			
4. Transfer orbit coast	X	X	X	X	X		X		X			
5. Synchronous orbit injection	X	X	X	X	X		X		X			
5a. Payload operations	X	X	X	X	X		X		X			
6. Transfer orbit injection	X	X	X	X	X		X		X			
7. Transfer orbit coast	X	X	X	X	X		X		X			
8. Refueling orbit injection	X	X	X	X	X		X		X			
9. Refuel	X	X	X	(2)	X		X		X			

Space Division
North American Rockwell

Table B-6. Sun-Synchronous Mission—Unmanned for Mode 2 (Roundtrip From EOSS With Refueling Stop at 100 n mi and 101.7 Degrees Inclination) (Cont)

Mission Phase	Intelligence Module							Propulsion Module				
	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply
HARDWARE SAFETY (CONT)												
10. Phasing orbit injection	X	X	X	X	X		X		X			
11. Phasing orbit coast	X	X	X	X	X		X		X			
12. Transfer orbit injection	X	X	X	X	X		X		X			
13. Transfer orbit coast	X	X	X	X	X		X		X			
14. Circular orbit injection (including docking with EOSS)	X	X	X	X	X		X		X			

(1) Inadvertent separation of PM from IM.

(2) In case of failure during refueling, return to ground by Shuttle.

(3) Cooling system must work for EPS operation.

(4) Use GN of recovery vehicle.

Code:

- Sep - Separation system
- Str - Structure subsystem
- EPS - Electrical power subsystem
- CS - Cooling subsystem
- ACS - Auxiliary control subsystem

Table B-7. Sun-Synchronous Mission—Unmanned for Mode 3
(Roundtrip to Shuttle in 101.7 Degrees Inclination)

Mission Phase	Intelligence Module							Propulsion Module				
	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply
MISSION SUCCESS												
1. Phasing orbit injection (including separation from Shuttle)	X	X	X	X	X	X	X	X	X	X	X	X
2. Phasing orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
3. Transfer orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
4. Transfer orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
5. Synchronous orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
5a. Payload operations	X	X	X	X	X	X	X	X	X	X	X	X
6. Transfer orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
7. Transfer orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
8. Phasing orbit injection	X	X	X	X	X	X	X	X	X	X	X	X
9. Phasing orbit coast	X	X	X	X	X	X	X	X	X	X	X	X
10. Circular orbit injection (including docking to Shuttle)	X	X	X	X	X	X	X	X	X	X	X	X
HARDWARE SAFETY												
1. Phasing orbit injection (including separation from Shuttle)	X	X	X	X	X		X		X			
2. Phasing orbit coast	X	X	X	X	X		X		X			
3. Transfer orbit injection	X	X	X	X	X		X		X			

Table B-7. Sun-Synchronous Mission—Unmanned for Mode 3 (Roundtrip to Shuttle in 101.7 Degrees Inclination) (Cont)

Mission Phase	Intelligence Module							Propulsion Module				
	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply
HARDWARE SAFETY (CONT)												
4. Transfer orbit coast	X	X	X	X	X		X		X			
5. Synchronous orbit injection	X	X	X	X	X		X		X			
5a. Payload operations	X	X	X	X	X		X		X			
6. Transfer orbit injection	X	X	X	X	X		X		X			
7. To coast	X	X	X	X	X		X		X			
8. Phasing orbit injection	X	X	X	X	X		X		X			
9. Phasing orbit coast	X	X	X	X	X		X		X			
10. Circular orbit injection (including docking to Shuttle)	X	X	X	X	X		X		X			

Code:

Sep - Separation subsystem

Str - Structure subsystem

EPS - Electrical power subsystem

CS - Cooling subsystem

ACS - Auxiliary control subsystem

Table B-8. Manned Lunar Landing Mission for NASA Modes A and B

Mission Phase	Crew Module							Intelligence Module							Propulsion Module					Landing Gear	Cargo Module
	Sep	Str	EC/LSS	EPS	CS	CDM	GN	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply		
MISSION SUCCESS																					
1. Lunar orbit coast (including separation from RNS)	X$^{(1)}$	X	X	X	X	X	X	X$^{(1)}$	X	X	X	X	X	X	X	X	X	X	X	X	X
2. De-orbit	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
3. De-orbit coast	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
4. Braking and landing	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	X
5. Lunar surface operations	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	X
6. Liftoff through phasing orbit injection	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	X
7. Phasing orbit coast	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X		X
8. Lunar orbit injection	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X		X
9. Lunar orbit coast (including docking to RNS)	X	X	X$^{(2)}$					X	X	X	X			X		X					X
CREW SAFETY																					
1. Lunar orbit coast (including separation from RNS)	X$^{(1)}$	X	X$^{(2)}$					X$^{(1)}$	X	X	X			X							
2. De-orbit	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X		
3. De-orbit coast	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X		
4. Braking and landing	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	
5. Lunar surface operations	X	X	X	X	X			X	X	X	X					X	X			X	
6. Liftoff through phasing orbit injection	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	
7. Phasing orbit coast	X	X	X	X	X			X	X	X	X			X							
8. Lunar orbit injection	X	X	X	X	X			X	X	X	X			X							
9. Lunar orbit coast (including docking to RNS)	X	X	X$^{(2)}$					X	X	X	X			X							

(1) Inadvertent separation from IM.
(2) Oxygen supply to the suit circuit is sufficient for emergency return to LOSS.

Code:

Sep - Separation subsystem
Str - Structure subsystem
EPS - Electrical power subsystem
CS - Cooling subsystem
ACS - Auxiliary control subsystem

Table B-9. Manned Lunar Landing Mission for NASA Mode C

Mission Phase	Crew Module							Intelligence Module							Propulsion Module					Landing Gear	Cargo Module
	Sep	Str	EC/ LSS	EPS	CS	CDM	GN	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply		
MISSION SUCCESS																					
Crew Transport																					
1. Lunar orbit coast	X	X	X	X	X			X	X	X	X	X		X		X	X		X		NA
2. De-orbit	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		NA
3. De-orbit coast	X	X	X	X	X			X	X	X	X	X		X		X	X		X		NA
4. Braking and landing	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	NA
5. Lunar surface operations	X	X	X	X	X			X	X	X	X					X	X		X	X	NA
6. Liftoff through phasing orbit injection	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	NA
7. Phasing orbit coast	X	X	X	X	X			X	X	X	X			X		X	X		X		NA
8. Lunar orbit injection	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X		NA
9. Lunar orbit coast	X	X						X	X	X	X			X		X	X				NA
Cargo Transport																					
1. Lunar orbit coast								X	X	X	X	X		X		X	X				X
2. De-orbit								X	X	X	X	X	X	X	X	X	X	X	X		X
3. De-orbit coast								X	X	X	X	X		X		X	X				X
4. Braking and landing								X	X	X	X	X	X	X	X	X	X	X	X	X	X
5. Lunar surface operations	Not applicable							X	X	X	X	X				X	X			X	X
6. Liftoff through phasing orbit injection								X	X	X	X	X	X	X	X	X	X	X	X	X	X
7. Phasing orbit coast								X	X	X	X	X		X		X	X				X
8. Lunar orbit injection								X	X	X	X	X	X	X	X	X	X	X	X		X
9. Lunar orbit coast								X	X	X	X	X		X		X	X				X
CREW SAFETY																					
Crew Transport																					
1. Lunar orbit coast	X	X	X	X	X			X	X	X	X			X							
2. De-orbit	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X		
3. De-orbit coast	X	X	X	X	X			X	X	X	X			X		X	X		X		
4. Braking and landing	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	
5. Lunar surface operations	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	
6. Liftoff through phasing orbit injection	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	
7. Phasing orbit coast	X	X	X	X	X			X	X	X	X			X							
8. Lunar orbit injection	X	X	X	X	X			X	X	X	X			X							
9. Lunar orbit coast	X	X	X					X	X	X	X			X							
Cargo Transport																					
1. Lunar orbit coast																					
2. De-orbit																					
3. De-orbit coast																					
4. Breaking and landing																					
5. Lunar surface operations	Not applicable																				
6. Liftoff through phasing orbit injection																					
7. Phasing orbit coast																					
8. Lunar orbit injection																					
9. Lunar orbit coast																					

Code:

- Sep - Separation subsystem
- Str - Structure subsystem
- EPS - Electrical power subsystem
- CS - Cooling subsystem
- ACS - Auxiliary control subsystem

Table B-10. Manned Lunar Landing Mission for NASA Mode D

Mission Phase	Crew Module							Intelligence Module							Propulsion Module					Landing Gear	Cargo Module
	Sep	Str	EC/ LSS	EPS	CS	CDM	GN	Sep	Str	EPS	CS	CDM	GN	ACS.	EPS	Sep	Str	Eng	Prop. Supply		
MISSION SUCCESS																					
Crew Transport																					
1. Lunar orbit coast	X	X	X	X	X			X	X	X	X	X		X		X	X		X		NA
2. De-orbit	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		NA
3. De-orbit coast	X	X	X	X	X			X	X	X	X	X		X		X	X		X		NA
4. Braking and landing	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	NA
5. Lunar surface operations	X	X	X	X	X			X	X	X	X					X	X		X	X	NA
6. Liftoff through phasing orbit injection	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	NA
7. Phasing orbit coast	X	X	X	X	X			X	X	X	X			X		X	X		X		NA
8. Lunar orbit injection	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X		NA
9. Lunar orbit coast	X	X	X	X	X			X	X	X	X			X		X					NA
Cargo Transport																					
1. Lunar orbit coast	Not applicable							X	X	X	X	X		X		X	X		X		X
2. De-orbit								X	X	X	X	X	X	X	X	X	X	X	X		X
3. De-orbit coast								X	X	X	X	X		X		X	X		X		X
4. Braking and landing								X	X	X	X	X	X	X	X	X	X	X	X	X	X
5. Lunar surface operations								X	X	X	X	X				X	X		X	X	X
6. Liftoff through phasing orbit injection								X	X	X	X	X	X	X	X	X	X	X	X	X	X
7. Phasing orbit coast								X	X	X	X	X		X		X	X		X		X
8. Lunar orbit injection								X	X	X	X	X	X	X	X	X	X	X	X		X
9. Lunar orbit coast								X	X	X	X	X		X		X	X		X		X
Fuel Transport																					
1. Lunar orbit coast	Not applicable							X	X	X	X	X		X		X	X		X		NA
2. De-orbit								X	X	X	X	X	X	X	X	X	X	X	X		NA
3. De-orbit coast								X	X	X	X	X		X		X	X		X		NA
4. Braking and landing								X	X	X	X	X	X	X	X	X	X	X	X	X	NA
5. Lunar surface operations (including fuel transfer)								X	X	X	X	X				X	X		X	X	NA
6. Liftoff through phasing orbit injection								X	X	X	X	X	X	X	X	X	X	X	X	X	NA
7. Phasing orbit coast								X	X	X	X	X		X		X	X		X		NA
8. Lunar orbit injection								X	X	X	X	X	X	X	X	X	X	X	X	X	NA
9. Lunar orbit coast								X	X	X	X	X		X		X	X		X		NA
Fuel and Cargo Transport																					
1. Lunar orbit coast	Not applicable							X	X	X	X	X		X		X	X		X		
2. De-orbit								X	X	X	X	X	X	X	X	X	X	X	X	X	
3. De-orbit coast								X	X	X	X	X		X		X	X		X		

Table B-10. Manned Lunar Landing Mission for NASA Mode D (Cont)

Mission Phase	Crew Module							Intelligence Module							Propulsion Module					Landing Gear	Cargo Module
	Sep	Str	EC/ LSS	EPS	CS	CDM	GN	Sep	Str	EPS	CS	CDM	GN	ACS	EPS	Sep	Str	Eng	Prop. Supply		
MISSION SUCCESS (CONT)																					
<u>Fuel and Cargo Transport</u> (Cont)																					
4. Braking and landing	Not applicable							X	X	X	X	X	X	X	X	X	X	X	X	X	
5. Lunar surface operations (including fuel and cargo transfer)								X	X	X	X	X				X	X		X	X	X
6. Liftoff through phasing orbit injection								X	X	X	X	X	X	X	X	X	X	X	X	X	X
7. Phasing orbit coast								X	X	X	X	X		X		X	X		X		X
8. Lunar orbit injection								X	X	X	X	X	X	X	X	X	X	X	X		X
9. Lunar orbit coast								X	X	X	X	X		X		X	X		X		X
CREW SAFETY																					
<u>Crew Transport</u>																					
1. Lunar orbit coast	X	X	X	X	X			X	X	X	X			X							
2. De-orbit	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X		
3. De-orbit coast	X	X	X	X	X			X	X	X	X			X		X	X		X		
4. Braking and landing	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	
5. Lunar surface operations	X	X	X	X	X			X	X	X	X					X	X		X	X	
6. Liftoff through phasing orbit injection	X	X	X	X	X			X	X	X	X			X	X	X	X	X	X	X	
7. Phasing orbit coast	X	X	X	X	X			X	X	X	X			X							
8. Lunar orbit injection	X	X	X	X	X			X	X	X	X			X							
9. Lunar orbit coast	X	X						X	X	X	X			X							
<u>Cargo Transport</u>																					
1. Lunar orbit coast	Not applicable																				
2. De-orbit																					
3. De-orbit coast																					
4. Braking and landing																					
5. Lunar surface operations (including refueling)																					
6. Liftoff through phasing orbit injection																					
7. Phasing orbit coast																					
8. Lunar orbit injection																					
9. Lunar orbit coast																					
<u>Fuel Transport</u>																					
Fuel and cargo transport	Not applicable																				

<u>Code:</u>
- Sep - Separation subsystem
- Str - Structure subsystem
- EPS - Electrical power subsystem
- CS - Cooling subsystem
- ACS - Auxiliary control subsystem

criticalities for each mission phase in unmanned geosynchronous; unmanned sun-synchronous Modes 1, 2, and 3; and manned lunar landing NASA Modes A, B, C, and D.

At this level of assembly, the subsystem functions are all required for mission success during some portion of the mission. All necessary functions have been included in subsystem operation according to the mission performance requirements. This is characteristic of a minimum system design and is an excellent starting place for redundancy considerations. Increased durability can be added as required by cost and criticality trade-offs. Since one of the success criteria is a low number of single points of failure, each subsystem needs further analysis to enable recommendations of complete subsystem or major element redundancy.

Safety requires return of tug and payload to a haven under normal or emergency conditions. All normal operating modes for mission success constitute safe conditions whether or not failures have occurred. Subsystem failures which leave the tug in a stable orbit with attitude control are still safe because rescue of crew or retrieval of equipment should always be possible by a stand-by tug in low earth or lunar orbit. However, the primary tug return mode is with internal equipment, whenever possible; e. g., loss of fine attitude control necessary for docking need not deter the tug from translating into the haven vicinity before being retrieved.

As indicated by module criticality, most subsystems in the IM are safety critical for unmanned operation. The separation subsystem must not separate inadvertently because the modules attached to the IM have no attitude control; i. e., no means of remaining stable for possible re-attachment. Structural failure in general could result in loss of any or all functions. Loss of electrical power would stop all active functions. Retention of adequate cooling is necessary to retain electrical power.

The communications and data management subsystem (CDM) contains the computers for sequential commanding of active functions. Attitude control can only be accomplished by the auxiliary control subsystem. The only IM functions not safety critical for unmanned operation are contained in the guidance and navigation subsystem (GN) which are duplicated by a retrieval tug. Two redundancies are achieved by the addition of crew: CDM and GN. The crew can take over any computer and guidance function.

The PM fares much better for number of safety critical subsystems; i. e., the separation subsystem is the only one for stable orbits. Inadvertent separation (if not ground-assembled) would leave the PM stranded. All other subsystems can fail and the assembly would remain in orbit under controlled attitude until retrieval. For suborbital flight, such as lunar landing and liftoff, propulsion becomes critical since retrieval is considered

impractical within the time constraints. Proper operation of the electrical power, structure, engine, attitude control, and propellant supply subsystems is required to prevent catastrophic lunar surface impact .

Propulsion subsystem success is always required for mission success but seldom for crew safety. Transport of the IM, CM, and cargo modules with high impulse in a short time requires a large reservoir of propellants used at reasonable thrust. From the first mission phase (phasing orbit injection) to the last (circular orbit injection,) propulsion must provide the energy for payload insertion in a desired orbit and crew return to a long-duration haven.

Crew safety is affected by propulsion subsystem failure only when crew rescue or hardware recovery is not possible. Rescue or recovery is possible from all orbital trajectories and from the lunar surface by a standby tug in earth or lunar orbit. Critical propulsion failures occur during lunar suborbital trajectories; i.e., deorbit from lunar orbit, deorbit coast, braking and landing, and liftoff through phasing orbit injection. Loss of propulsion during these critical phases would result in a crash landing on the lunar surface since rescue time is so limited.

Redundancy should be applied to the propulsion subsystem only where necessary to provide reasonable mission success and crew safety, i.e., "selected multiple redundancy" (RFP). Multiple full size propellant tanks are not feasible, but an emergency tank for lunar landing may be just the addition necessary to minimize crew loss due to single failures. Table B-11 illustrates an increasing reliability sequence. Pressurization can multiply redundancy easily because of the many small components and assemblies involved, e.g., quad check valves.

Engine redundancy is decided by a combination of mission requirements and development expense. Rocket engines are notorious for catastrophic failure since they contain and control a mixture of extreme environments. Therefore, development of reasonable reliability has been expensive. Mission characteristics define criticality of the propulsion function, e.g., non-critical in orbital flight and critical during lunar sub-orbital flight. Matching reasonable reliability with mission requirements dictates multiple engines unless extensive internal redundancy and large margins are allowed.

Tug propulsion reliability is satisfied by a four engine configuration. The mission can be successfully accomplished on any three, and emergency landing and return can be accomplished on any two. Successful performance of any one engine would provide a large translation capability in orbit and may even supplement IM-RCS translation capabilities. For lunar landing, the redundancy is Fail Operational-Fail Safe; and for all other missions, the redundancy is Fail Operation-Fail Operational-Fail Safe. Failure

Table B-11. Propellant Tank Redundancy

Redundancy Criterion	Hardware Definition
Fail Critical	1 oxidizer tank 1 hydrogen tank
Fail Safe	1 oxidizer tank with separate sump for emergency 1 hydrogen tank with separate sump for emergency
Fail Operational-Fail Safe	1 oxidizer tank with separate sump and an 8 percent increase in strength 1 hydrogen tank with separate sump and an 8 percent increase in strength
Fail Operational-Fail Operational-Fail Safe	1 oxidizer tank with separate sump + 15 percent increase in strength 1 hydrogen tank with separate sump + 15 percent increase in strength

detection and shutdown means are required. Single engine failure during the critical lunar landing phases would not compromise mission success because the failed engine and the opposite engine would be shut down. Double engine failure (engines opposite each other) would not compromise crew safety, since the failed engines would be shut down and the crew boosted to orbit or landed safely, depending on descent or ascent trajectory location. The capability of cargo jettison may be required to provide for safe operation on any two engines.

The critical phases during the lunar landing mission describe the PM configuration need for redundancy. Other missions and mission phases need not have that redundancy because of the good possibility of rescue from orbit.

Manned missions require a CM with several safety critical subsystems. Five CM subsystems are safety critical: separation, structure, environmental control and life support, electrical power, and cooling. Inadvertent separation leaves the CM stranded without attitude control. Any structural break could damage necessary, adjacent functional equipment. Obviously, life support is required. Electrical power is necessary to run the life support equipment, and cooling is required for electrical power. Manned operation of the CDM and GN in the CM is redundant with those similar functions in the IM.

Single Point of Failure Modes Identification

Since reliability theory cannot determine the exact times of failures, any failure can occur on any mission. The stochastic model generally used predicts failures over the entire population (hopefully large) of vehicles. Modelling a probability distribution over mission times predicts a range of probabilities with only zero time representing complete success.

Special treatment of single points of failure is necessary in order to limit the risks. Identification provides the visibility to conduct analyses, tests, and preflight checks for each design configuration, each manufactured item, and each mission. Such attention provides corrective emphasis and tends to reduce the immediate failure probabilities by verifying design and manufacturing excellence.

Single points of failure are identified by their effects. Many individual failures may not in themselves cause appreciable loss because of alternate operating modes or redundant equipments. These are single failures; but since their effects are not serious, they are not identified as single points of failure. More detailed lists of all failure modes may be made during subsequent contract phases. Single failures which could result in serious losses receive specific recognition. The definition in MHB5300.4 (1A), Reliability Program Provisions for Aeronautical and Space System Contractors (April 1970) has been followed:

> "Single Failure Point. A single element of hardware, the failure of which would result in loss of objectives, hardware, or crew, as defined for the specific application and/or project for which the single point failure analysis is performed."

"Objectives" is associated with mission success, "hardware" is limited to major modules, and "crew" is identified with crew safety. Further generalization has been made to summarize the modes of failure, as required by S. O. W. 4.2.4.

Identification of single points of failure has been based on work already accomplished on this contract and experience from prior failure mode analyses. The subsystem criticality tables of the previous section narrow the considerations from modules to subsystems at the expense of identifying mission phases. Table B-12 is an illustration of the subsystem single points of failure for the lunar landing mission. Major component redundancies, as defined in Section 4.2.8.1, were considered, since the number of redundancies influences the single failure effects. Next, the effects of failures in non-redundant and singly redundant major components were determined, Table B-13. Figures B-11 and B-12 illustrate the relative effects of subsystem failures and major component single points of failure for typical single stage geosynchronous missions. All subsystem functions are required for mission success, fewer are required for safety, and scattered subsystems include single points of failure.

A summary of the failure modes is presented in Table B-14. Explosions were presumed to result in serious loss from shrapnel impingement on critical hardware. No special shielding was considered. A support structure break also implied the subsequent failure of adjacent critical components for the CM and IM with the primary landing leg failure causing serious loss. Meteoroid penetration could occur anywhere on the vehicle with the larger surfaces more likely to be damaged than the smaller. Penetration into the cabin could result in immediate crew loss. Propellant loss could result in either insufficient impulse to complete a mission or lunar landing trajectory, or uncontrolled motion when a directed leak (small thrust vector) came from a large propellant tank. Biological contamination is especially insidious because of immediate crew effects. Cabin rupture could result in atmosphere loss if the break were large.

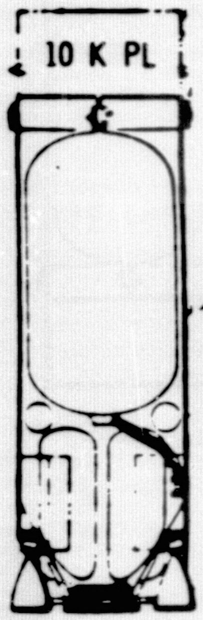
Single Point of Failure Modes Justification

Justification of potential failures may be possible before the fact but very seldom after the fact. If the justified failure never occurs, it is a triumph of analysis. If it occurs, then the identification of an expedient preventive is necessary. Analysis must be assisted by test to identify failure modes and to evaluate their relative magnitudes. Specific justification can be made only when all other alternatives have been considered, compared, and rejected.

Table B-12. Single Points of Failure for the Manned Lunar Landing Mission - NASA Modes A, B, C, and D

1. All failures in the IM which result in shrapnel (e.g. pressurized component burst) except during Phase 5.
2. All failures in the CM which result in shrapnel.
3. All failures in the PM which result in shrapnel during Phase 5 only.
4. All meteroid penetrations of CM.
5. All meteroid penetration of IM.
6. All meteroid penetrations of PM during Phases 2, 3, 4, and 6.
7. All tank failures which result in uncontrolled motion.
8. Inadvertent separation of CM, IM, or PM (during Phases 2, 3, 4, 5, and 6).
9. Structural failure of CM, IM, or PM (during Phases 2, 3, 4, 5, and 6).
10. Failure of EC/LSS in CM, except during Phase 9.
11. Failure of IM-ACS except during Phase 5.
12. PM-EPS failure during Phases 2, 4, and 6.
13. PM-Engine failure during Phases 2, 4, and 6.
14. PM-Propellant Supply failure during Phases 2, 3, 4, 5, and 6.
15. Landing gear failure during Phases 4, 5, and 6.
16. Failure of CM-EPS and -CS for all Phases except 9.
17. Failure of CM-CDM and -GM during the critical Phases - 4 and 6.
18. Failure of IM-EPS, -CS, and -CDM for all phases.
19. IM-GN failure during Phases 2, 4, and 6.

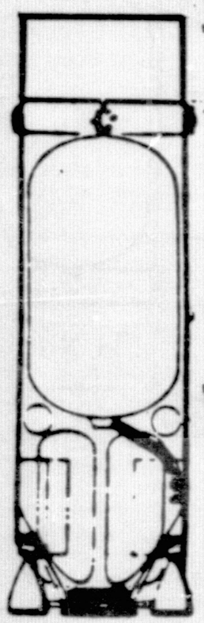
Concept 1



Module	Subsystem	Function Req d		Single Points of Failure	
		Mission Success	Hardware Safety	Mission Objectives Loss	Hardware Loss
Payload Module	- Structure	X	X	X	X
	- Separation	X	X		
Intelligence Module	- Structure	X	X	X	X
	- Separation	X	X		
	- Electrical Power	X	X	X	X
	- Cooling	X			
	- Communications & Data Mgmt	X	X		
	- Auxiliary Control	X	X	X	X
	- Guidance, Navigation & Control	X	X		
Propulsion Module	- Structure	X			
	- Separation	X			
	- Electrical Power	X		X	X
	- Cooling	X			
	- Engine	X		X	X
	- Propellant Supply	X		X	X

Figure B-11. Subsystem Criticality—Unmanned Earth Orbital Mission

Concept 1



Module	Subsystem	Function Req'd: Mission Success	Function Req'd: Crew Safety	Single Points of Failure: Mission Objectives Loss	Single Points of Failure: Hardware Loss
Crew Module	- Structure	X	X	X	X
	- Separation	X	X		
	- Electrical Power	X	X		
	- Cooling	X	X	X	
	- Communications & Data Mgmt	X	X		
	- Guidance, Nav. & Control	X	X		
	- EC/LSS	X	X	X	X
Intelligence Module	- Structure	X	X	X	X
	- Separation	X	X		
	- Electrical Power	X	X	X	X
	- Cooling	X	X		
	- Communications & Data Mgmt	X	X		
	- Auxiliary Control	X	X	X	X
	- Guidance, Nav. & Control	X	X	X	
Propulsion Module	- Structure	X			
	- Separation	X			
	- Electrical Power	X		X	X
	- Cooling	X			
	- Engine	X		X	X
	- Propellant Supply	X		X	X

Figure B-12. Subsystem Criticality—Manned Earth Orbital Mission

Table B-13. Failure Effects

Subsystem	Failure Description	Mission Objectives Loss	Crew Loss	Hardware Loss
Cooling	CM space mission fixed equipment - coolant leak	X		
Structure	CM - cabin wall failure - atmosphere leak	X	X	
	IM - member failure causing subsequent critical component failure	X	X	X
	Payload module - member failure causing subsequent payload failure	X		X
EC/LSS	CM space mission fixed equipment			
	Improper sterilization	X	X	
	O2 supply during EVA	X		
	CM lunar landing additional fixed equipment			
	Release of biological agents	X	X	
	Improper sterilization	X	X	
	O2 supply during EVA	X		
	CM expendables - improper sterilization	X	X	
Guidance, Navigation, and Control	IM basic equipment - navigation sensor base shift	X		
Active Thermal Control	CM equipment - coolant leak	X		
Auxiliary Control	IM equipment - explosion	X	X	X
	PM equipment - explosion	X	X	X
Electrical Power	IM basic equipment - explosion	X	X	X
	PM equipment - explosion	X	X	X
Main Propulsion	Propellant tanks			
	Explosion	X	X	X
	Directed leak	X	X	X
	Engines			
	Explosion	X	X	X
Lunar Landing Legs	Leg structure - material yield	X	X	X

Table B-14. Single Points of Failure Modes

Failure Mode	Description	Example
	Hardware and Crew Safety	
Explosions	Loss of critical functions by shrapnel	ACS engine
	Uncontrolled motion	H_2 tank leak
Support Structure Break	Lunar Landing Leg Fold	Material yeild
Meteroid Penetration	Loss of Critical Functions	ACS
Propellant Loss	Leak during lunar impact trajectory	Propellant tank leak
	Stoppage during lunar impact trajectory	Propellant tank outlet
	Added for Crew Safety Only	
Biological Contamination	Improper Sterilization of Food	Crew ingestion
	Improper Sterilization of Water	Crew ingestion
	Improper Sterilization of Waste	Crew ingestion
	Experiment Agent Release	Crew ingestion
Cabin Rupture	Loss of Crew Atmosphere	CM cabin leak

Enough detail must be generated for each alternative to estimate the many significant factors involved; e.g. risk, cost, and schedule of accomplishment of objectives. The present evaluation has considered these factors in general, since details are not yet available. Complete elimination of risk is deemed impractical as requiring, at a minimum, excessive weight, development cost, and schedule extension. Tradeoff analyses among these factors in subsequent contract phases should resolve the specific levels of risk and effort.

Justification of single point of failure modes has been made based on major mission requirements. The realization that a minimum number of such modes is more desirable has forced specification of many redundancies which may be reconsidered later in favor of more development or more risk. The remaining modes can be justified by four rationalizations:

1. Weight Reduction - redundancy of large equipments presents a step function in weight which prevents mission objective accomplishment with current technology.

2. Alternate Protection Mode Reduces Risk - Reduced risk can be obtained from more careful analysis, larger design margins, additional training, and additional checkout tests. These two alternatives can be extended until cost and schedule goals are seriously compromised. Risk tradeoff is essential for final determination. Meteroid shielding is a prime case of easily determined low probability from the accepted penetration models. The initial probability of penetration of a normal outside shell is already small and any additional insulation improves success in resisting micrometeroids. Protecting against all possible sizes of meteroids is patently impossible, since they can be extremely large.

3. Extensive Development Reduces Risk - Increased probability of equipment success can be achieved by increasing development. As potential failure modes are identified by analysis or test, redesign can reduce the risk to an acceptable level. Extremely unlikely failure modes require more detailed analyses and more tests for detection. Risk tradeoff is essential for final determination.

4. Operational Ease - Increased crew efficiency can be obtained by providing a shirtsleeve atmosphere in the CM cabin, but the cabin is too large to be made redundant. Another method of providing redundancy is to require the crew to wear pressurized suits at all

times, but crew inconvenience would severely limit mission duration. The single point of failure can be tolerated for the benefits derived with attention to further risk reduction.

Table B-15 identifies the single point of failure modes with specific justifications most applicable, and Table B-16 summarizes the relationships in matrix form. It is apparent from the kinds of equipment identified that further reduction of single point of failure modes would be very difficult. For example, use of highly energetic fluids is currently a prerequisite to successful accomplishment of any mission objective. Without these fluids, the mission would be currently impossible. Again, the use of high pressure tanks, piping, pumps, and rocket engines are mission requirements; no other method exists. Providing redundancy here does not help since a single rupture could cause extensive adjacent damage without extra shielding. A possibility exists with the landing legs to reduce risk by redundancy (increasing the number of legs to 5 or 6), but this adds more wieght than current analyses can afford. Double-walled propellant tanks are obviously impractical, but some later consideration of internal baffling for safety may be possible. Biological hazards are not solved by redundancy since the ingestion of toxic materials is immediately a problem. Finally, a double-walled cabin would add excessive weight; but special design of the wall for added margin, non-propagation of initial hole size, and internal patching could reduce the risk to an acceptable level.

Table B-15. Justification of Single Point of Failure Modes

Failure Mode	Justification
Explosions	Reduction of weight by use of energetic fluids Reduction of weight by use of high pressure fluid storage Provision of large safety margin reduces risk Extensive development testing reduces risk
Support Structure Break	Reduction of weight with non-redundancy Reduction of risk possible by large safety margins and extensive development
Meteoroid Penetration	Reduction of weight - excessive weight required for absolute safety Low probability of occurrence Reduction of risk by some shielding
Propellant Loss	Reduction of weight with non-redundancy of prop. tanks Reduction of risk by large safety margins and extensive development
Biological Contamination	Not solved by redundancy or inflight margin because of direct ingestion by crew Reduction of risk possible by crew training in waste management, ground inspection of packaged food, and individual sampling of potable water tanks
Cabin Rupture	Reduction of weight by eliminating double walls Easier crew motion by using shirtsleeve environment Reduction of risk possible by large safety margin and extensive development

Table B-16. Single Point of Failure Justification Matrix

Justification	Single Point of Failure Modes					
	Hardware and Crew				Crew Only	
	Explosions	Support Structure Break	Meteroid Penetration	Propellant Loss	Biological Contamination	Cabin Rupture
Weight reduction	X	X	X	X		X
Alternate protection mode reduces risk	X	X		X	X	X
Extensive development reduces risk	X	X		X		X
Operational ease						X

APPENDIX C. WEIGHTS

By F.G. Chapel, Jr.

APPENDIX C. WEIGHTS

TUG PROPELLANT MODULE STRUCTURE

The best available structural and configuration data have been used in evaluating the structural weight of the tug propellant module (PM).

The configurations of the tug specifically sized are noted in Figure C-1. All configurations have four engines (propulsion system weights, however, are not included in this analysis), one LH_2 tank and one or four LOX tanks. The criteria and ground rules used for sizing are presented in Tables C-1 and C-2.

Cylindrical tank wall thicknesses are shown in Figure C-2. The average thickness of the 1.4/1.0 elliptical ends is assumed to be proportional to the cylindrical thickness. The thickness of the external structural shell is shown in Figure C-3. The basic thickness of the 15-ft diameter (4.57m) PM was obtained from the structural analysis section. Equivalent thickness for 12-ft and 22-ft-diameter (3.66m and 6.71m) PM's were obtained by dividing the equivalent 15-ft (4.57-m) PM loading by the desired PM circumferences. It was further assumed that the thickness at the bottom of the stage increased with stage length increase, as noted in Figure C-3.

Tables C-3 through C-6 are the resulting vehicle weight data for four propellant loadings of each of the noted four configurations. Data presented in Table C-2 and Figures C-2 and C-3 are used. Figures C-4 and C-5 present plots of these data. Superimposed on the data are the structural data used in the performance analyses as of August 1970. A plot of the data is shown in Figure C-6, as RBD-133. It would appear that RBD-133 is low by about 500 pounds (~230 kg) at a 12-ft (3.66m) PM diameter and 200 to 300 pounds (~90 to 140 kg) at a 15-ft (4.57m) PM diameter.

Figure C-7 is a plot of the lengths of the four configurations. These data were cross-plotted onto Figure C-2 to illustrate the required external wall equivalent thicknesses.

Table C-7 presents the resulting parametric equations of the weight analysis. The data are approximate (±10 pounds or ±5 kg) but are considered to be well within the limits of parametric weight data.

INTEGRAL VERSUS NON-INTEGRAL STRUCTURE

An additional analysis was performed to evaluate the difference between the nonintegral concept, as previously presented, and and integral structural concept. The integral concept envisions use of the cylindrical portions of the propellant tankage as primary flight-load-carrying structure as well a fluid retention structure. The tank structure remained an aluminum alloy; however, the interstage and skirt structures were changed to boron epoxy to minimize heat transfer. With same insulation unit weight used as presented in the nonintegral concept (for durations under 14 days), plots of the resulting PM weight data were computed. Figure C-8 presents the results of this analysis. In the figure a difference in PM inert weight of 200 to 300 pounds (90 to 140 kg) at a propellant load of 80,000 pounds (36,000 kg) is noted, while at a load of 40,000 pounds (18,000 kg) a negligible inert weight change is noted. If longer-term cryogenic storage is considered, this difference may become smaller. If other structural configurations are considered (i.e. multiple LOX tanks), this trend may even be reversed. The final design selection of integral or nonintegral tankage is, therefore, beyond the scope of this study and is a subject for future tug studies.

4 ENGINES—AFT DOCKING
1 LH_2 TANK + 1 OR 4 LOX TANKS

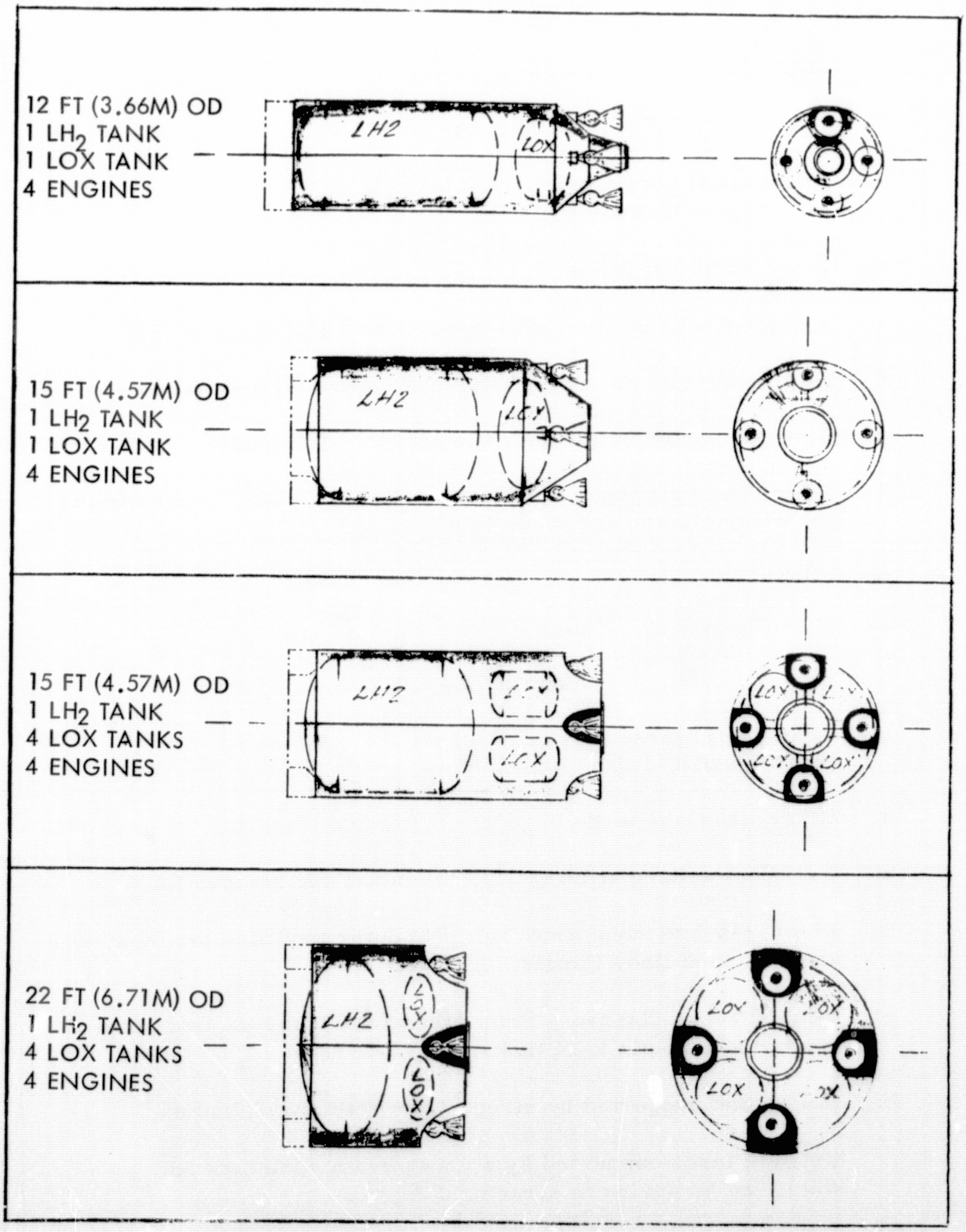


Figure C-1. Structural Configurations

Table C-1. Sizing Criteria, LH_2 and LOX Tanks

1. LH_2 tank
   a. 2014-T6-type aluminum
   b. F_{TU} = 85.0 KSI (58,500 N/cm^2), F_S = 1.5, FNO = 1.33
   c. Operational pressure = 19.0 psia (13.1 N/cm^2), (See Figure C-2 for $\bar{t}$.)
   d. 1.4/1 elliptical heads
   e. 8 percent volume allowance for residual and trapped LH_2
   f. 8-inch (20-cm) clearance from PM outer moldline for insulation
   g. Tank supported by straps from structural frames
   h. 24-in. (61-cm) separation between LH_2 and LOX tank moldline for sump, lines, and insulation
2. LOX tank
   a. 2014-T6-type aluminum
   b. F_{TU} = 73.5 KSI (50,600 N/cm^2), F_S = 1.5, FNO = 1.33
   c. Operational pressure = 24.0 psia (16.5 N/cm^2) (See Figure C-2 for t.)
   d. 1.4/1 elliptical heads
   e. 8 percent volume allowance for residual and trapped LOX
   f. 18-in. (46-cm) clearance from PM outer moldline for insulation and LH_2 feed lines (single LOX tank)
   g. 6-in. (15-cm) clearance from other LOX tanks or structural supports (multiple LOX tanks)
   h. Single tank supported by straps from structural frames
   i. Multiple tanks supported by a cross beam structure and stabilized by straps to structural frames

Table C-2. Sizing Criteria, External Structure

Insulation, and Meteoroid Protection

1. External structure

   a. 2014-T6-type aluminum

   b. F_{TU} = 60.0 KSI (41,400 N/cm^2) at 70 F (295 K)

   c. F_S = 2.0, FNO = 1.5

   d. t for 30K lb (13.600 Kg) Wp, 15 feet (4.57 M) outside diameter lunar lander derived from Paragraph 2.1.2.

   e. t assumed to vary directly with length and inversely with diameter. (See Figure C-3.)

   f. Stiffended by tank support frames -1.4 in.^{2} (9 cm^2) cross sectional area

      12 ft (3.66m) OD - 64 lb (29 kg)
      15 ft (4.57m) OD - 80 lb (36 kg)
      22 ft (6.71m) OD - 117 lb (53 kg)

   g. Frame added at top for IM attachment

2. Insulation and meteoroid protection

   a. LH_2 tank, high performance insulation (HPI) with two single sheet F/G covers

2-in. (5.1-cm) HPI of 20 layers/in. =	0.40 lb/ft^2 (1.95 kg/m^2)
F/G Covers =	0.10 lb/ft^2 (0.49 kg/m^2)
Total LH_2 Insul.	0.50 lb/ft^2 (2.44 kg/m^2)

   b. LOX Tank - HPI + F/G Cover

1/2 inch (1.3 cm) HPI of 20 layers/in	0.10 lb/ft^2 (0.49 kg/m^2)
F/G cover	0.10 lb/ft^2 (0.49 kg/m^2)
Total LOX insulation	0.20 lb/ft^2 (0.98 kg/m^2)

NOTE: AVE $\bar{t}$ OF TANK ELLIPTICAL END DOMES
$= (1.4/2) = 0.7 \times$ TANK WALL $\bar{t}$

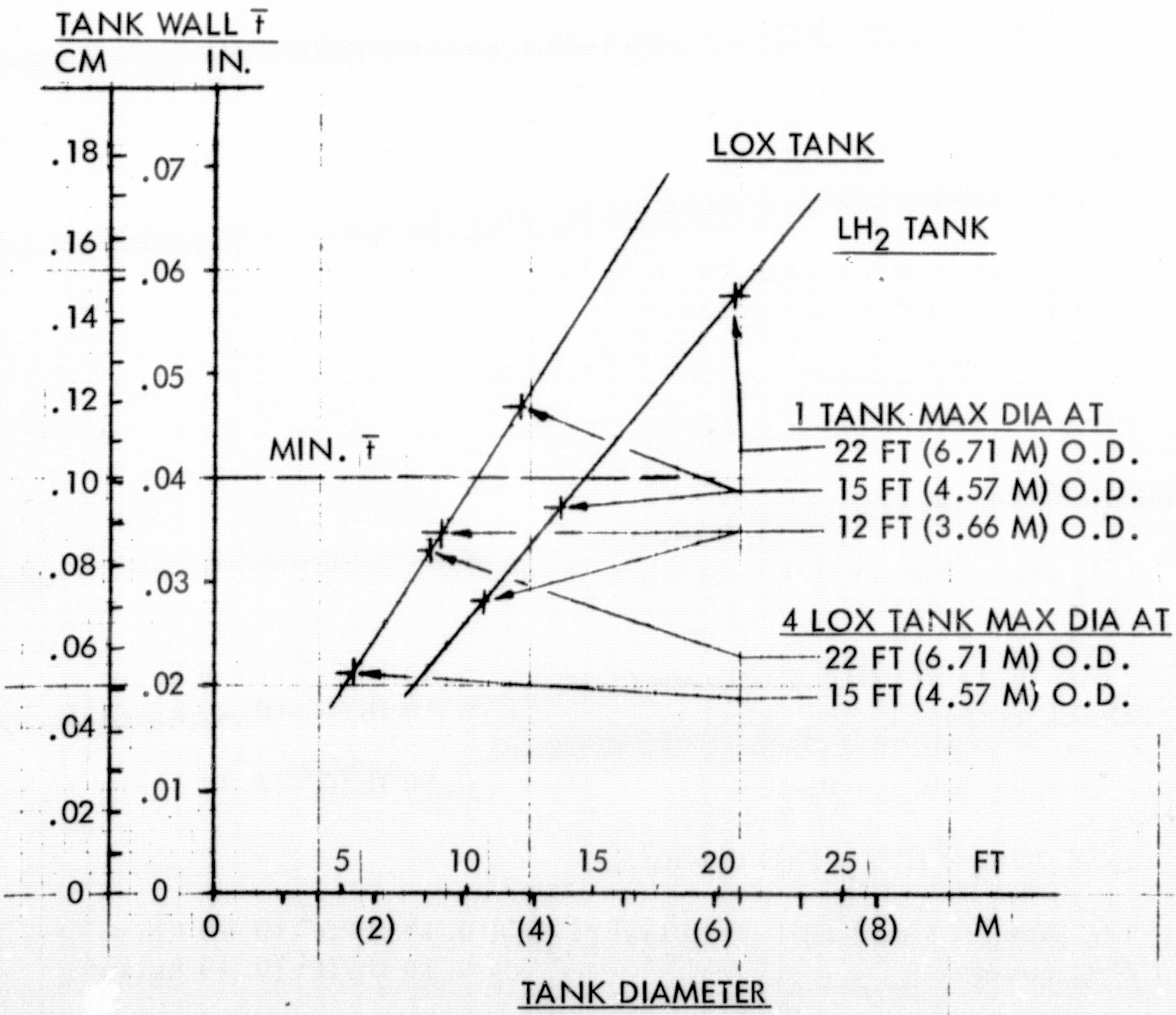


Figure C-2. LH_2 and LOX Tank Wall $\bar{t}$

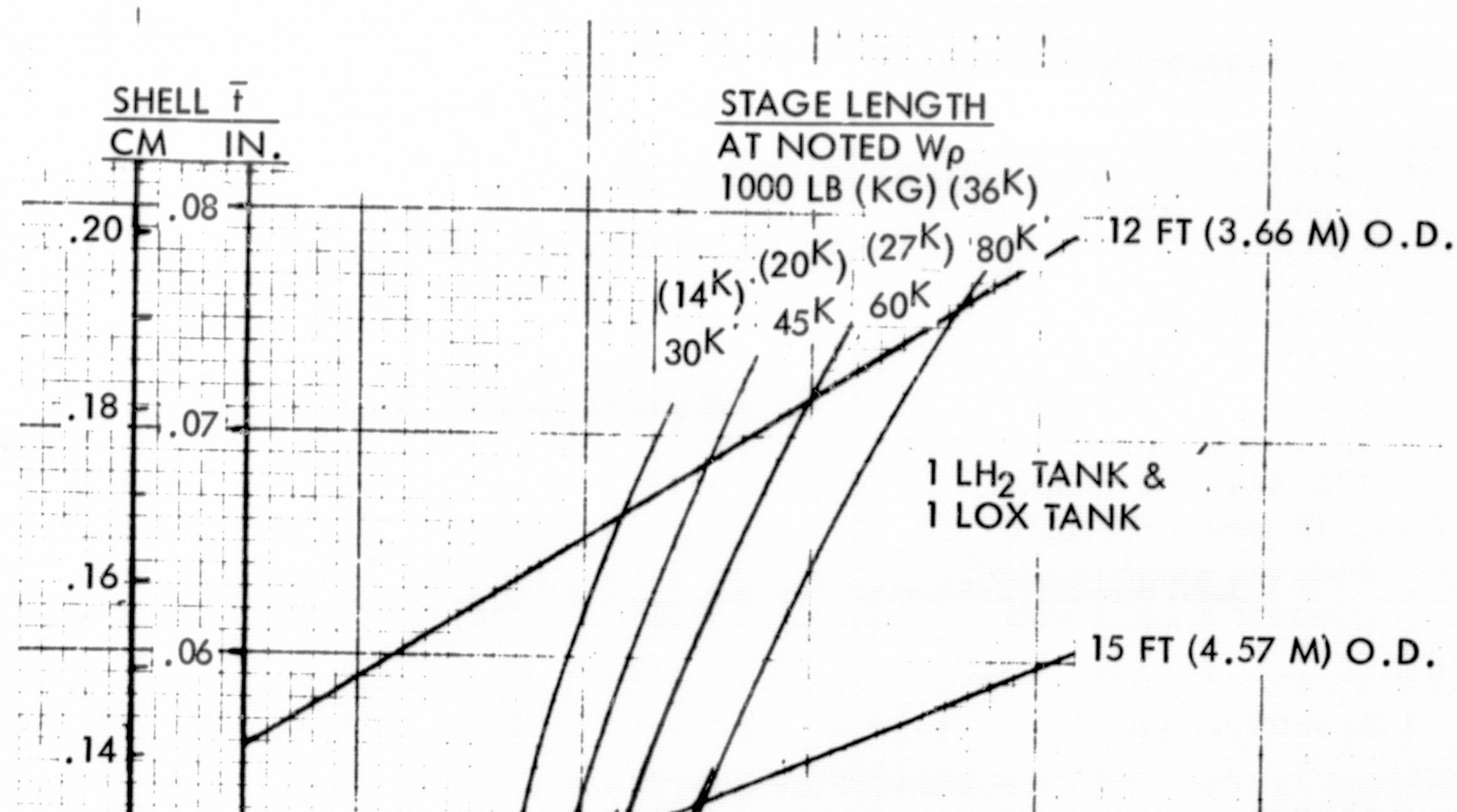


Figure C-3. External Structural Shell $\bar{t}$

Table C-3. Structural Weight Summary,

One LH_2 Tank, One LOX Tank, Apollo-Type

Passive Docking, For Four Engines

Stage outside diameter - ft	12.0 ft (3.6m)			
Propellant Capacity (10^3 lb)	21.0	40.0	60.0	80.0
Dimensions (ft)				
LH_2 tank dia	10.7	10.7	10.7	10.7
LOX tank dia	9.0	9.0	9.0	9.0
LH_2 tank cyl length	3.7	11.9	20.5	29.1
LOX tank cyl length		3.9	8.0	12.1
Full dia shell length	16.5	28.6	41.3	54.0
Overall stage length	25.5	37.6	50.3	63.0
LH_2 tank weight	(540)	(845)	(1,165)	(1,490)
Domes	177	177	177	177
Cylinder	75	239	412	585
Straps	70	70	70	70
Insulation	218	359	506	658
LOX tank weight	(260)	(350)	(445)	(540)
Domes	127	127	127	127
Cylinder(s)	--	67	138	209
Cross Beam	--	--	--	--
Straps	90	90	90	90
Insulation	43	66	90	114
Stage structure	(1,140)	(15,560)	(2,025)	(2,520)
External shell	530	950	1,415	1,910
Frames (No.)	(5) 320	(5) 320	(5) 320	(5) 320
Thrust structure	170	170	170	170
Docking structure	120	120	120	120
Space operations - (lb)	1,940	2,755	3,635	4,550
- (kg)	(880)	(1,250)	(1,649)	(2,064)

Table C-4. Structural Weight Summary,

One LH_2 Tank, One LOX Tank,

Passive Neuter Docking, For Four Engines

Stage outside diameter - ft	15.0 ft (4.6m)			
Propellant Capacity (10^3 lb)	25.0	40.0	60.0	80.0
Dimensions (ft)				
LH_2 tank dia	13.7	13.7	13.7	13.7
LOX tank dia	9.3	10.9	12.0	12.0
LH_2 tank cyl length	--	3.9	9.1	14.2
LOX tank cyl length	--	--	1.2	3.4
Full dia shell length	15.2	19.6	26.4	33.7
Overall stage length	22.2	26.6	33.4	40.7
LH_2 tank weight	(635)	820)	(1,055)	(1,295)
Domes	294	294	294	294
Cylinder	--	98	227	356
Straps	90	90	90	90
Insulation	251	338	444	555
LOX tank weight	(290)	(360)	(465)	(570)
Domes	135	186	226	226
Cylinder(s)	--	--	30	86
Cross Beam	--	--	--	--
Straps	110	110	110	110
Insulation	45	64	99	148
Stage structure	(1,425)	(1,570)	(1,805)	(2,060)
External shell	485	630	865	1120
Frames (No.)	(5) 400	(5) 400	(5) 400	(5) 400
Thrust structure	180	180	180	180
Docking structure	360	360	360	360
Space operations - (lb)	2,350	2,750	3,325	3,925
- (kg)	(1,066)	(1,447)	(1,508)	(1,780)

Table C-5. Structural Weight Summary,

One LH_2 Tank, Four LOX Tanks,

Passive Neuter Docking, For Four Engines

Stage outside diameter - ft	15.0 ft (4.6m)			
Propellant Capacity (10^3 lb)	25.0	40.0	60.0	80.0
Dimensions (ft)				
LH_2 tank dia	13.7	13.7	13.7	13.7
LOX tank dia (4)	5.5	5.5	5.5	5.5
LH_2 tank cyl length	--	3.9	9.1	14.2
LOX tank cyl length	0.7	2.8	5.6	8.4
Full dia shell length	13.5	19.5	27.5	35.4
Overall stage length	18.5	24.5	32.5	40.4
LH_2 tank weight	(635)	(820)	(1,055)	(1,295)
Domes	294	294	294	294
Cylinder	--	98	227	356
Straps	90	90	90	90
Insulation	251	338	444	555
LOX tank weight (4)	(505)	(615)	(770)	(920)
Domes	190	190	190	190
Cylinder(s)	28	113	226	339
Cross beam	100	100	100	100
Straps	110	110	110	110
Insulation	77	102	144	181
Stage structure	(1,370)	(1,570)	(1,845)	(2,130)
External shell	430	630	905	1,190
Frames (No.)	(5) 400	(5) 400	(5) 400	(5) 400
Thrust structure	180	180	180	180
Docking structure	360	360	360	360
Space Operations - (lb)	2,510	3,005	3,670	4,345
- (kg)	(1,139)	(1,365)	(1,665)	(1,973)

Table C-6. Structural Weight Summary,

One LH_2 Tank, Four LOX Tanks,

Passive Neuter Docking, For Four Engines

Stage outside diameter - ft	22.0 ft (6.7m)			
Propellant Capacity (10^3 lb)	20.0	40.0	60.0	80.0
Dimensions (ft)				
LH_2 tank dia	12.7	16.0	18.3	20.2
LOX tank dia (4)	5.6	7.0	8.1	8.8
LH_2 tank cyl length	--	--	--	--
LOX tank cyl length	--	--	--	--
Full dia shell length	13.1	16.4	18.9	20.7
Overall stage length	18.1	21.4	23.9	25.7
LH_2 tank weight	(630)	(950)	(1,245)	(1,565)
Domes	252	455	652	890
Cylinder	--	--	--	--
Straps	160	150	140	130
Insulation	218	345	453	545
LOX tank weight (4)	(575)	(725)	(865)	(965)
Domes	197	309	413	488
Cylinder(s)	--	--	--	--
Cross beam	150	150	150	150
Straps	160	160	160	160
Insulation	68	106	142	167
Stage structure	(1,545)	(1,680)	(1,780)	(1,850)
External shell	525	660	760	830
Frames (No.)	(3) 350	(3) 350	(3) 350	(3) 350
Thrust structure	230	230	230	230
Docking structure	440	440	440	440
Space operations - (lb)	2,750	3,355	3,890	4,380
- (kg)	(1,447)	(1,522)	(1,765)	(1,987)

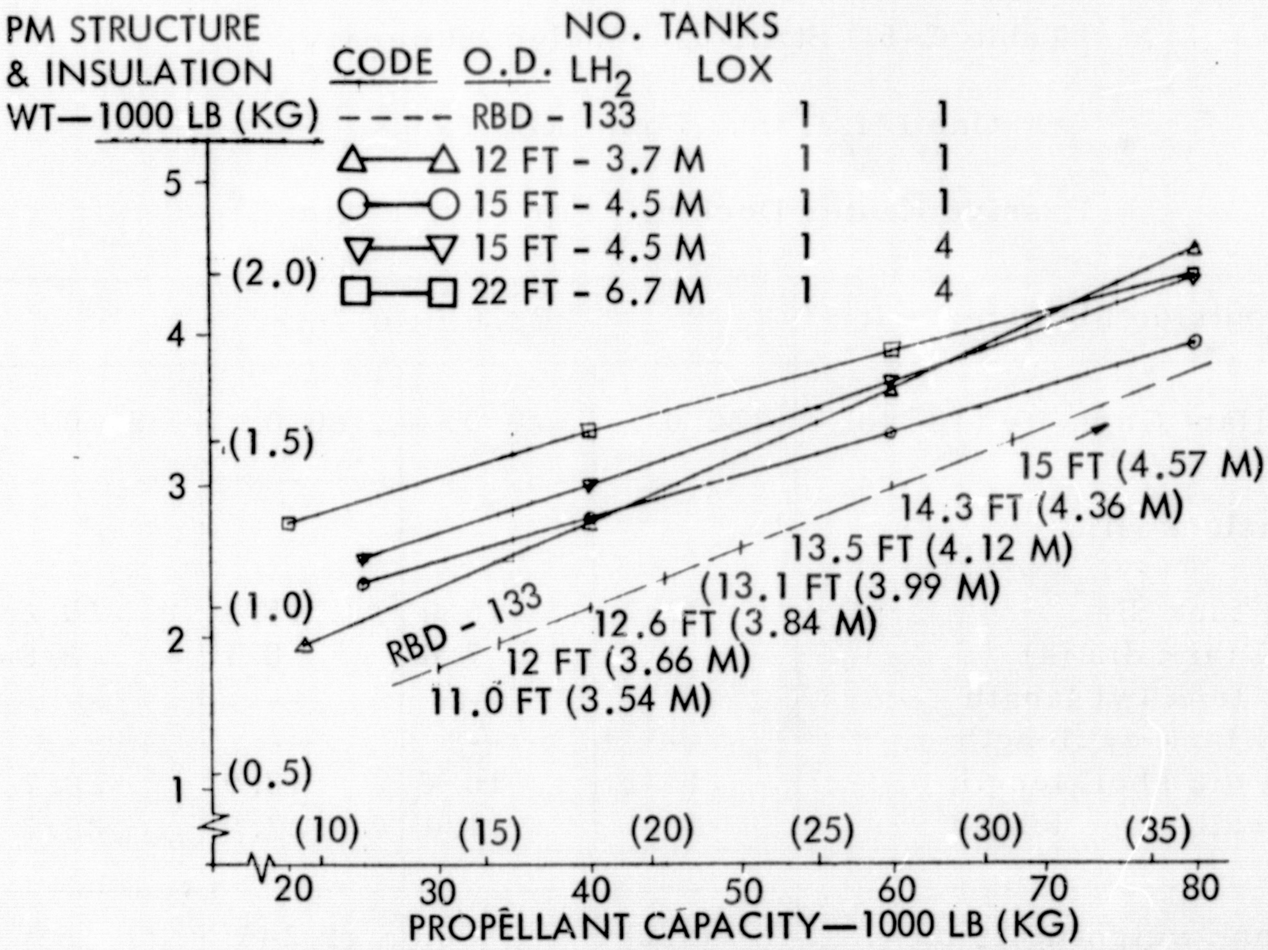


Figure C-4. PM Structure Weight Versus Propellant Weight

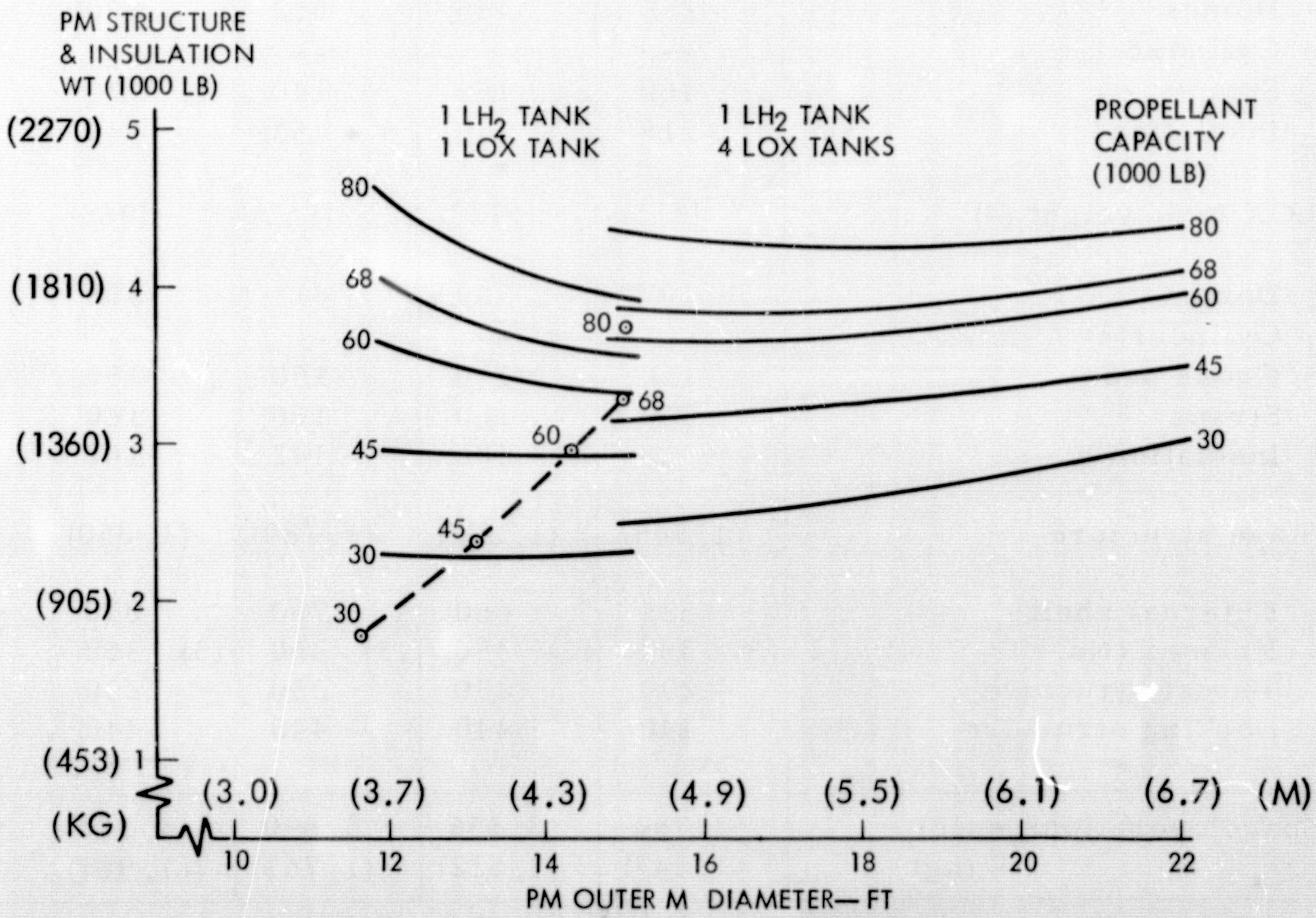


Figure C-5. PM Structure Weight Versus PM Diameter

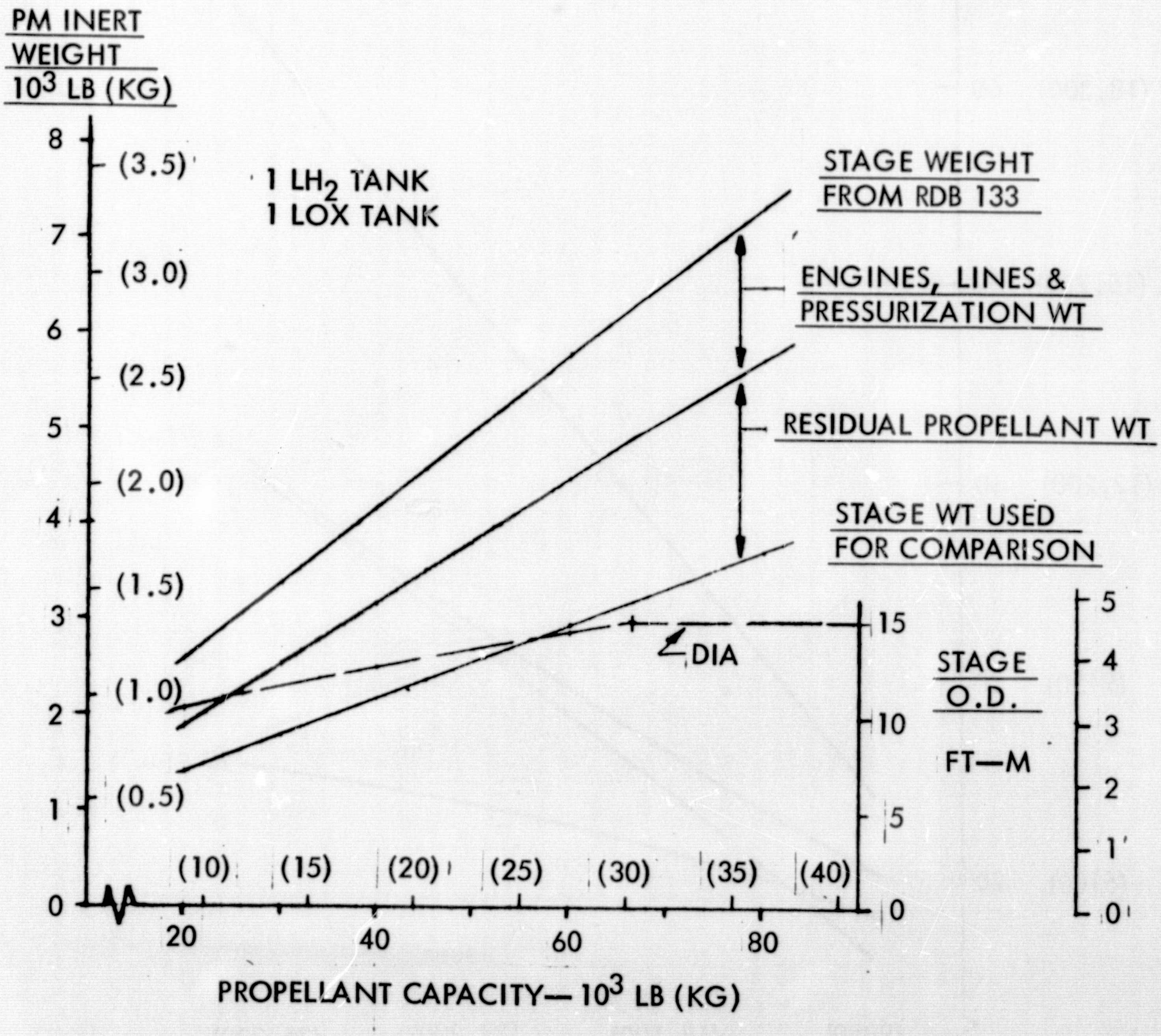


Figure C-6. RBD 133 PM Weight Data

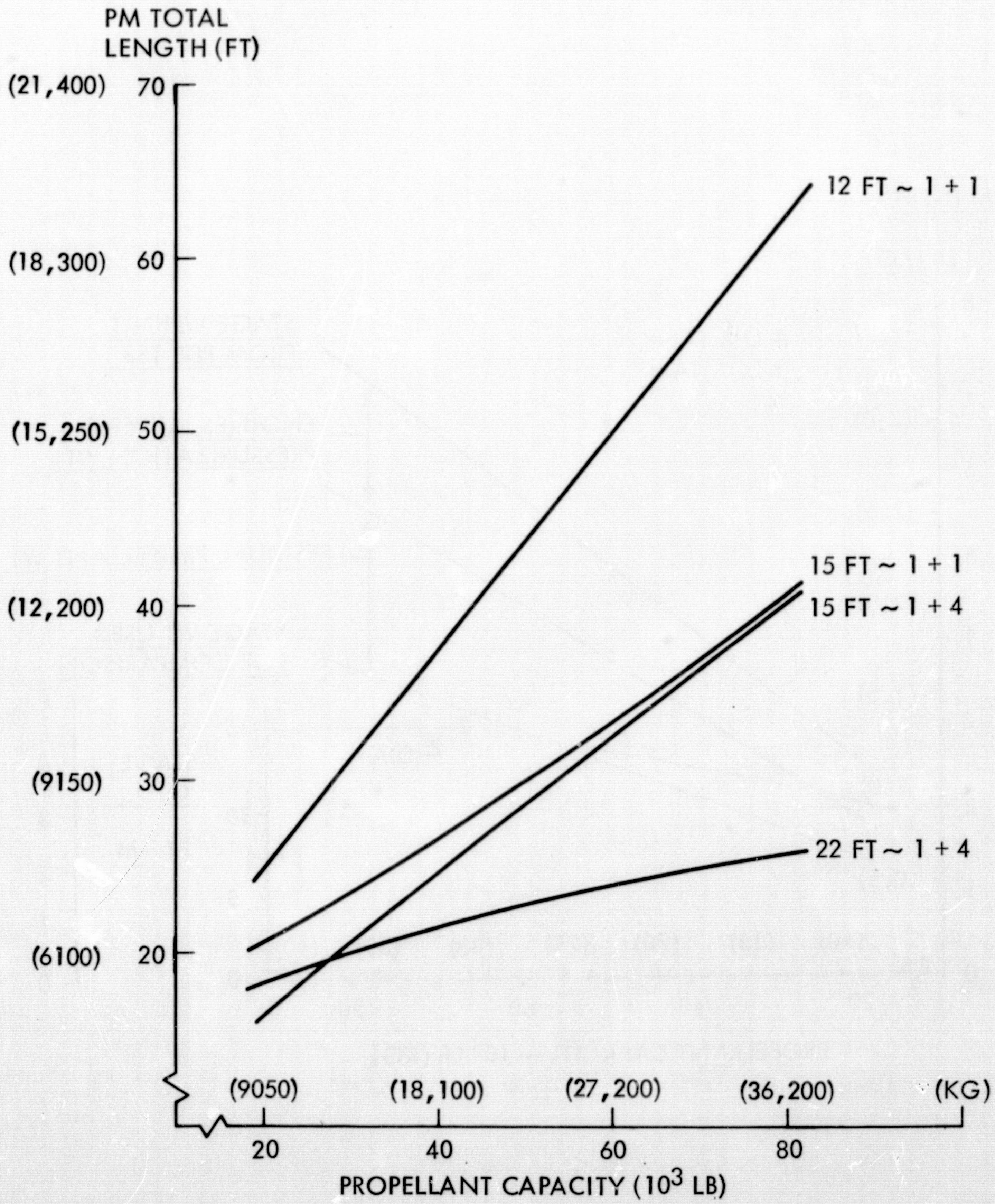


Figure C-7. PM Length Versus Propellant Weight

Table C-7. Parametric Weight Results

Configuration		English Units (W_p in pounds)	Metric Units (Wp in kilograms)
Variable diameter			
RBD 133 Program,	wt	= $630. + 39.0*10^{-3}*Wp$, lb	= $285.8 + 39.0*10^{-3}*Wp$, kg
one LOX + one LH_2 tank,	lgth	= $9.00 + 88.2*10^{-6}*Wp$, ft	= $2.74 + 59.4x10^{-6}*Wp$, m
		= 15.00 when $Wp>68^K$ lb	= 4.57 when $Wp>31^K$ kg
Fixed diameter			
12.0 ft, 3.66 m one LOX + one LH_2 tank	wt	= $1040. + 43.2*10^{-3}*Wp$, lb when $Wp>21^K$ lb	= $471.7 + 43.2*10^{-3}*Wp$, kg when $Wp>9.5^K$ kg
15.0 ft, 4.57 m one LOX + one LH_2 tank	wt	= $1635. + 28.6*10^{-3}*Wp$, lb when $Wp>25^K$ lb	= $741.7 + 28.6*10^{-3}Wp$, kg when $Wp>11.3^K$ kg
15.0 ft, 4.57 m four LOX + one LH_2 tank	wt	= $1660. + 33.6*10^{-3}*Wp$, lb when $Wp>25^K$ lb	= $753.0 + 33.6*10^{-3}*Wp$, kg when $Wp>11.3^K$ kg
22.0 ft, 6.71 m four LOX + one LH_2 tank	wt	= $2125. + 28.8*10^{-3}*Wp$, lb when $20^K< Wp<60^K$ lb = $2460 + 24.0*10^{-3}*Wp$, lb when $Wp>60^K$ lb	= $964.0 + 28.8*10^{-3}*Wp$, kg when $9.1<Wp<27.2$ kg = $1115.9 + 24.0*10^{-3}*Wp$, kg when $Wp>27.2^K$ lb

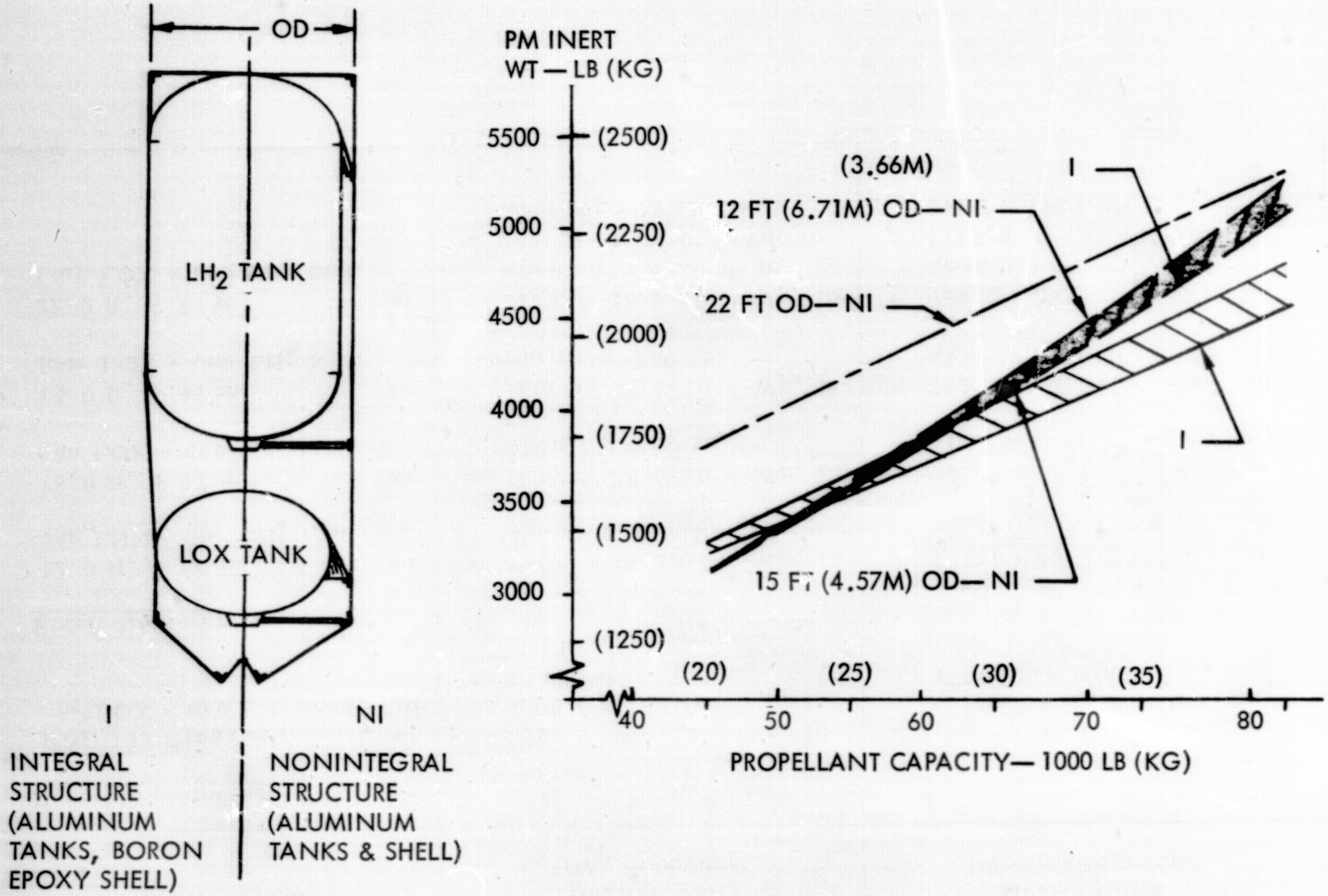


Figure C-8. PM Structure Weight Comparison

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APPENDIX D
ENVIRONMENTAL (METEOROID) PROTECTION

By A.J. Richardson

APPENDIX D. ENVIRONMENTAL (METEOROID) PROTECTION

ENVIRONMENT

In the RST study the meteoroid environment as described by NASA SP 8013 was employed to establish protection requirements. The environment is composed of sporadic and stream cometary particles distributed throughout near-earth space, and secondary ejecta particles occurring on the lunar surface. The cometary particles are of low density (0.5 grams per cubic centimeter) and could impact at very high speed (20 kilometers per second average). The secondary ejecta particles are lunar surface debris and would impact at very low velocities (0.1 kilometers per second).

ALLOWABLE DAMAGE

The design meteoroid damage which will be permitted to the RST must be established before protection requirements can be computed. The critical components in the RST are the tanks and plumbing, and past analysis has shown that accepting some damage to these will reduce or eliminate shielding. Tests and theory have shown that partial penetration and local bulge damage to these components can be tolerated and for this study it was assumed the penetration into the aluminum tank wall equal to or less than 0.55 times the wall thickness was acceptable.

PROTECTION WEIGHT

Figures D-1 through D-3 show parametric meteoroid protection weight data for a space craft operating in near-earth space. No planetary shielding or gravitational effects are included. The weights shown are a function of the RST surface area-design time product (AT) and the propellant tank thickness (t_r). The single bumper protection is in the form of a bumper (t_1) and an inner shield (t_s). The dual bumper shield is the same except a second bumper shield is the same except a second bumper (t_2) is added. No credit was taken for the insulation, which would be located next to the tank wall. Minimum gage limits were set for the bumpers; for the lower AT values, this material plus the pressure wall provides sufficient protection. As the value of AT is increased, a point is reached where thicker bumpers are required and shielding must be added to the pressure wall ($t_s > 0$).

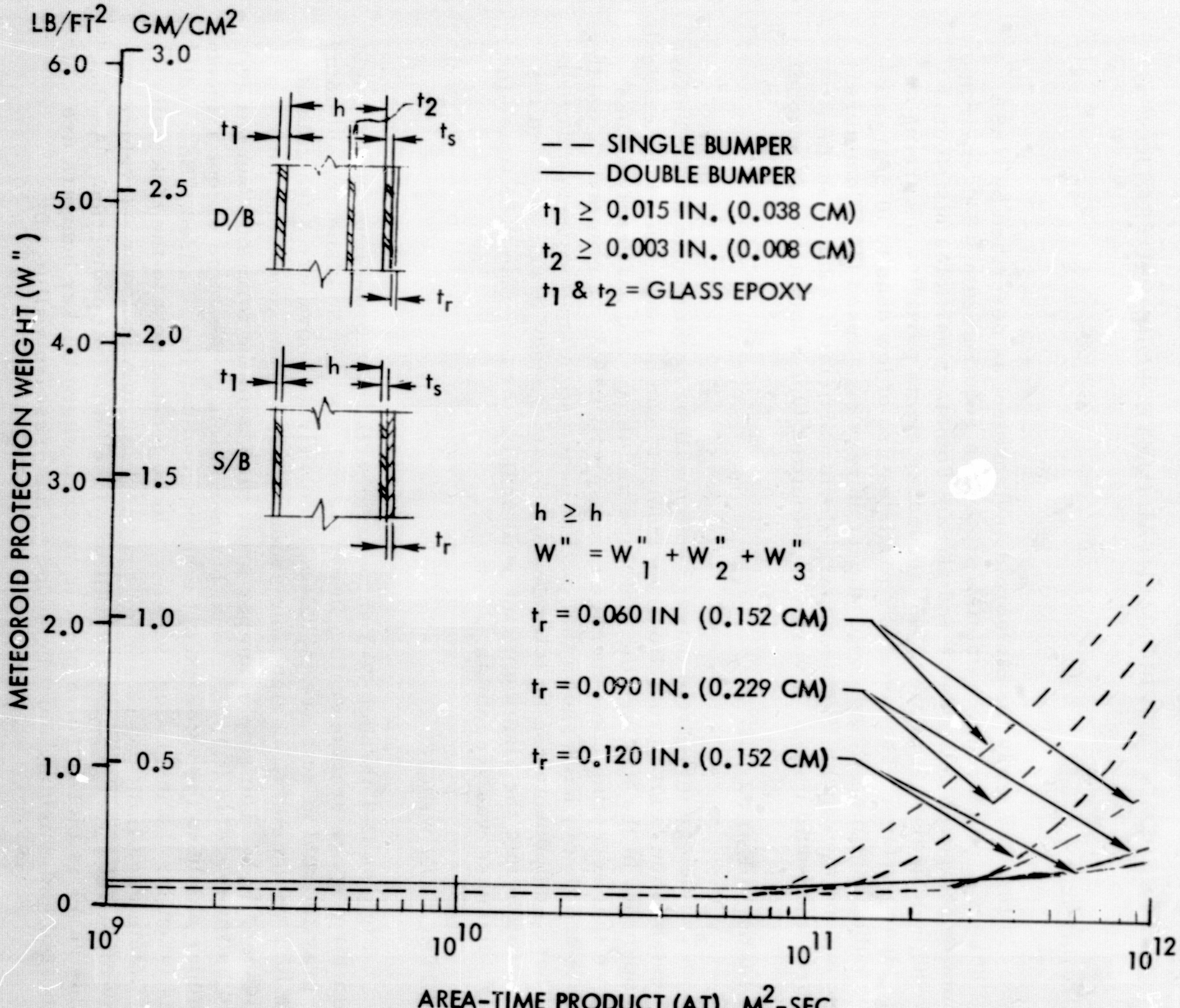


Figure D-1. Comparison of Meteoroid Shield Types for Space Tug — $P_{nf} = 0.9$

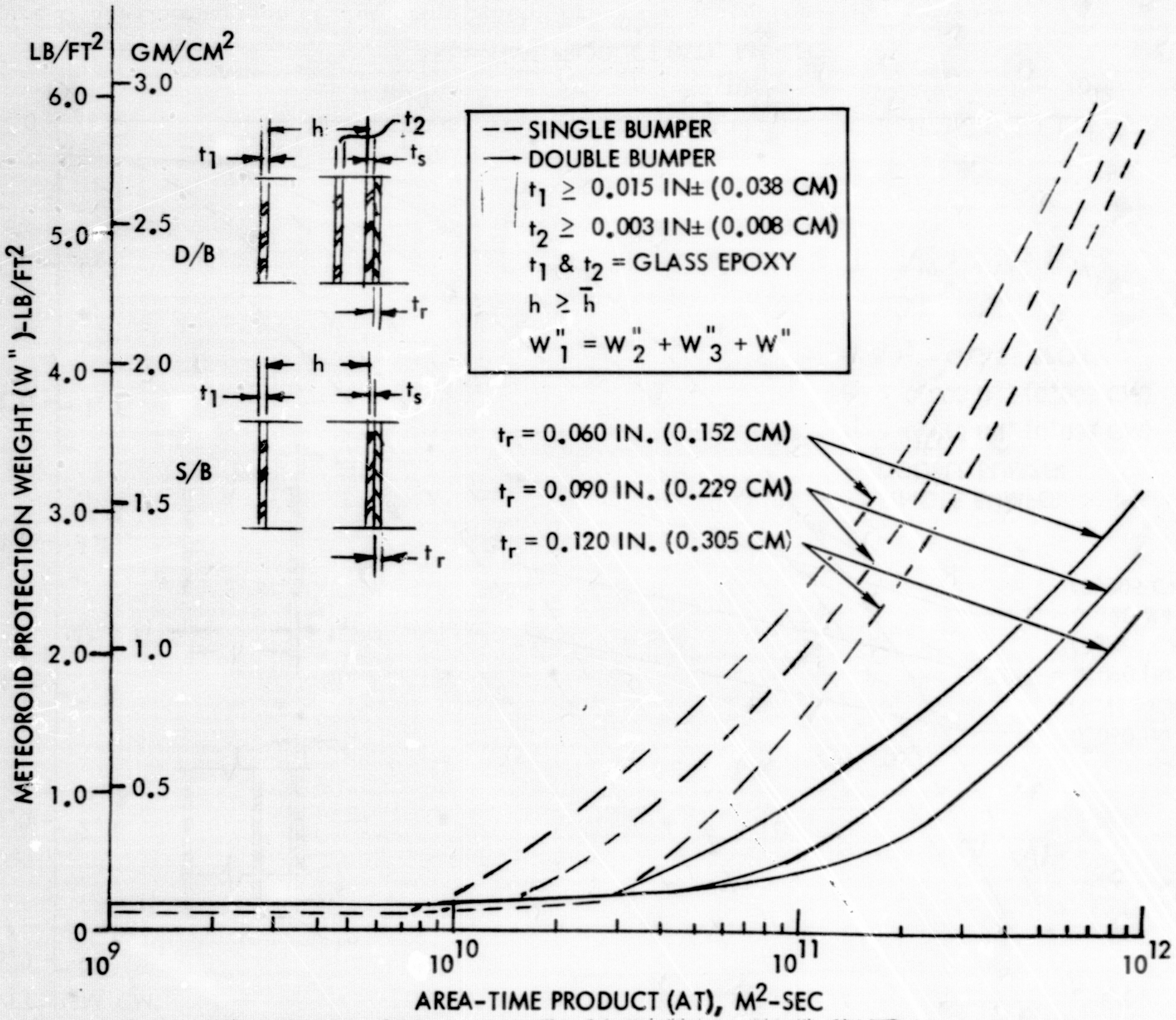


Figure D-2. Comparison of Meteoroid Shield Types for Space Tug — P_{nf} = 0.99

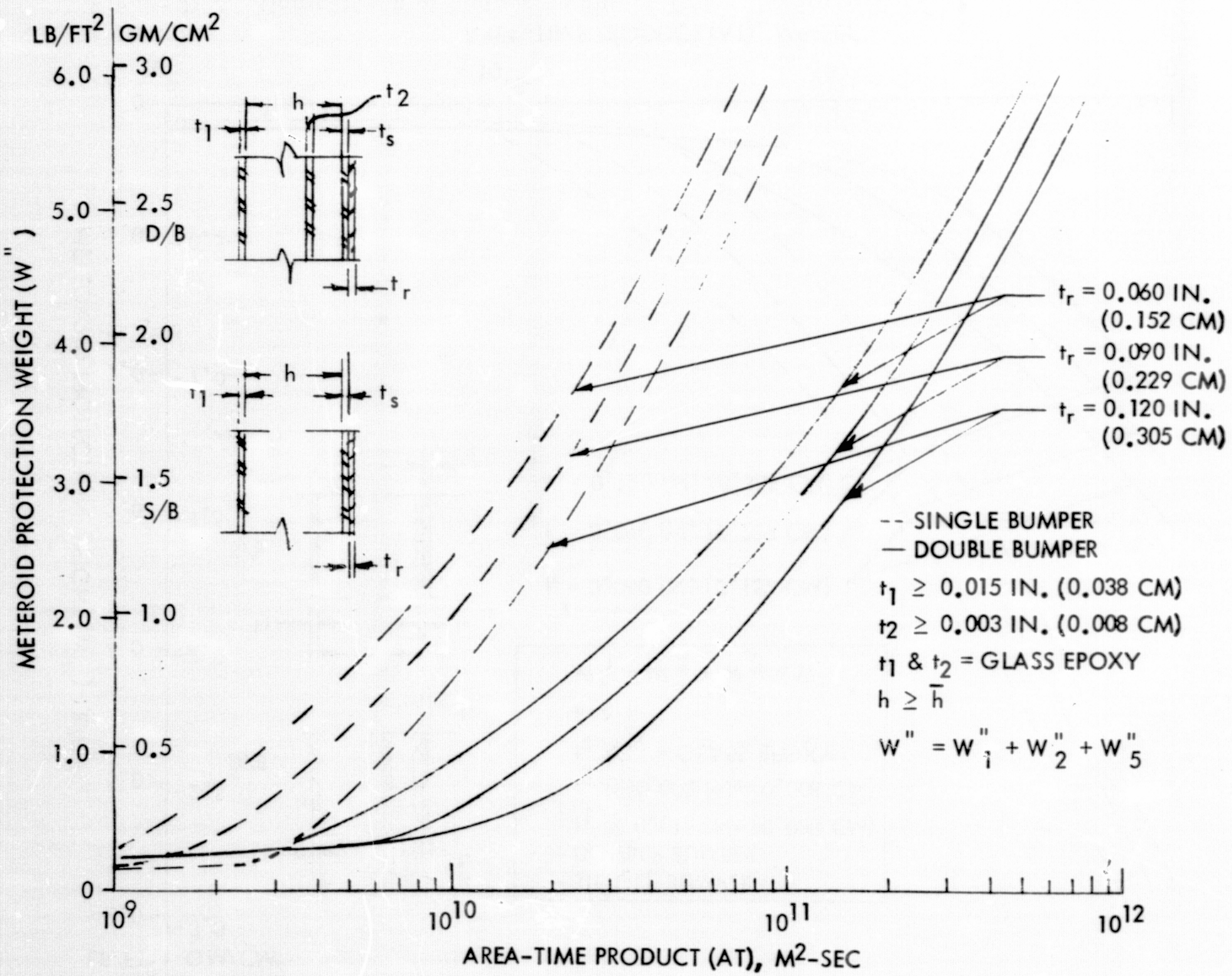


Figure D-3. Comparison of Meteoroid Shield Types for Space Tug— $P_{nf} = 0.999$

PROTECTION PHILOSOPHY

For advanced manned spacecraft missions the philosophy by which protection requirements will be established in not clear. Two general approaches can be summarized as follows.

1. Define the probability requirement for a design mission basis so that on a given mission there is a guaranteed overall probability of no module failure (P_{nf}) due to meteoroid impact, considering all modules required to complete the mission. The probability requirement for each module (P_{nfi}) is something greater than P_{nf} so that

$$P_{nf} = \prod_{i=1}^{n} P_{nfi}.$$

2. Define a probability requirement for the RST without consideration of the other modules and let it apply over the design life of the RST.

Applying Case 1 to the RST, let $P_{nf} = 0.99$ and let the requirement for the RST be 0.999. Let the RST design mission duration be 45 days and tank thickness 0.06 inch (0.153 centimeter). The surface area of a 60,000 pound tug, its instrument module, and payload is 230 square meters and the resulting AT value is 8.95×10^8 square meters-second. Refering to Figure D-3 the meteoroid protection required is minimum gage or approximately 0.2 pound per square foot (0.1 gram per square centimeter).

Applying Case 2, let the probability requirement for the RST be 0.99 and its design life be 3 years. The resulting AT value is 2.18×10^{10} square meters-second. In Figure D-2 (for $t_r = 0.06$) the meteoroid protection required is 0.25 pound per square ft (0.12 gram per sqaure centimeter) with the dual bumper and 0.8 pound per square foot (0.4 gram per square centimeter with single bumper shielding.

For this study it was assumed that Case 1 applied with the result that minimal shielding would be required for the RST. Consideration of the insulation present would further minimize this problem for RST. These findings apply to both the integral and non-integral type vehicles. Consideration was not given to protection against the secondary ejecta particles. NR studies on lunar shelters has indicated that protecting against these particles will be more difficult than for the cometary particles, providing the ejecta environment modeling is correct. A promising solution is to employ lunar soil for protection, a solution which could be applied to a RST if extended lunar stay time is required.

APPENDIX E

PROPULSION

By J. F. Nichols

APPENDIX E. PROPULSION

MAIN PROPULSION

The objectives of main propulsion studies were to establish the principal characteristics — performance, weight and dimensions — of the main propulsion system (MPS) best suited for each of the competing reusable space tug (RST) concepts. This involved, first, a parametric investigation of the influence of major MPS design variables; second, a series of design refinement studies concerned with MPS internal optimizations and improvements in parametric phase design assumptions; and, throughout the study, the selection and description of MPS design characteristics that the study evaluation indicated were best for each RST concept.

SYSTEM PARAMETRIC STUDIES

Synthesis Model

The computer model used in the parametric phase for design synthesis of the propulsion module computes weights and dimensions of major elements of the system. It uses material and fluid properties and design characteristics as input data. Empirical scaling coefficients are used only for complex subassemblies. Table E-1 is an abbreviated listing of the input variables. Propellant and pressurization system initial and final weights are based on an end-point analysis that used initial and final pressures and temperatures and fluid properties. The primary structure model is illustrated in Figure E-1. It is necessarily more simplified than the propulsion system model, but is capable of rational scaling if single point design data are used as the baseline. The resulting structure scaling is sensitive to many independent variables, including, besides those shown, several primary propulsion variables such as propellant densities and mixture ratio.

Alternate Propellants

Four propellant combinations were considered as competitors to LOX-hydrogen. These were fluorine-hydrogen, flox-methane, nitrogen tetroxide-aerozine 50, and flox-hydrogen. The propulsion synthesis program was used to develop the propulsion module (PM) weight properties needed to evaluate these systems. Table E-2 is an abbreviated list of input data showing the major performance variable assumptions for each candidate. Table E-3 shows the resulting PM propellant fraction at a reference size of 60,000 pounds (27200 kilograms) of usable propellant, along with the PM size needed to place a 10,000-lb (4536-kg) payload in synchronous orbit (reference mission).

Table E-1. Propulsion System Synthesis Model
Major Input Parameters

Propellant System	Pressurization System	Engine System
Fluid properties	Fluid properties	Engine scaling coeff
Material properties	NPSP start/run	Line velocities
Number of tanks	Helium states	TVC actuator scaling
Head shape	Bleed gas states	Gimbal mount scaling
Dia constraint	Heater scaling coeff	
Minimum gage	Vessel design	
Internals	Insulation allowance	
Residuals Est		
Insulation thickness		

This study revealed that the performance of Earth-storable propellants NTO-A50, is inadequate for a recoverable tug of practical size. The remaining propellants use either fluorine or fluorine compounds. Thus safety and materials compatibility problems arise because of the high toxicity and corrosion properties of such propellants. It is considered that none of these show sufficient performance gain to offset this disadvantage for RST operations.

Parametric Phase Propulsion Concept Selection

As described in a following section (engine system studies), existing, modified, and advanced new engines were considered for application to the tug propulsion systems. For evaluation of these six concepts, their impact on the total propulsion module was needed. This was accomplished by using the propulsion synthesis program with variations in primary input data, as shown in Table E-4. The resulting weight characteristics are also shown in the table. Their performance on the synchronous (reference) mission is shown in Figure E-2. These results revealed the RST to be extremely performance limited. It also revealed the importance of low NPSH requirements and the effect of high specific impulse. For a recoverable concept, this result argues strongly in favor of an advanced new engine tailored to RST requirements.

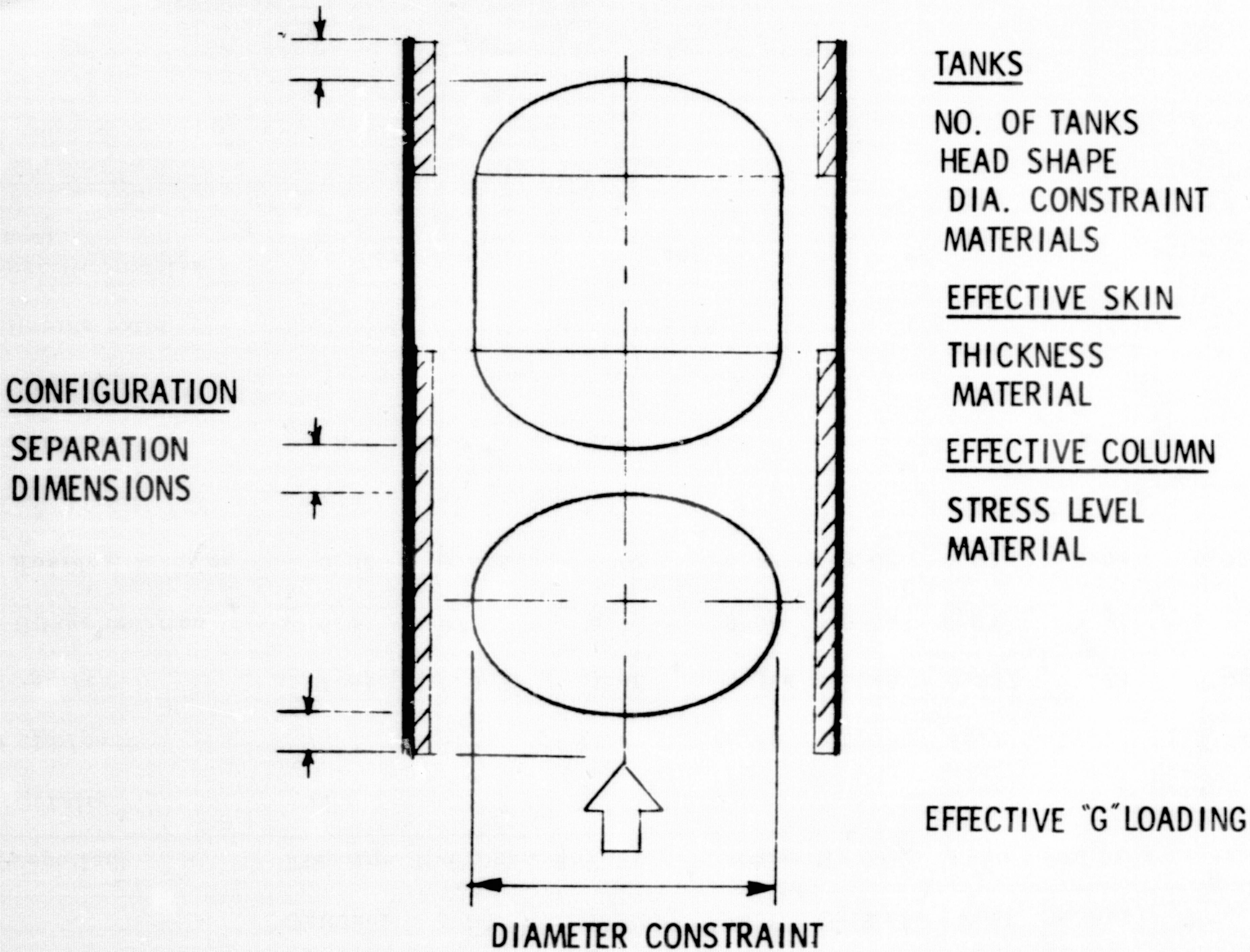


Figure E-1. PM Primary Structure Synthesis Model

Table E-2. Alternate Propellants Design Data

	Oxidizer	Fuel	Oxidizer	Fuel	Oxidizer	Fuel	Oxidizer	Fuel
Propellant	Fluorine	Hydrogen	85% Flox	Methane	N_2O_4	A-50	30% Flox	Hydrogen
Density	94							
Lb/cu ft	94.2	4.24	91.2	26.46	90.5	56.1	77.0	4.24
gm/cc	1.51	0.0678	1.46	0.424	1.45	0.897	1.23	0.0678
Ullage fraction	0.1	0.1	0.1	0.06	0.04	0.03	0.1	0.1
Residual fraction	0.02	0.025	0.02	0.02	0.02	0.02	0.02	0.025
Tank Material								
Aluminum	X	X	X				X	X
Titanium				X	X	X		
Mixture ratio	13		5.75		1.6		6	
Specific impulse (sec)	477		405		325		467	

Space Division
North American Rockwell

Table E-3. Alternate Propellant Comparison

	PM Propellant Fraction[2]	PM Propellant Required[1] (lb)	(kg)
Fluorine-hydrogen	0.946	56,300	25,500
82% Flox-methane	0.956	83,000	37,600
Nitrogen tetroxide-aerozine 50	0.963	240,000	109,000
30% Flox-hydrogen	0.932	64,200	29,100
LOX-hydrogen concept 6	0.930	70,000	31,700

Notes:
1. For reference synchronous equatorial mission
2. At 60,000 lb (27100 kg) Propellant Capacity

In a subsequent effort, a four-engine design was selected as a baseline (designated as Concept 7). Its principal design characteristics are shown in Table E-5. The table also shows propellant, pressurization, tank, insulation and structure assumptions used as input to the synthesis program for space-based designs. A schematic of the selected system is shown in Figure E-3.

The concept incorporates an engine-bleed gas circuit for tank run pressurization. For starting, several options are available. A pumped idle start would permit a bootstrap sequence until engine bleed is adequate. A faster start would be provided by the option shown for start pressurization from the ACS accumulators.

Thermal management provisions include (1) destratification systems to minimize ullage pressure rise during prolonged near-full periods; (2) thermodynamic vents to allow boiloff at zero g - mixed flow is expanded through a heat exchanger sized to accept any quality and yet assure only vapor at exit; and (3) a regenerator to intercept heat leak for use on the lunar surface or prolonged space storage but unnecessary for short space missions.

Capillary screens and channels (wicks) are incorporated for propellant retention and low-rate feed-out in zero g. This provision is a development risk item and backed up with alternates for venting and main engine and ACS feed.

Table E-4. Propulsion Concepts Design Assumptions

Concept No.	1	2	3	4	5	6
Data set No.	112	114	116	115	118	113
Engine model	RL10-7	RL10-8A	RL10-8A	RL10-8B	RL10-8B	New
No. of engines	2	2	1	2	1	2
Suction press.	Hi	Hi	Hi	Low	Low	Low
Nozzle type	Fixed	2 pos	2 pos	2 pos	2 pos	Fixed
Throttle ratio	10	10	10	10	10	10
Mixture ratio	5	5	5	5	5	6
Chamber press (psia)	420	600	600	600	600	1500
(N/cm^2)	389	413	413	413	413	1045
Area ratio	57	300/100	300/100	300/100	300/100	300
Specific impulse (sec)	444	461/450	461/450	461/450	461/450	463
Max thrust (ea) (lbf)	15000	22500	22500	22500	22500	Var.
Engine (N)	66700	10000	10000	10000	10000	
Engine weight (ea) (lbm)	310	425	425	425	425	425 at 30klb thrust
(kg)	141	193	193	193	193	193 at 13.4 N thrust
Propellant fraction at 60000 lbm (27100 kg) propellant	0.907	0.902	0.909	0.921	0.928	0.930

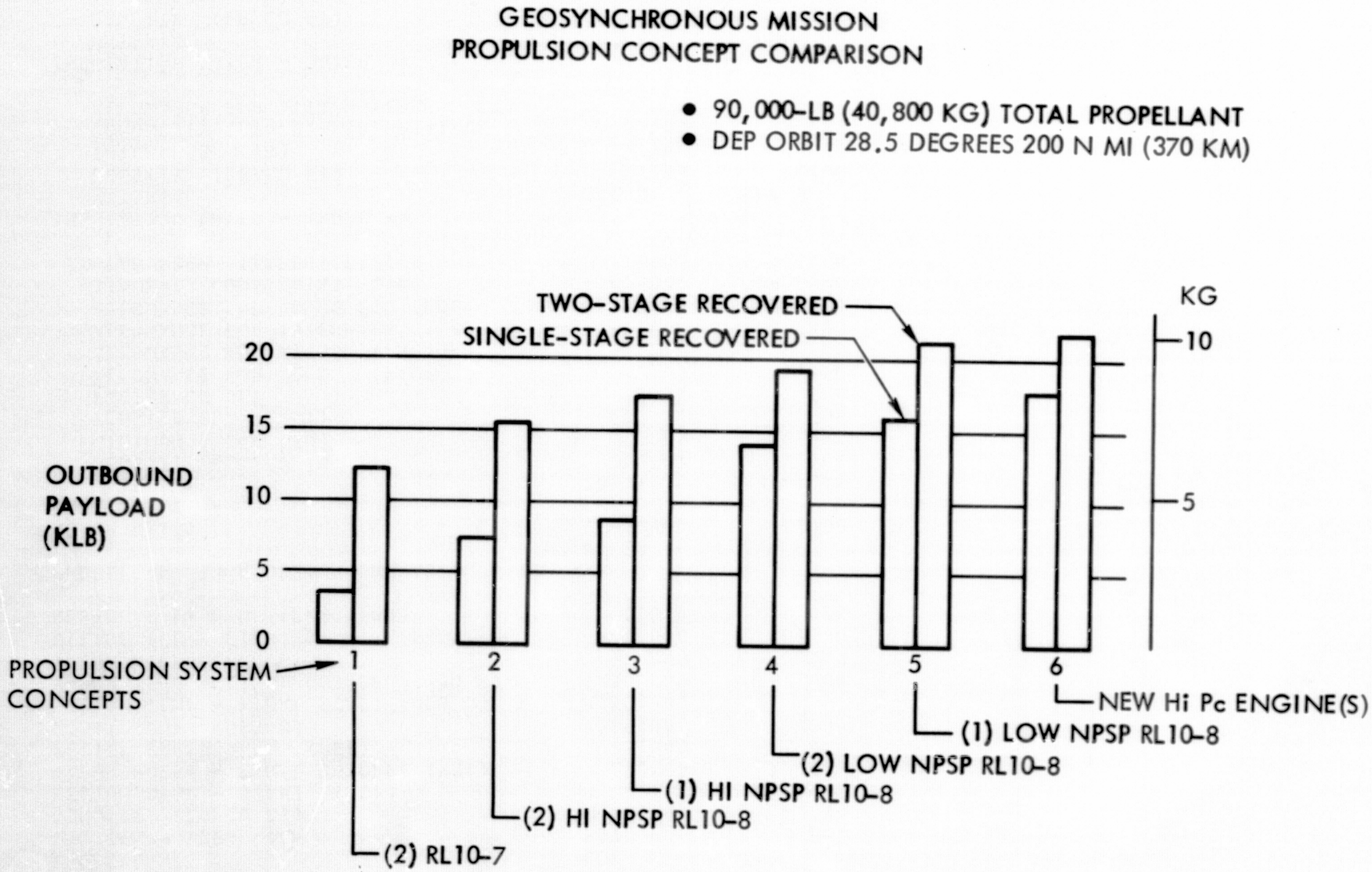


Figure E-2. Geosynchronous Mission Propulsion Concept Comparison

Table E-5. Principal Design Characteristics of Four-Engine Design

***** MAIN PROPULSION SYSTEM *****

SEPT 23,1970 FOUR ENGINE 900 PC

*** INPUT DATA PROPULSION SYSTEM DESIGN CASE NO 133 ***

PROPELLANT TANK CONDITIONS ****

	OXIDIZER	FUEL
PROPELLANT	LOX	LH2
LIQ TEMP (DEG R)	170.000	40.000
DENSITY (LB/CU FT)	71.100	4.240
HELIUM NPSP - START (PSI)	4.000	
VAPOR PRESS - FINAL CUTOFF (PSI)	15.000	12.000
RELIEF PRESS BAND (PSI)	5.000	5.000
LIMIT TEMP (DEG R)	240.000	70.000
HELIUM TEMP - FINAL START (DEG R)	230.000	
VAPOR TEMP - FINAL CUTOFF (DEG R)	230.000	56.000
VAPOR MOLECULAR WEIGHT	32.000	2.020
ULLAGE VOL - FIRST START (FRACTION)	0.060	0.100
RESIDUAL LIQUID (FRACTION)	0.020	0.025

PROPELLANT TANK DESIGN CHARACTERISITCS ****

	OXIDIZER	FUEL
NO OF UNITS	1.000	1.000
TANK SHELL **		
MATERIAL	2014T6AL	2014T6AL
DENSITY (LB/CU IN)	0.101	0.101
MINIMUM GAUGE (IN)	0.045	0.045
TENSILE ULT (KSI)	73.500	85.000
FACTOR ON ULT	1.500	1.500
ALLOWANCE FOR WELDS (FACTOR)	1.100	1.100
ALLOWANCE FOR ATTACH.(FACTOR)	1.200	1.200
ALLOWANCE FOR INTERNALS (LBS)	30.000	30.000
ALLOWANCE FOR DOORS ETC (LBS)	25.000	25.000
DIAMETER CONSTRAINT (IN)	-162.000	-162.000
HEAD FORM (ASPECT RATIO)	-1.400	-1.400
INSULATION LAYER - INNER **		
MATERIAL	GAC 9	GAC 9
DENSITY (LB/CU FT)	2.200	2.200
THICKNESS (IN)	1.500	2.500
CONDUCTIV.(BTU/FT-DEG-HR)*100K	3.200	3.200
ALLOWANCE FOR ATTACH.(LBS/SQ FT)	0.0	0.0
INSULATION LAYER - OUTER **		
MATERIAL	NONE	NONE
DENSITY (LB/CU FT)	0.0	0.0
THICKNESS (IN)	0.0	0.0
CONDUCTIV.(BTU/FT-DEG-HR)*100K	0.0	0.0

Table E-5. Principal Design Characteristics of Four-Engine Design (Cont)

INSULATION COVER **		
MATERIAL	FI GLASS	FI GLASS
DENSITY (LB/CU IN)	0.072	0.072
THICKNESS (IN)	0.025	0.025
GAS VESSEL CONDITIONS ***		
FUNCTION	START HD OXIDIZER	START HD FUEL
GAS	HELIUM	HELIUM
MOLECULAR WEIGHT	4.000	4.000
MAX PRESSURE (PSIA)	3500.000	3500.000
MAX TEMP (DEG R)	400.000	400.000
BLOW DOWN PRESSURE (PSIA)	200.000	200.000
BLOWDOWN TEMP (DEG R)	400.000	400.000
RELIEF PRESS BAND (PSI)	100.000	100.000
LIMIT TEMP (DEG R)	440.000	440.000
VESSEL DESIGN CHARACTERISTICS ***		
FUNCTION	START HD	START HD
NO OF SUBASSEMBLIES	1.000	1.000
VESSEL SHELL **		
MATERIAL	6AL4VATI	6AL4VATI
DENSITY (LB/CU IN)	0.164	0.164
MINIMUM GAUGE (IN)	0.030	0.030
TENSILE ULT (KSI)	145.000	145.000
FACTOR ON ULT	1.500	1.500
ALLOWANCE FOR ATTACH.(FACTOR)	0.100	0.100
INSULATION + COVER (LBS/SQ FT)	0.500	0.500
INTERNAL HEATER (LBS/LB GAS)	0.100	0.100
ENGINE DESIGN CHARACTERISTICS ***		
NO OF UNITS	4.000	
FEED TYPE	PUMPED	
NOZZLE TYPE	FXD BELL	
MANUFACTURER	P+W	
COOLING METHOD	H2 REGEN	
CONTROL ACTUATION	HE 500 L	
PROPELLANT	LOX LH2	
MIXTURE RATIO	6.000	
SPECIFIC IMPULSE (SEC)	463.000	
CHAMBER PRESSURE (PSIA)	1500.000	
EXPANSION AREA RATIO	300.000	
THROTTLING RATIO	1.000	
THRUST (LBS)	0.0	
WEIGHT (LBS)	0.0	

Table E-5. Principal Design Characteristics of Four-Engine Design (Cont)

ENGINE TARE WEIGHT (LB)	28.750
SPECIFIC F/WE (LBF/LBM)	54.000
ENGINE WT COEF. NO 3	-0.020
ENGINE WT COEF. NO 4	1.000
PREVALVE PAIR SPECIFIC WT (LBM/LBF)	0.0
LINE INSULATION + COVER (LB/SQ FT)	0.500
FEED LINE VELOCITY (FPS)	8.000
NPSP START (PSI)	0.0
NPSP RUN (PSI)	0.0
GIMBAL ACTUATORS **	
DESIGN TYPE	HYDRAULC
TVC ACTUATOR TARE WT (LB)	25.000
SPECIFIC F/WE (LBF/LBM)	0.001
THRUST/GIMBAL MOUNT TARE WT (LB)	20.000
THRUST MOUNT SPECIFIC WT (LBM/LBF)	0.002
STRUCTURE SYNTHESIS DATA ****	
STRUCTURE TARE WT (LB)	0.0
EFFECTIVE G LOADING (G'S)	5.000
EFFECTIVE STRESS LEVEL (PSI)	10000.000
COLUMN MATERIAL DENSITY (LB/CU IN)	0.101
EFFECTIVE SKIN THICKNESS (IN)	0.020
EFFECTIVE SKIN DENSITY (LB/CU IN)	0.101
TOP SEPARATION LENGTH (IN)	12.000
INTERTANK SEPARATION (IN)	15.000
TANK TO ENGINE SPACING (IN)	12.000

STOP 00000
M.0072 BEGIN

900 20 40

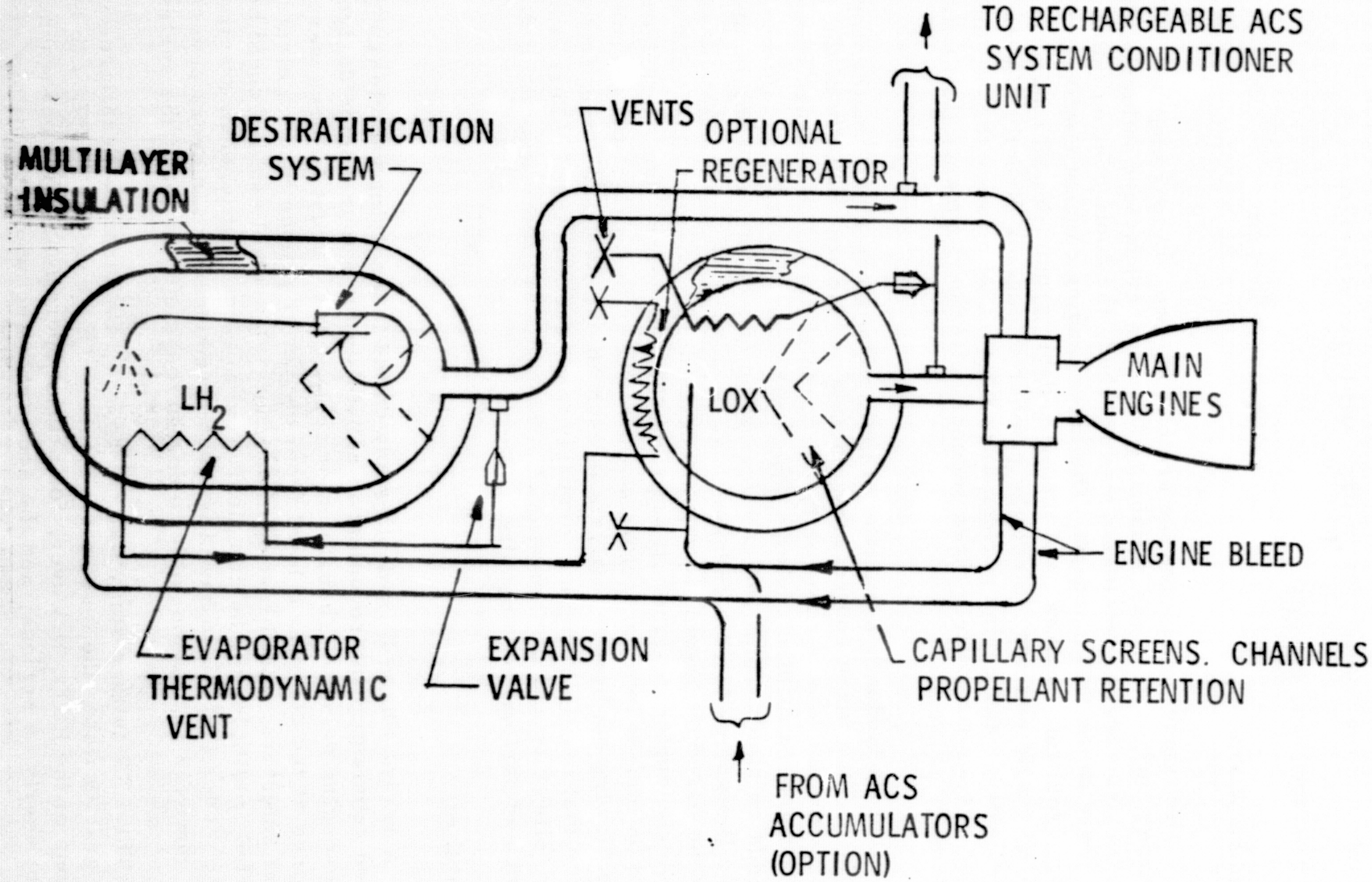


Figure E-3. Main Propulsion System

The thermal isolation concept uses multilayer insulation for both propellants and thermal blocks in lines and to hot structure.

ENGINE SYSTEM STUDIES

Number of Engines

This study considered the use of one, two, or four engines for the RST. Vehicle configuration and reliability (redundancy) factors influence this choice. No general conclusion was reached, the configuration influence varying with the RST concept. The lunar landing concept configurations favored multiple engines, and the expendable ground-based configurations favored single engines. Since the number of engines directly affects engine-thrust rating, and engine performance falls off with lower thrust, the single engine has approximately three seconds better performance than four engines.

The redundancy advantage of multiple engines also remains imperfectly resolved since much controversy exists as to the true validity of the redundancy after increased complexity is accounted for. To measure this correctly requires failure analysis down to the component level, a process that depends on single-point design information of a quality only obtainable in a Phase B study or later. Judgment exercised at this study phase (prephase A) could be based on previous program decisions, e.g., the Apollo SM and LM are single-engine designs. It is preferable, however, that any judgment be based on insight into the factors at work rather than historical precedent. The following qualitative analysis is such an estimate and comes to the conclusion that multiple engines (probably four) are preferred. This is because their reliability and configurational advantages are more important than any weight and performance penalty.

Reliability

A recent Phase B in-depth study on the lunar flying vehicle (LFV) showed substantial improvement in reliability for a four-engine design over single-engine design. Thrust vector control and, thrust failures were examined at the component level from the power source through electronics, gimbal actuators, and thrust chamber/nozzle assemblies. As could be expected, multiple-engine complexity led to identification of interactive failure modes that would have invalidated redundancy. In every severe case, however, it was possible to design these effects out of the system.

The LFV failure study was, of course, concerned with failures in a lunar gravity field, the most severe case. This case also applies to lander versions of the RST. However, for orbital operations, the consequences are not nearly as serious, and the acceptable engine-out conditions are

much broader. For example, in orbit, any degree of thrust misalignment with the vehicle axis is acceptable. A lander has a tolerance of only a few degrees, depending on landing gear capability, and an atmospheric booster has effectively no tolerance. It follows, then, that the potential for reliability gain through use of multiple engines, if substantial for a lunar lander RST, is even greater for an orbital RST.

Weight, Performance and Length

Multiple engines incur greater engine system weight but this is partially offset in some RST configurations by structure savings due to better load-path integration. They also have lower specific impulse at the same expansion area ratio, but since length is at a premium on the RST, the single-engine design can lose this advantage. Multiple engines can be located so that nozzle length does not influence vehicle length. For concepts requiring aft docking gear, the single engine also requires a larger diameter, heavier concentric docking ring to clear the nozzle.

Engine Type

Engine types considered were all pump fed with regenerative/ transpiration-cooled thrust chambers. The only existing LOX-hydrogen engine within the applicable thrust range is the Pratt & Whitney RL10-3-3. Another engine in development and now (pending) redirected toward tug requirements is the Rocketdyne 25,000-pound-thrust LOX-hydrogen Aerospike. In a Phase B planning status is the concept for a new LOX-hydrogen engine for the shuttle orbiter maneuvering system (EOS-OMS). It is currently rated at 10,000 lb_f (44482 N) with a remotely located turbopump integrated with an ACPS conditioning unit. In addition to these engines, which offer potential RST program development cost savings, the study considered uprated versions of the RL-10 and an RST-oriented new engine incorporating demonstrated state-of-the-art advancements.

With the exception of the EOS OMS engine, which is not yet a firm concept, the principal characteristics of all of these engines are listed in Table E-6. The effect of their weight and performance is illustrated in Figure E-4 for the single-stage Concept 1. In this plot, the effects of variations in installation and suction requirements, as well as gravity losses from thrust differences, are neglected. This oversimplification does indicate, however, that the performance advantages of an RST-oriented new engine are substantial, giving a 30 percent reduction in propellant over the existing RL10-3-3. Additional savings would result from lower NPSH requirements that could be designed into the new engine. (Figure E-2 and a following discussion of the MPS pressurization system concern this advantage).

Table E-6. Engine Characteristics (English Units)

	P_c (Psia)		MR	I_{sp}	W_e (lb)	Length	Thrust
PARAMETRIC ENGINES (25000-lb thrust)							
1	2000	100	5 or 6	450	393		
2	2000	200		460	404		
3	2000	400		466	418		
4	1500	100		450	384		
5	1500	200		460	397		
6	1500	300		464	410		
7	1500	400		465	426		
8	1000	100		450	366		
9	1000	200		456	388		
10	1000	300		460	410		
11	1000	400		464	434		
12	500	100		445	386		
13	500	200		454	428		
14	500	300	5 or 6	458	470		
POINT DESIGN ENGINES							
△1 RL10-3-8	600	300	5	462.5	440	150	22500
*△2 RS 23 OMS	800	200	5	461	(132 x 3) 396	62.7	24000
△3 Aerospike	750	110	5.5	457	360	23	25000
△4 AGC Parametric	2500	200	5	466	354	77.5	25000
△5 RL10-3-3	450	57	5	444	325	70	15000

*△2 Assumes use of three RS 23 8000 lbf engines

Table E-6. Engine Characteristics (Metric Units) (Cont)

	P_c Newtons / (cm²)	ϵ	MR	I_{sp} Newtons-sec / kg	W_e (kg)	L	Thrust Newtons
PARAMETRIC ENGINES - 1.11×10^5 NEWTONS							
1	1380	100	5 or 6	4420	178		
2	1380	200	↓	4510	184		
3	1380	400		4576	190		
4	1035	100		4420	174		
5	1035	200		4510	181		
6	1035	300		4550	186		
7	1035	400		4560	194		
8	690	100		4420	166		
9	690	200		4470	176		
10	690	300		4510	186		
11	690	400		4550	197		
12	345	100		4370	175		
13	345	200		4450	195		
14	345	300		4490	214		
POINT DESIGN ENGINES							
△1 RL10-3-8	414	300	5	4530	200	3.80	10^5
△2 RS230MS	552	200	5	4520	(60 x 3) 180	1.59	1.07×10^5

Table E-6. Engine Characteristics (Metric Units) (Cont)

	P_c $\frac{\text{Newtons}}{(\text{cm}^2)}$	ϵ	MR	I_{sp} $\frac{\text{Newtons-sec}}{\text{kg}}$	W_e (kg)	L	Thrust Newtons
△3 Aerospike	519	110	5.5	448	227	0.59	1.11×10^5
△4 AGC Parametric	172	200	5	457	161	1.97	1.11×10^5
△5 RL10-3-3	310	57	5	435	148	1.78	$.666 \times 10^5$

Figure E-4 also shows that the Aerospike has only moderate performance. It does have a length advantage, but, at its proposed mixture ratio of 5.5, its total vehicle length advantage over a two-position bell single-engine design shrinks to 25 inches (63 cm), as shown by Figure E-5. There is also the possibility of incorporating the aft docking drogue in the plug base. However, since its development is far from complete, its evaluation must be on the same basis as for a new bell nozzle engine, for which an ample technology base already exists. On this basis it would appear that the RST is not now a favorable application for the Aerospike, particularly since its most unique feature, an altitude compensating nozzle, is of no advantage in space.

The existing RL10-3-3 has inferior performance, as indicated by Figure E-4. The addition of a larger nozzle extension could improve specific impulse, but the nozzle is then excessively large and heavy (Figure E-6). Increasing the RL10 chamber pressure to its maximum limit of 600 psia (416 N/cm^2), as proposed for the RL10-3-8, reduces this problem somewhat and gives a specific impulse approaching the attainable new-engine value. However, the nozzle is still large, and the modifications required to turbopumps, injector, and thrust chamber entail a development effort approaching that of a new engine. Considering the constraints that always exist when a fixed design is departed from, it does not appear that all of the operational features desired of a new engine could be obtained. Engine cost savings (yet to be established) could be an overwhelming factor, but, in view of the disadvantages described, primarily engine size, the use of any of the RL10 series does not appear desirable for the RST.

As shown by Figure E-4, a new high-chamber-pressure engine has substantial performance advantages. It could also use advanced-technology low NPSH pumps with an attendant reduction in pressurizing system weight. The high chamber pressure has several beneficial effects. At the same engine weight, a higher expansion ratio is obtained. The specific impulse dropoff with increase in mixture ratio is then less than at lower expansion ratios, permitting a higher vehicle optimum mixture ratio as propellant system weight is taken into account. The result is better performance and a shorter vehicle.

Aside from performance gains, a new engine developed specifically for the RST could incorporate currently attainable technology advances to provide operational capabilities that are most valuable to the RST. These are in the areas of long life and reuse; status monitoring provisions and maintainability; operational flexibility (providing fast start response with low chilldown, run-up, and tail-off losses); shutdown impulse repeatability for accurate delta-V cutoff, tank head, or low pumped idle for small delta-V maneuvers; and lunar landing missions (throttling).

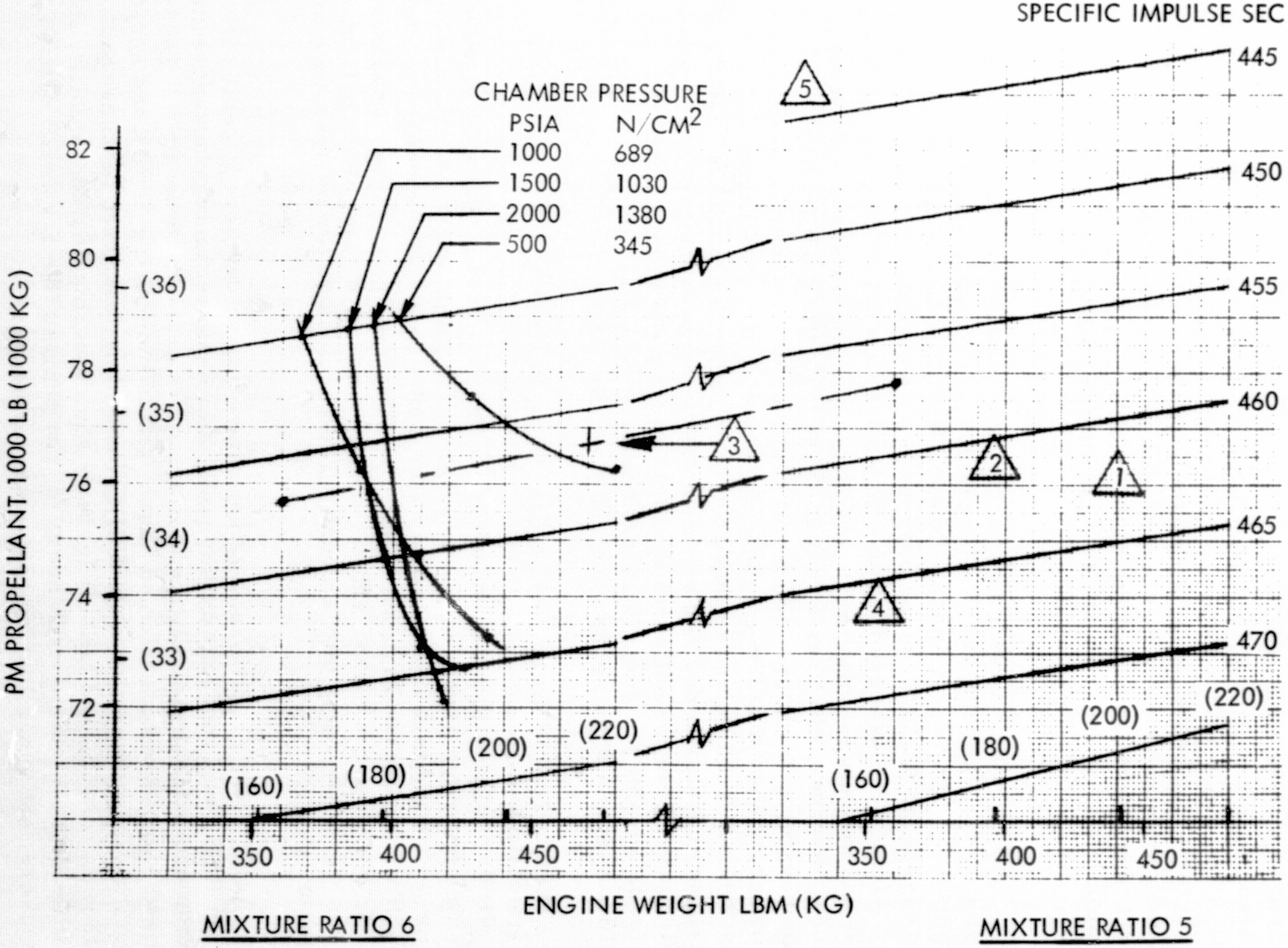


Figure E-4. Single Stage Concept 1 Weight and Performance Effect

Space Division
North American Rockwell

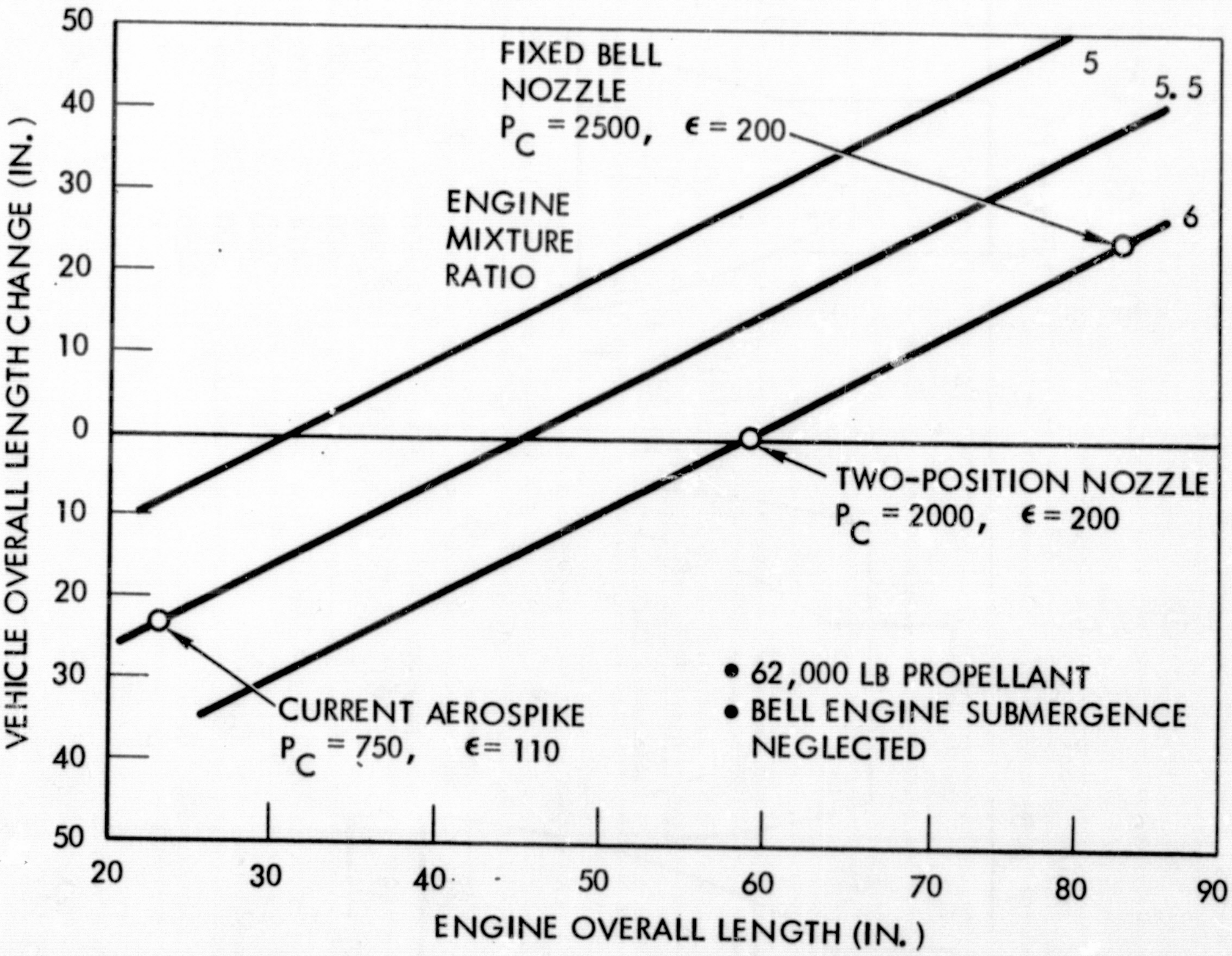
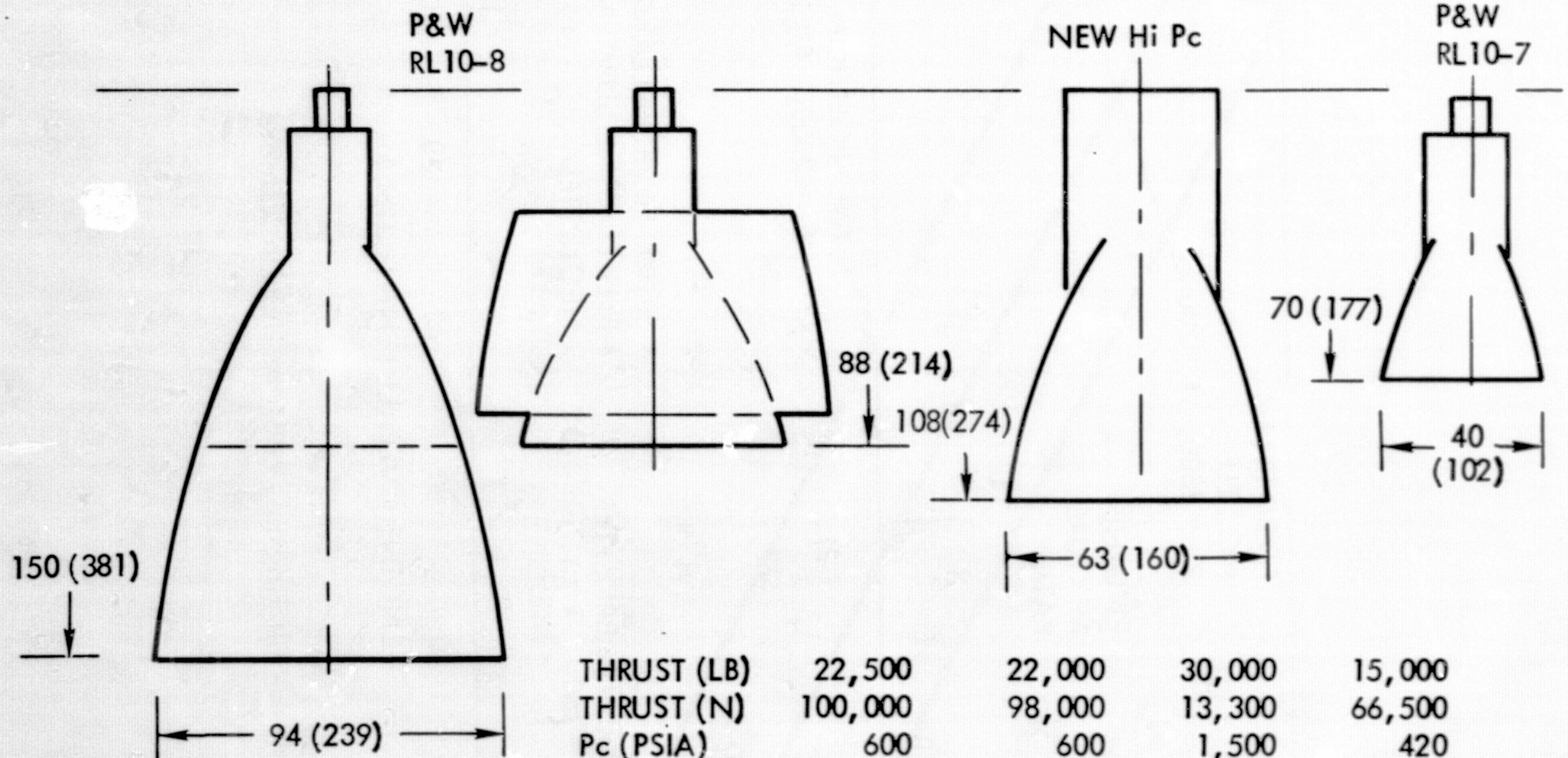


Figure E-5. Engine-Vehicle Length Effects



THRUST (LB)	22,500	22,000	30,000	15,000
THRUST (N)	100,000	98,000	13,300	66,500
Pc (PSIA)	600	600	1,500	420
Pc (N/CM^2)	414	414	900	252
E	300	600	300	57
ISP	462.5	100	466	444
MR	5	450	6	5
		5		

Figure E-6. Engine Concept Dimensions

Thrust Level Selection

The optimum thrust level must be based on the most performance-demanding mission that involves the greatest velocity sensitivity to finite thrust. These conditions occur on the synchronous equatorial mission from an orbit of 100 nautical miles (185 kilometers) by 28.5 degrees, with the departure burn controlling. The optimum is obtained by balancing velocity requirements against engine weight. The optimization criteria selected here is minimum initial gross weight at constant payload. For a four-engine design, the optimization is shown in Figures E-7 and E-8, with the result that a engine thrust rating of 7000 pounds (3180 kilograms) (1250 seconds burn time) is optimum.

Mixture Ratio Selection

The performance at a mixture ratio of six is superior to that at a ratio of five (Figure E-4). As noted, the vehicle is also shorter at the higher mixture ratio. Other studies have shown that the optimum is higher than six. However, this value is used now as an upper limit. Engine manufacturers have indicated increasing development difficulty at higher ratios since a higher mixture ratio reduces the hydrogen available for cooling and, for closed cycles, reduces turbine working fluid with corresponding increase in turbine temperature.

Chamber Pressure and Engine Cycle

As noted, high chamber pressure is advantageous. However, each engine cycle type (cycle refers to the combined process used to extract heat from the thrust chamber jacket and to provide energy for expansion through the turbine) has an applicable pressure range. The closed cycles are most efficient, since all fluid ultimately flows through the chamber. Closed cycles include the expander and augmented expander or staged combustion cycles. Open cycles use a separate gas generator for turbine fluid with a small inefficient contribution to thrust obtained from turbine exhaust either through a separate thruster or through a dump manifold at the low pressure end of the main nozzle. The closed cycles must be balanced, which places an upper limit on their chamber pressure. The precise limits depend on engine manufacturers' preferences with respect to cooling margin, and similar variables, and thrust rating of the engine. Since thrust rating has a strong effect, the RST chamber pressure depends on the number of engines in the configuration.

In a following section, Engine Parametric Model (Figure E-9), the staged combustion cycle is used down to the thrust range where the pressure limit is reduced to the 900 psi limit of the expander cycle. From there down, the more efficient expander cycle is used.

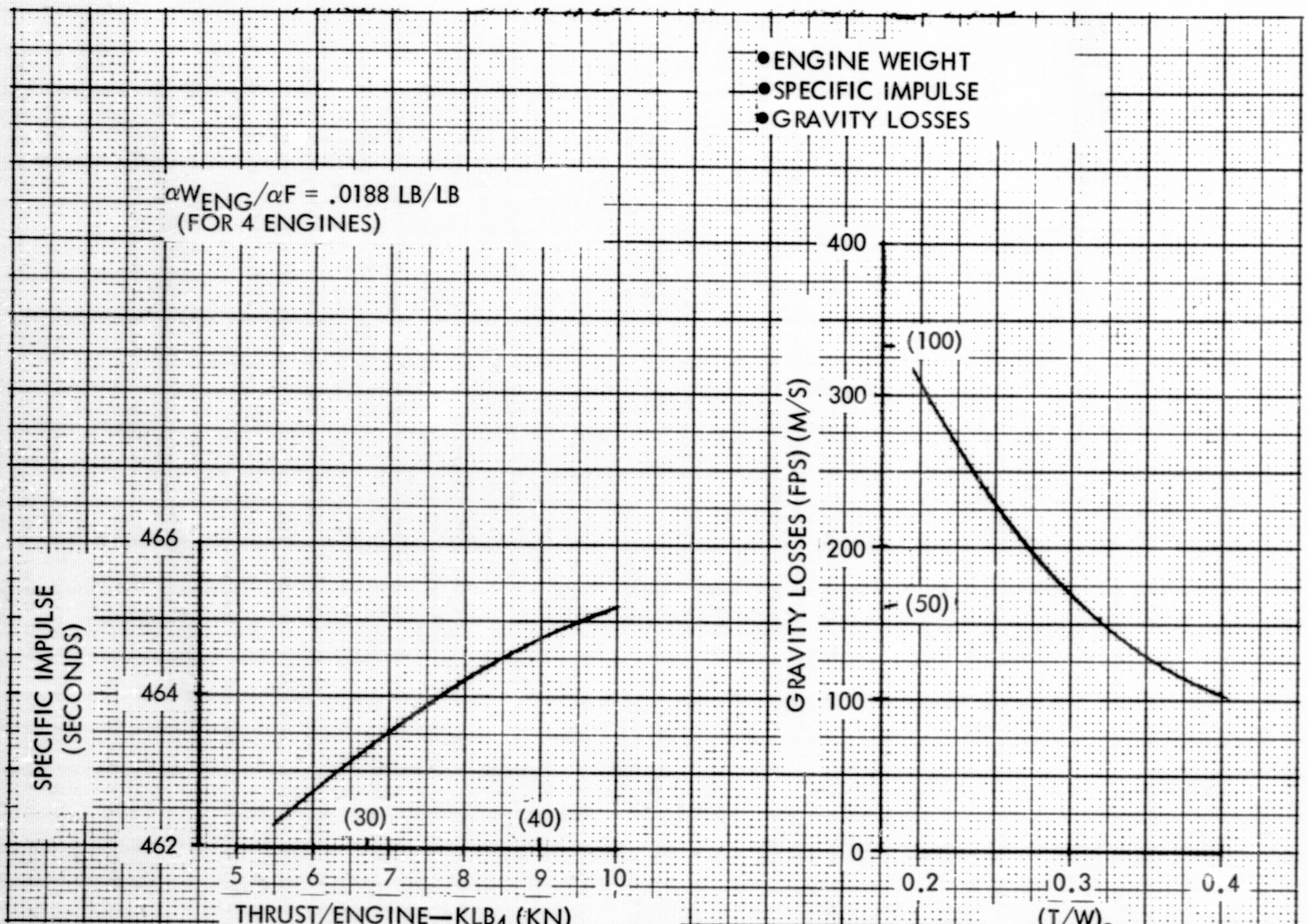


Figure E-7. Thrust Optimization Parameters

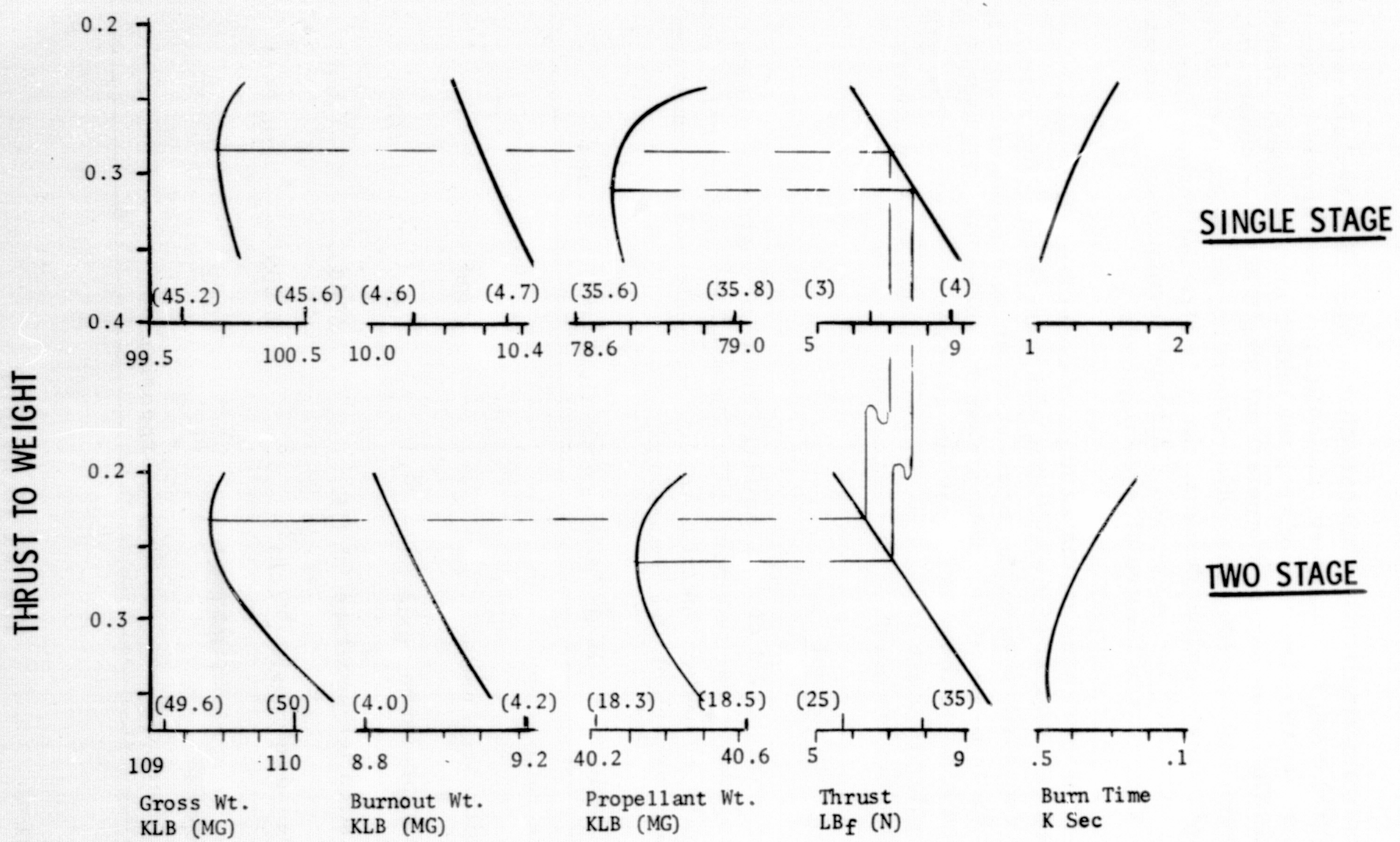


Figure E-8. Thrust Optimums

Expansion Area Ratio

The optimization studies indicate that at high chamber pressures of 1500 psia (1032 N/sq cm) or more, the optimum area ratio is higher than 400 since the specific impulse gain offsets the added nozzle weight. However, configuration limits on nozzle diameter and length act as a constraint, which occurs between 250 and 400 area ratio, depending on the number of engines. For the longer single-engine designs, there is also the option of adding a nozzle skirt extension for the periodic high-performance missions which are, fortunately, expected to involve shorter payloads.

Since expansion area ratios as high as considered here have not been demonstrated, a check of exit conditions was warranted. The result, using equilibrium composition of exhaust products during expansion from Sievers[1] is shown in Table E-7. At chamber pressures above 900 psia (620 N/sq cm), as anticipated for Tug there is no obvious cause for concern. The exit conditions are similar to those of the low-pressure Apollo service module engine and well above the ice point.

Engine Parametric Model

During development of a parametric model of engines applicable to the RST concepts, the disparity among different engine manufacturers' data was resolved to give a uniform specific impulse estimate and one consistent set of scaling data for weight and dimensions.

Table E-7. High Expansion Ratio Exit Conditions

Mixture ratio	6	
Expansion area ratio	400	
Chamber pressure, psia (N/cm^2)	900	(619)
Combustion temperature °R (°K)	6260	(3490)
Exit pressure*, psia (N/cm^2)	0.009	(0.006)
Exit temperature* (°R) (°K)	1010	(561)

*Equilibrium composition during expansion

[1] Reference 2, page E-44.

The methods of McKevitt (Reference 1) and Sievers (Reference 2) were used to establish the basis for the specific impulse values shown in Table E-8. The result is, in most cases, similar to the engine manufacturers' data. The data are plotted in Figure E-10 to give a smooth curve with engine thrust rating while chamber pressure is necessarily discontinuous, as shown in Figure E-9.

Engine weight and dimensions were taken from selected data of engine manufacturers. These are plotted in Figures E-11 and E-12 at an area ratio of 400 and at the chamber pressures assumed (Figure E-9). Variations in area ratio from 400 on dimensions and performance are shown in Figure E-13. Single-point design data with one, two, and four engines are shown in Table E-9.

Table E-8. Specific Impulse Estimate Basis

Engine concept No.	3	2	1A	1B	4
Thrust lbf (M)	6250 (27800)	12500 (55600)	25000 (111200)	24100 (107000)	25000 (111200)
Chamber pressure psia (N/cm^2)	900 (620)	1300 (895)	2000 (1380)	2000 (1380)	750 (516)
Expansion area ratio	400	400	400	100	110
Mixture ratio	6	6	6	6	5.5
Theoretical* specific Impulse (sec)	485	486	486	470	472
Nozzle efficiency (%)	96.4	96.7	96.9	96.2	97.8
Geometry loss	0.3	0.3	0.3	1.5	0.5
Drag loss	2.5	2.3	2.2	1.7	1.0
Nonrecombustion loss	0.8	0.7	0.6	0.6	0.7
Combustion efficiency	99	99	99	99	99
Delivered specific impulse (sec)	463	465	466	448	457
Equilibrium I_s - Frozen I_s - sec	19	18	16	15	16

*Equilibrium composition during expansion

Table E-9. Engine Characteristics Parametric Model

Engine concept No.	1A	1B	2	3	4
Nozzle type	Interchangeable Bell		Fixed Bell	Fixed Bell	PLU
Number of engines per stage	1	1	2	4	1
Thrust lbf (N)	25000 (111200)	24100 (107000)	12500 (55600)	6250 (27800)	25000 (111200)
Chamber pressure psia (N/cm^2)	2000 (1380)	2000 (1380)	1300 (895)	900 (620)	750 (516)
Expansion area ratio	400	100	400	400	110
Mixture ratio	6	6	6	6	5.5
Delivered specific impulse (sec)	466	448	465	463	457
Engine assembly weight lbm (KG)	425	403	250	180	500
Overall length in. (cm)	110 (269)	61 (152)	95 (241)	82 (208)	23 (58)
Nozzle length in. (cm)	83 (210)	34 (86)	73 (185)	62 (157)	- -
Throat diameter in. (cm)	2.8 (7.1)	2.8 (7.1)	2.5 (6.2)	2.1 (5.3)	- -
Exit diameter in. (cm)	56 (147)	28 (71)	49 (125)	42 (107)	59 (150)

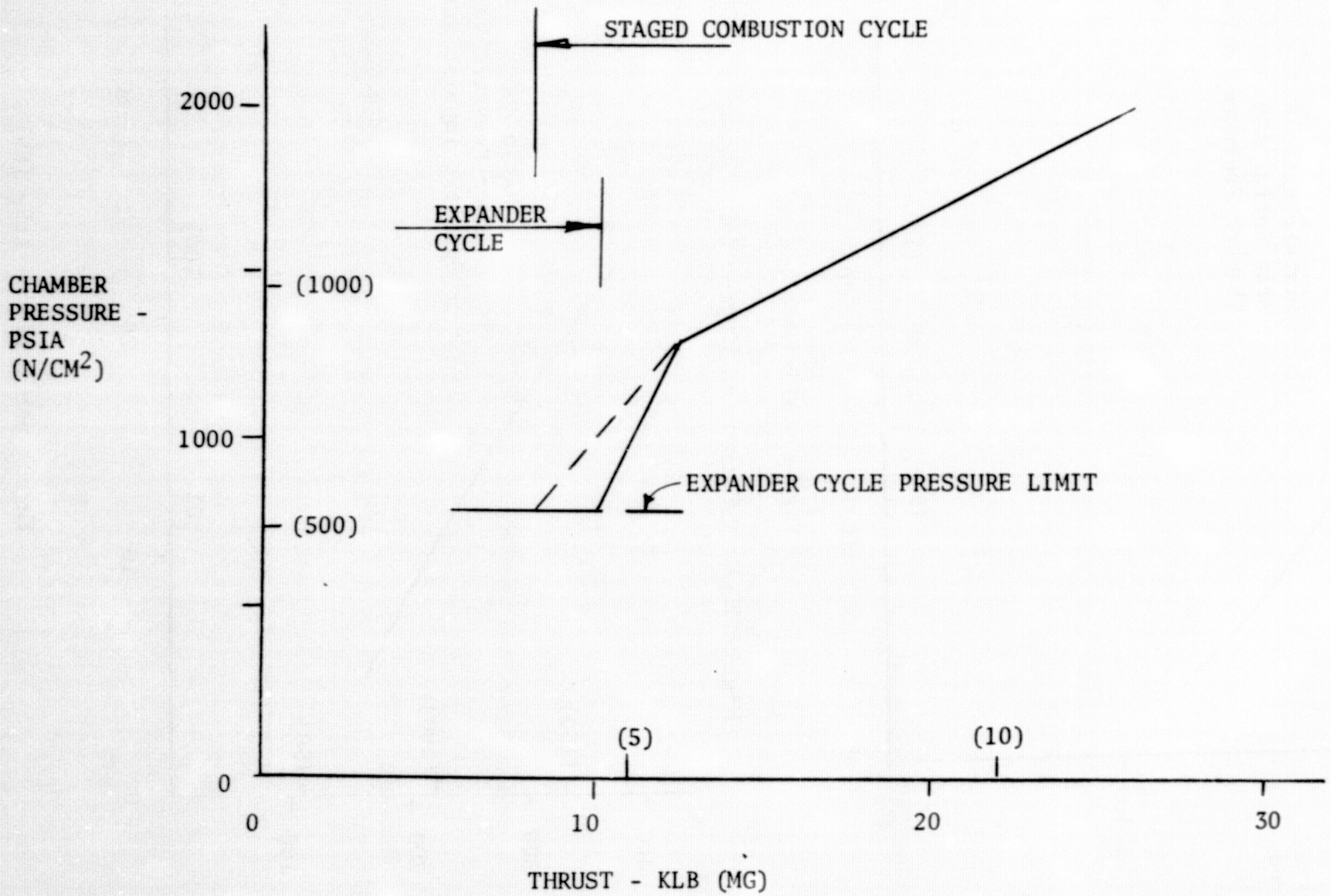


Figure E-9. Chamber Pressure Assumptions

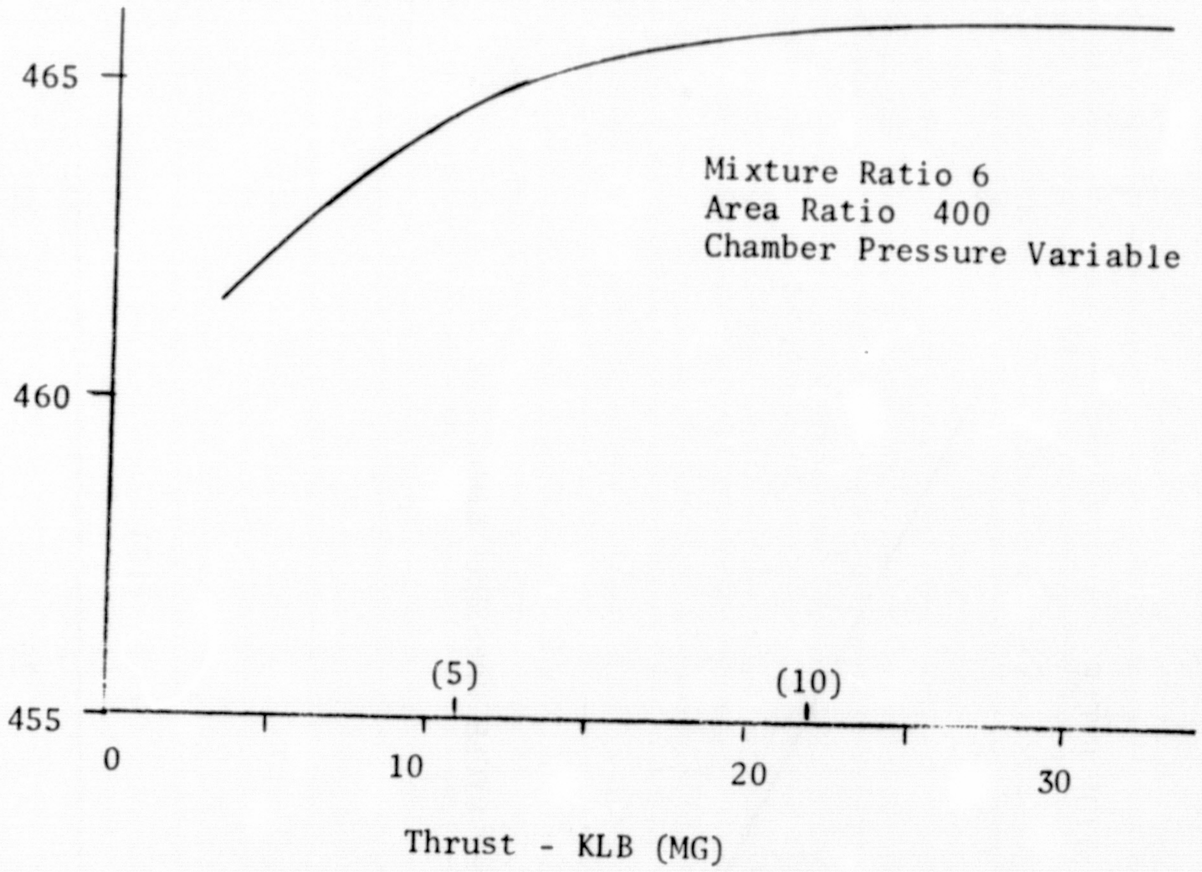


Figure E-10. Estimated Engine Performance

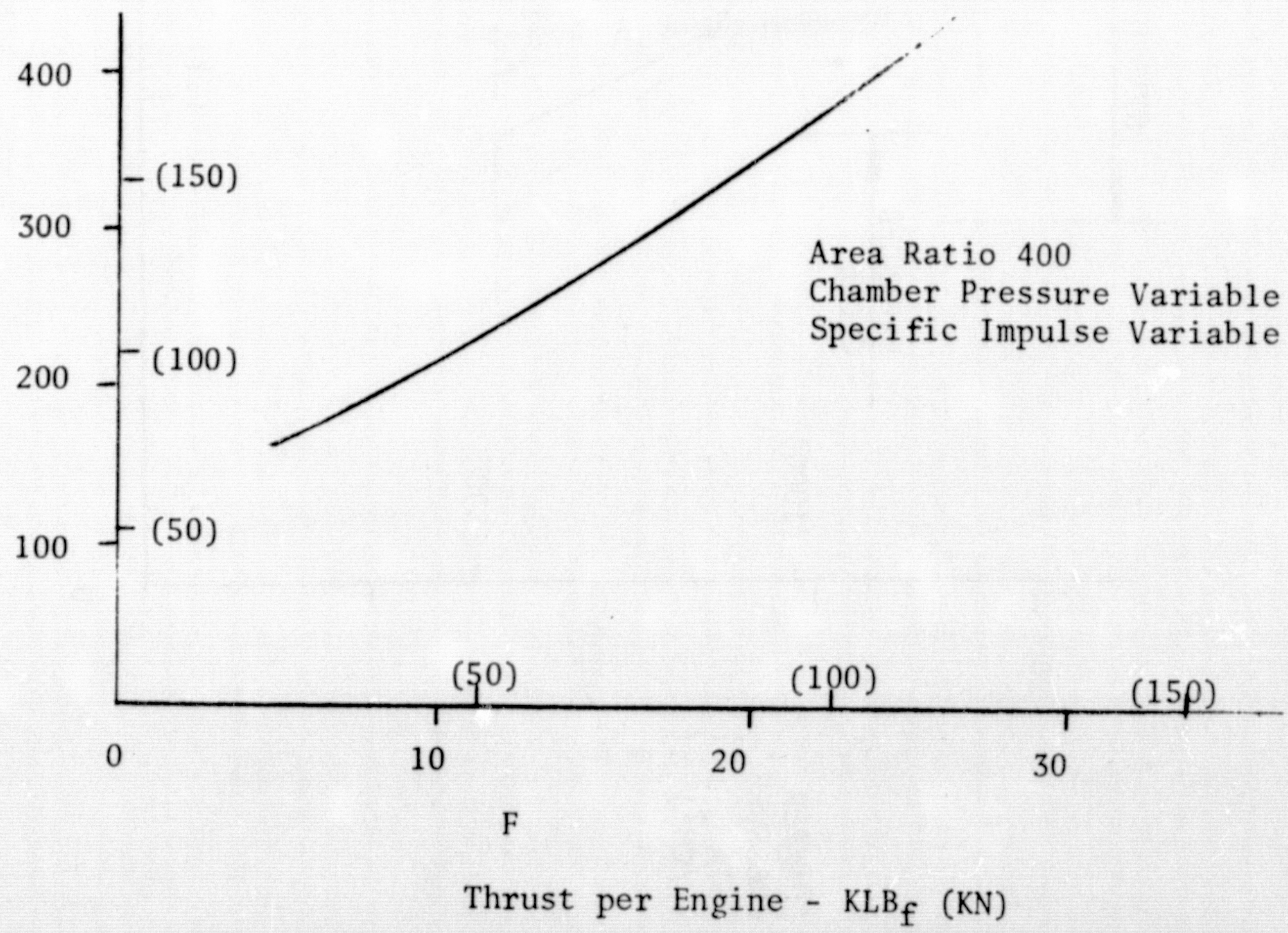


Figure E-11. Main Engine Weights

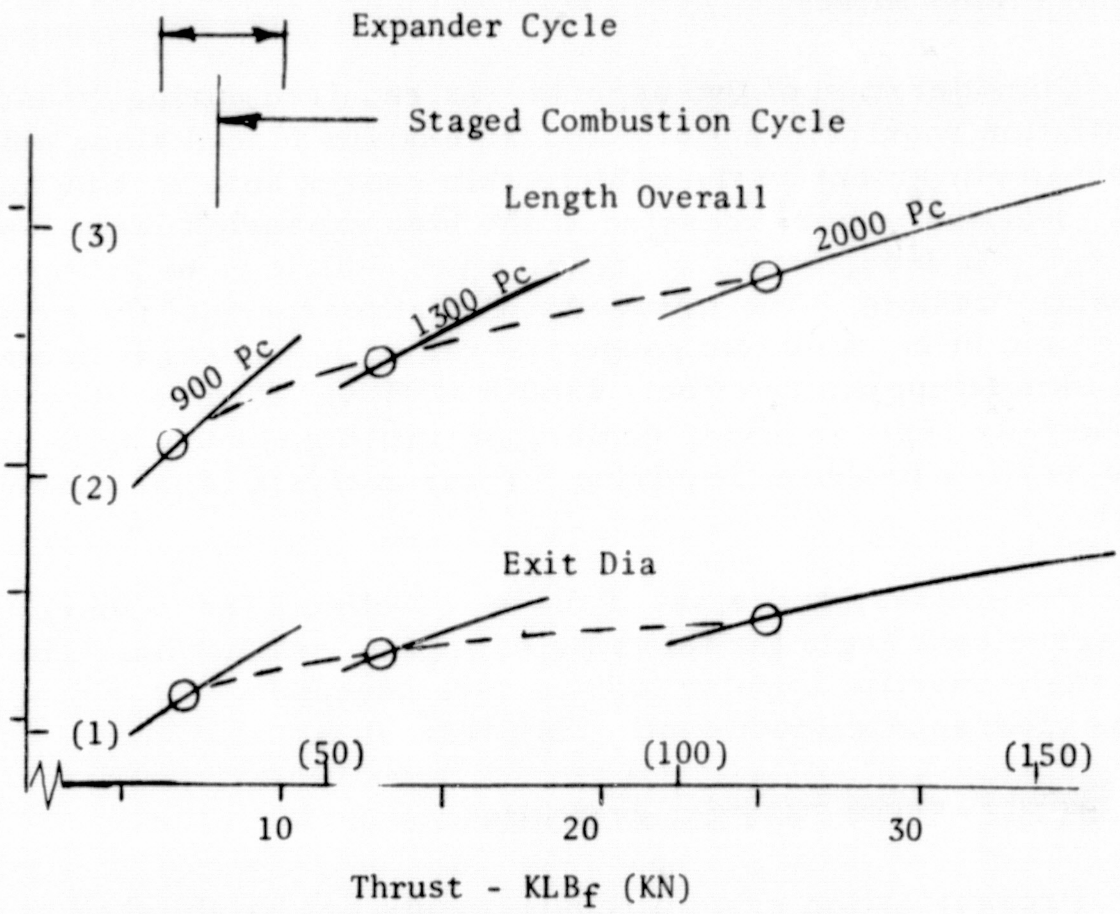


Figure E-12. Engine Scaling Thrust Effect at Area Ratio 400

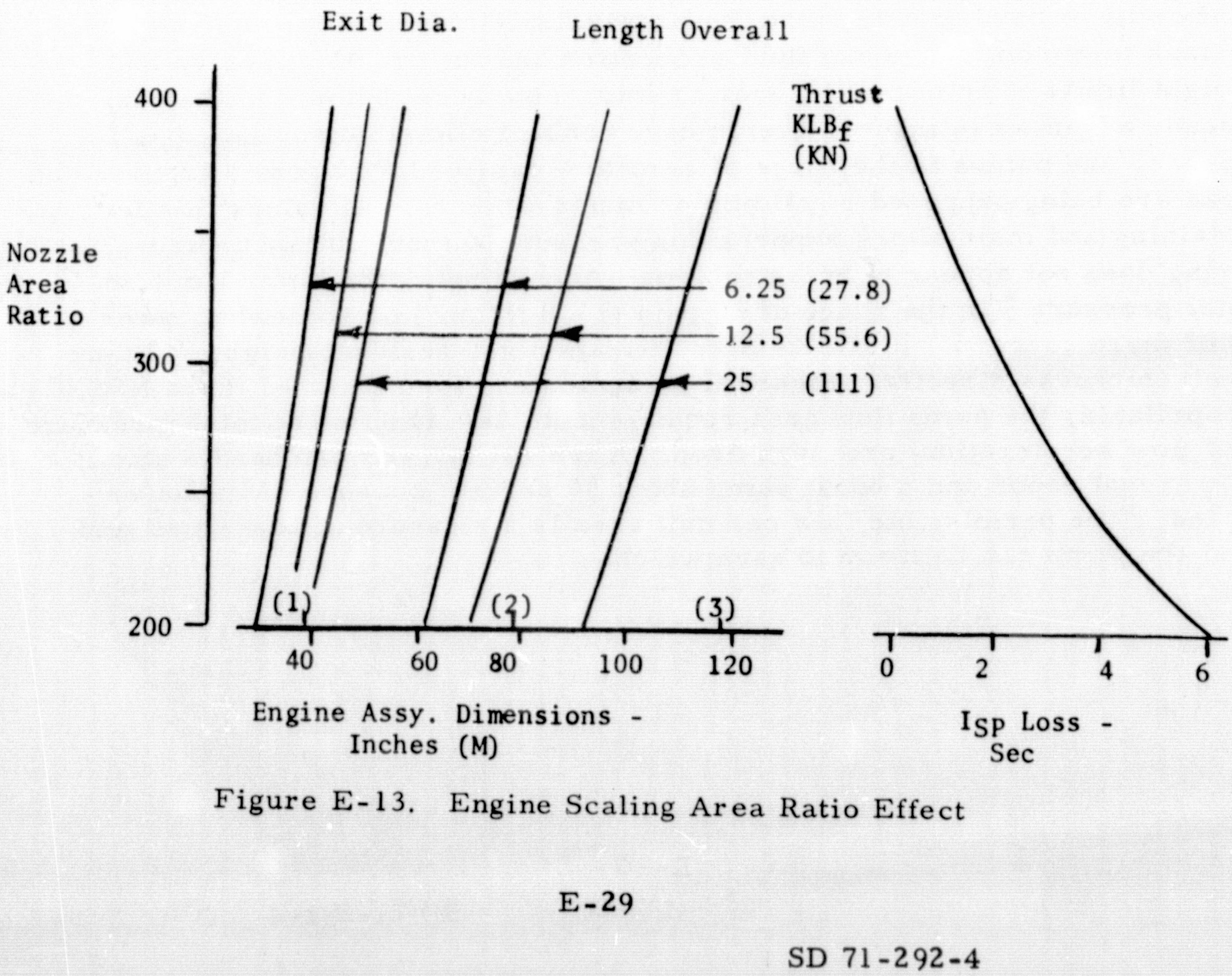


Figure E-13. Engine Scaling Area Ratio Effect

Engine Plume Effect

Consideration of two aspects was required during development of the installation requirements of outboard engines placed along side the vehicle. Plume impingement on the vehicle skin causes both heating and jet momentum loss. Heating is not excessive at the high expansion ratios and geometry involved. It does, however, require the vehicle to be protected with a shield, probably titanium. The infrared (IR) transparency of the exhaust will permit this shield to be minimum gauge and radiation cooling indicates an equilibrium skin temperature of only 1360 R (755 K). The nozzles most probably will be fully regeneratively cooled, permitting shadowing by the vehicle. A more serious problem requiring further analysis is the momentum (I_{sp}) loss effect.

Preliminary appraisal of these two effects led to selection of a 10-degree cant angle for this type of engine installation. The corresponding cant angle specific impulse loss is seven seconds. Providing nozzle and plume clearance without canting is preferable.

PRESSURIZATION SYSTEM STUDIES

Pressurization system requirements are established by engine suction limits on both NPSH (net positive suppression head) and total pressure. For a new engine, the inverse of this matching will be obtained by trade studies. They will be used to determine the pressurization system lower limit due to tank minimum gauge and fluid properties and the practical design and weight limits of an engine inducer or integrated boost pump. In the past decade advances in inducer design have doubled allowable suction specific speeds, and pumps in the range of zero to 4 psi (0 -2.76 N/cm^2) suppression head are being proposed by all engine manufacturers. At values this low, obtaining and maintaining suppression pressure without collapse during firing does not appear to be a problem. Accordingly, the lower limit on total pressure — in the range of 20 psia (13.8 N/cm^2) or somewhat lower — is of more concern. It affects pressurization gas residual weight and tank shell thickness directly. Figure E-14 indicates, for saturated (zero NPSH) propellants, the pump flow area requirements as a function of total (pressure and flow acceleration) pressure drop. A typical inducer can handle about 30 percent vapor and a boost pump about 50 percent because of its lower speed. The permissible flow per unit area is a measure of how small and fast the pump can be made to save weight.

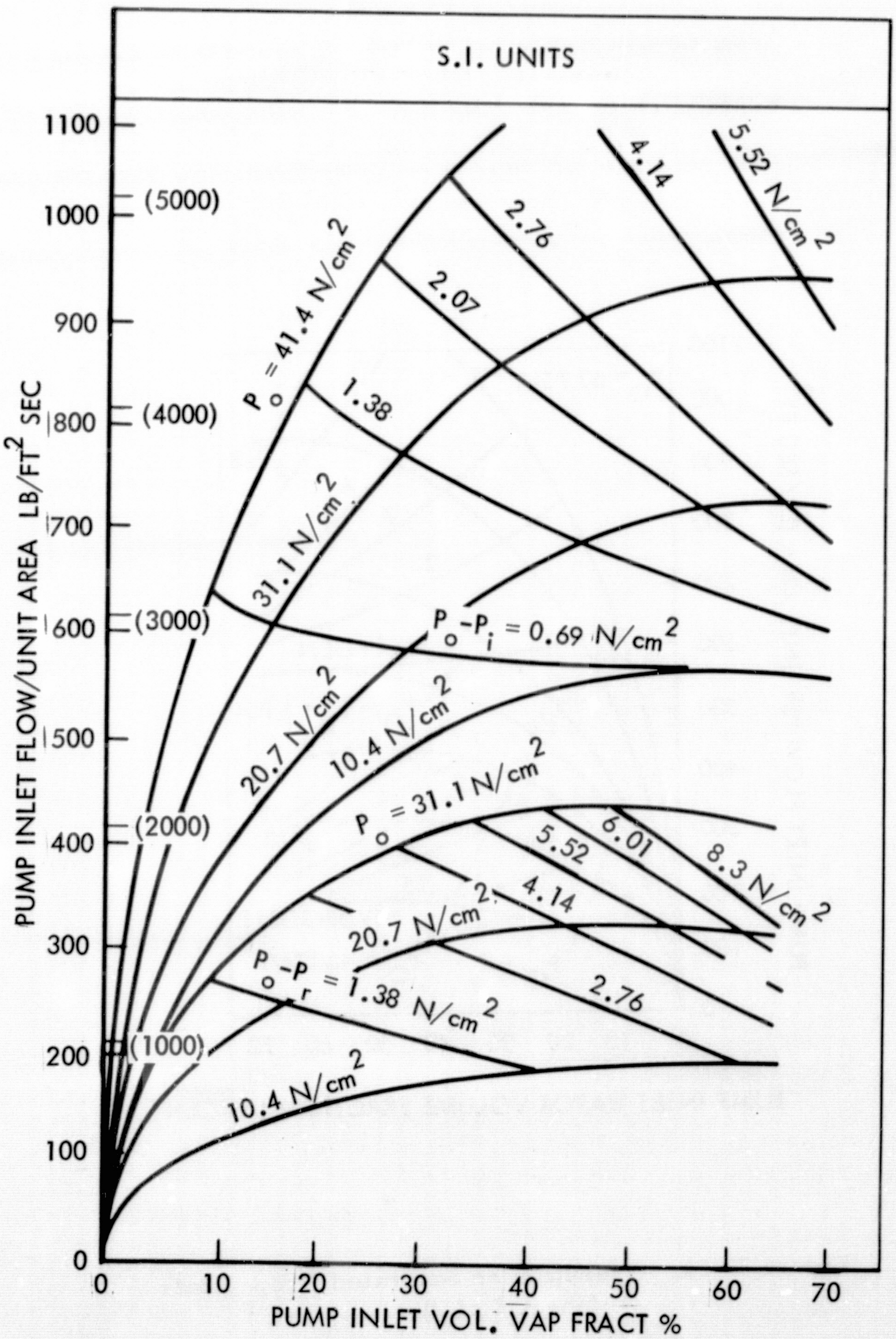


Figure E-14. Operation on Saturated Propellants
(Sheet 1 of 2)

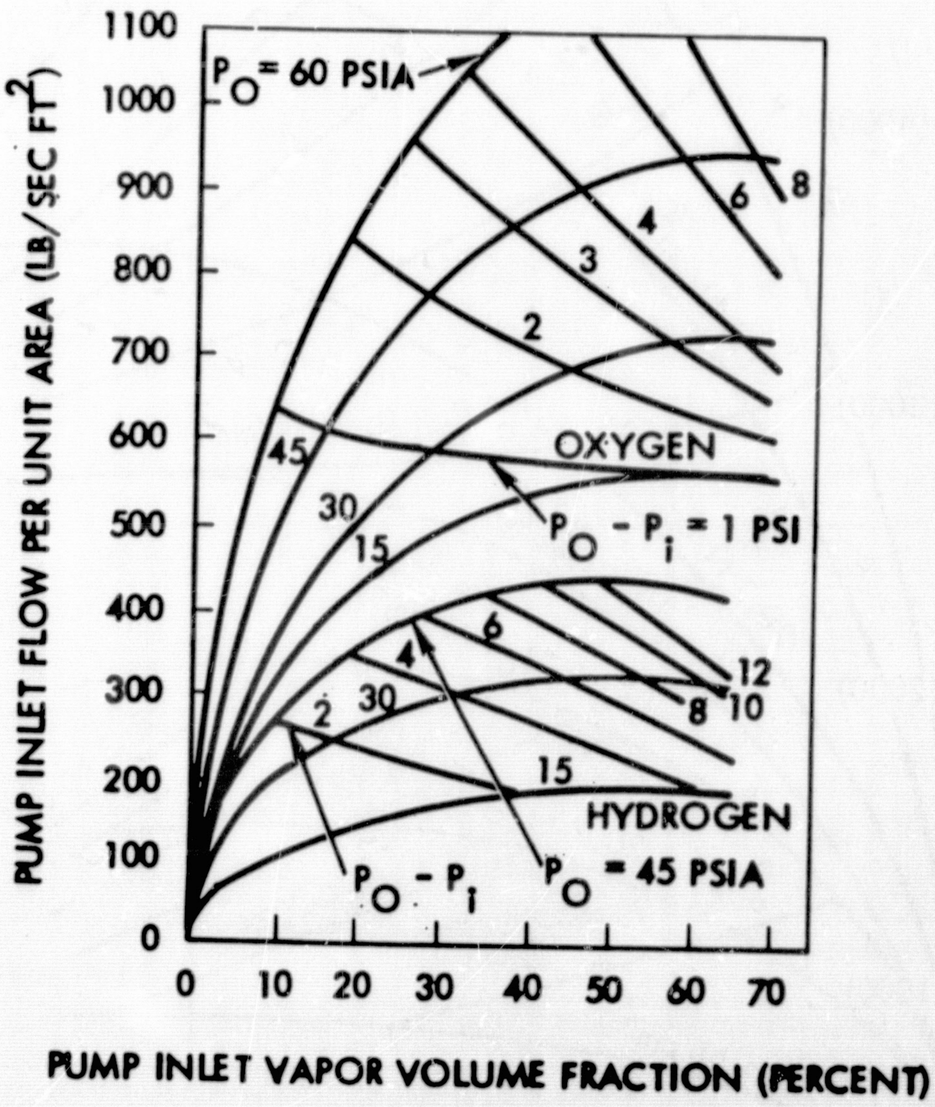


Figure E-14. Operation on Saturated Propellants
(Sheet 2 of 2)

Since the structural weight is sensitive to pressure,* this study has assumed a maximum tank operating pressure of 25 psia (17.2 N/cm^2) (vent valve operation) and operating pressure of 20 psia (13.8 N/cm^2). This assumption is based on the cryogenic fluid being maintained at a 15 psia (10.4 N/cm^2) saturation temperature (37 R (70.4 K) for H2, 162 R (90 K) for 02) continuously. It is also assumed that the device that maintains the fluid at 15 psia (10.4 N/cm^2) will be the thermodynamic venting system, which is discussed subsequently.

The main concern in the pressurization system is that the heat exchangers, tubes, and inflow regulators provide the force to prevent the pumps from cavitating. Similarly the weight of the residual gases is a dominant system performance parameter and one desirable to control. Since the tug is a multiburn vehicle, there is time for the pressurization gas to cool and assume a higher density, at least for the early burns. The last burn, however, is most influential if it can end with heated ullage gas.

The problem associated with adopting thermodynamic venting for boiloff conversion is that it operates better with the provision for single common propellants and pressurization gases (i. e., pure fluids). There have been many studies proving that helium is the lightest pressurant for oxygen propellant tanks, but, for the space based tug, it is far better to use the respective propellants as pressurant fluids than to add logistics complexity and the inherent unreliability of a foreign pressurization gas. These considerations would not be so dominant for a ground based tug, but the qualitative considerations still clearly favor the use of the same propellants and pressurant gases to simplify the zero-g storage and venting system. The need to permit condensation of pressurant on the colder surfaces and thus promote the ability to pump liquid in the near-zero-g environment is one of these qualitive considerations. The others are reliability implications, compatibility with the refueling system to be used, and use for auxiliary propellant functions.

Stratification is associated with pressurization systems as usually providing a natural ullage pressure greater than the vapor pressure of the fluid at the tank outlet. The thermal gradient in the fluid is then normal to the liquid surface in the gravitation or thrust fields since the warmer layers of fluid are lower in density and bouyant. However, under zero g, stratification is not dependable, so different methods must be devised to achieve the same purpose. If the thermodynamic venting device were located near the

*The use of separate boost pumps also can be considered as a way to achieve reduced pressure, but the qualitative feature of added unreliability in series with the engine has caused it to be eliminated from the following discussion.

propellant outlet, it would be possible to develop the thermal gradient by layer-to-layer conduction, much as if the fluid were a nonmoving solid substance. The opposite extreme is where the fluid is circulated and in which it is completely destratified at the time of engine start. The former case would be least severe, since the formation of vapor from the hot layers of fluid would help initiate pumping and prevent cavitation. The latter case is cause for concern. In the event of complete destratification, the system would require an accumulator or start tank to provide an initial pressure on which to start the engines and allow them to run for a short time while the engine-bleed pressurizing system comes up to operating temperature and pressure. While there is no specific engine yet available, typical pressurant supplies are derived from a direct bleed of oxidizer from the turbopump, with subsequent heating in a heat exchanger on the turbopump exhaust, and direct bleed of fuel from the thrust chamber or engine jacket cooling system. Since it requires a few seconds for these systems to come up to operating conditions, an accumulator will be required. The accumulator will be designed to provide pressurant for about five seconds, with a second restart capability. Furthermore, the accumulator will be recharged by the engine-bleed system once it is in operation.

For greater realism in estimating residuals and for better understanding of the implications of a multiburn mission on the pressurization system, a typical pressurization study was performed for the space tug. The mission assumed was a signle stage to synchronous equatorial orbit. The fuel side pressurant mass flow rate and total pressurant mass requirements were determined for this six-burn mission. A pressurization system computer program (No. 6N-922) was used in these computations. It considered thermal stratification in the tank as well as the details of interphase mass and heat transfer. The analysis assumed the initially stratified condition of 25 psia (17.2 N/cm^2) tank pressure at the start of burn with no pressurant flow rate until the pressure dropped to 20 psia (13.8 N/cm^2). The pressurization system is then used to maintain the tank pressure at 20 psia (13.8 N/cm^2) for the remainder of the burn. The assumptions and mission time sequences are shown in Table E-10.

The total accumulated pressurant consumed is shown in Figure E-15. This figure shows that the ideal stratified condition would require only a modest amount of hydrogen pressurant gas, a little over 21 pounds (9.5 kg). Furthermore, the later burns in the mission may operate on stratified fluid for over 100 seconds before the pressurizing gas begins to flow in. The first burn actually consumed the greatest amount of pressurant, and the second, fifth, and sixth burns are performed without any pressurant at all. Figure E-16 shows the mission time line, with the time duration spelled out between successive burns. This has been used to estimate the

Table E-10. Single Stage to Synchronous Orbit and Stage Return

Assumptions:

1. Initial ullage pressure of 25 psia (17.2 N/cm^2) for each burn
2. Initial ullage temperature corresponds to the saturation temperature at 25 psia (17.2 N/cm^2)
3. Pressurant flow begins when the ullage pressure drops to 20 psia (13.8 N/cm^2)
4. Pressurant temperature of 230 R (128 K)
5. Neglects start and shutdown transients
6. Total usable propellant mass of 62,258 pounds (28,400 kilograms)
7. Engine mixture ratio of 6:1

Results:

Event	Time (hrs)	Burn Time (sec)	Pressurant Flow lb/sec	Pressurant Used lb (kg)	Accumulated Pressurant lb(kg)
Docked EOS phase orbit injection	3.627		0.0337 (.0153 Kg/sec)	15.66 (7.1)	15.66 (7.1)
Trans orbit injection	13.016	13	0	0	15.66 (7.1)
Synch orbit injection	18.22	236	0.0295 (.0134 Kg/sec)	3.26 (1.48)	18.92 (8.6)
Trans orbit injecting return	29.805	203	0.207 (.0094 Kg/sec)	2.11 (0.96)	21.03 (9.65)
Phase orbit injection	35.235	127	0	0	21.03 (9.65)
Circular orbit injection	36.445	49	0	0	21.03 (9.65)

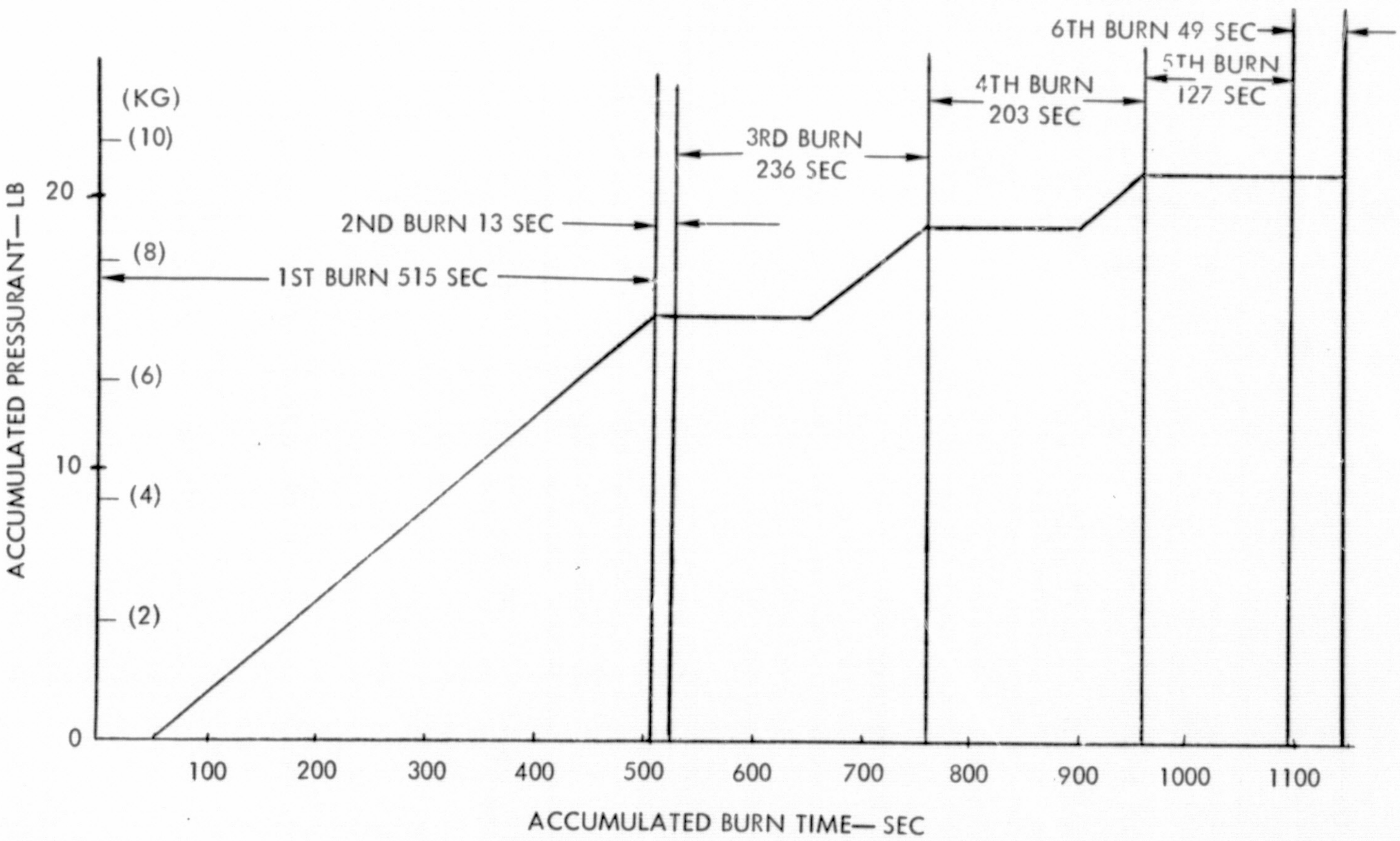


Figure E-15. Pressurant Versus Burn Time

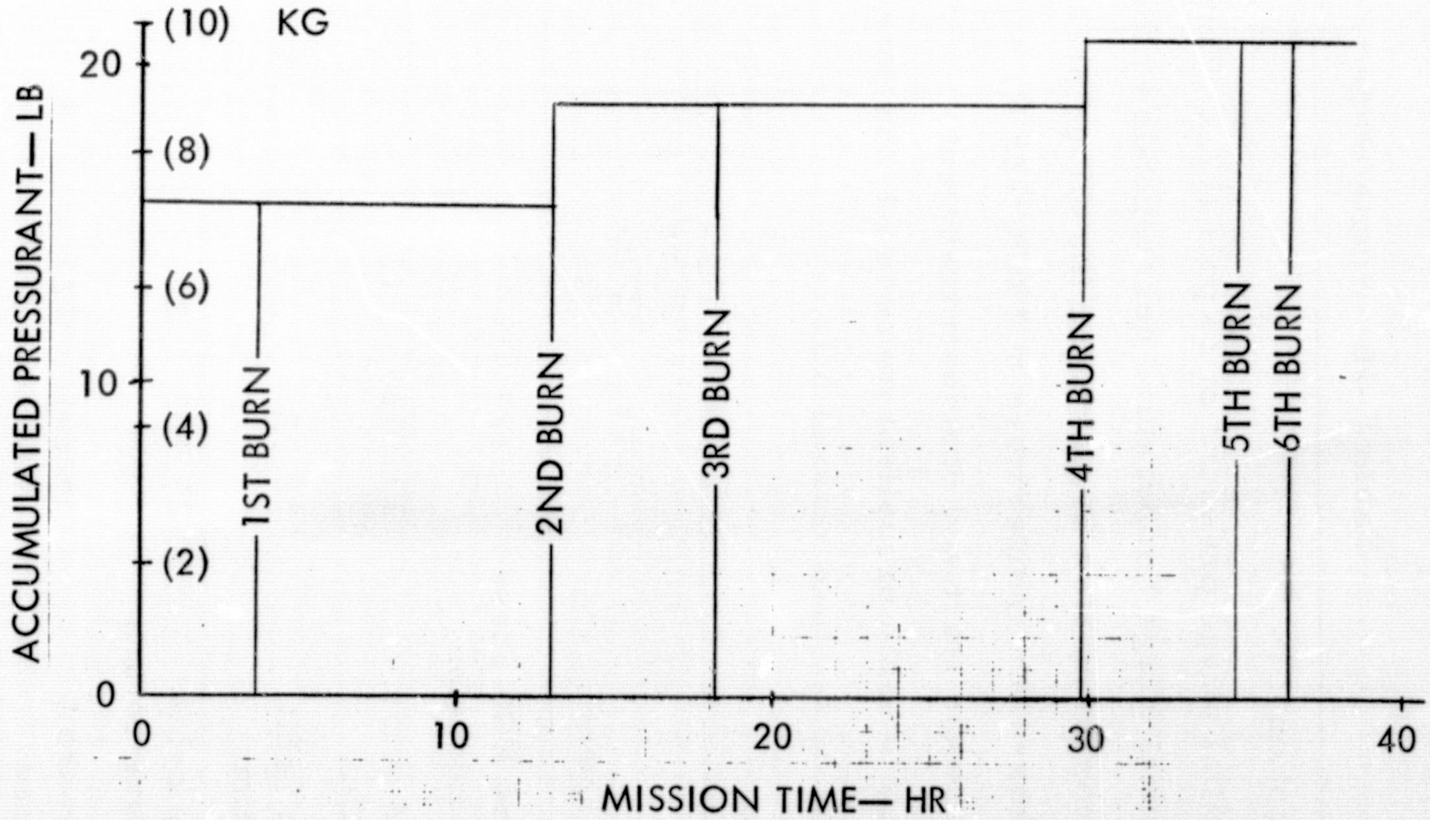


Figure E-16. Pressurant Versus Mission Time

reasonability of the 25-psia (17.2-N/cm^2) assumption at the initiation of burn. Longer mission times would provide a more startling stratification, but lunar lander modules would be less likely to benefit from this because of gravity. Thus, a practical pressurization system will still require much more consideration and greater experimental confirmation before either the stratified or destratified approaches can be deemed valid.

REFUELING OF THE TUG

Two basic modes of refueling the tug have been compared: They are ground (based) fill and orbital. The former method has the advantages of developed technology and procedures, easier checkout, small ullage volume, and accurate quantity gauging. The latter method (orbital refueling) provides greater operational flexibility, allows multiple tug missions, and permits a dry launch of the tug. However, the technical problems inherent in orbital refueling, including liquid location control, transfer and chilldown losses, and in-orbit quantity gauging, must be overcome. The two methods of refueling each can be accomplished in a variety of ways. Ground fill can use fill, drain, topping, purge, and vent plumbing, which is either separate from or integrated with the corresponding system for the earth to orbit shuttle (EOS). Orbit refueling* can be by direct transfer from the space shuttle and a transfer that uses two types of depots as intermediaries. Direct transfer avoids the additional complexity and losses connected with a depot, while use of a depot allows greater operational flexibility. One of the depot concepts uses INT-21 launched tankage and required fluid transfer both to and from the depot with a storage requirement dependent on the traffic plan. The other depot employs space shuttle payload propellant tankage placed in orbit by a modular exchange method. A study of propellant transfer to the reusable nuclear stage (RNS), under contract to NASA/MSFC, has led to the recommendation that transfer to the RNS be conducted via a depot. However, this recommendation was to a large extent based on the large quantity of propellant required by the RNS (300K pounds or 136 Mg) and safety considerations associated with a nuclear stage.

To refuel in orbit, it is necessary to dock, connect lines, supply propellant (rather than vapor) to the source tank outlet region, expel the propellant, maintain receiver tank pressure below supply tank level, gauge quantity, and control vehicle attitude and attitude rate within allowable limits. Other refueling methods are given in Table E-11. The important first phase for in-orbit refueling is source tank propellant positioning and acquisition. Of the methods identified in Table E-11, studies to date have

*Some of the orbital refueling alternatives are shown in Figure E-17. The methods shown are not independent alternatives but rather an interdependent set.

Table E-11. Orbital Refueling Alternatives

Source Tank		Tug Propulsion System Tankage	
Propellant Acquisition and Location Control	Expulsion	Hydrodynamic Control	Thermodynamic Control

Capillary devices
• Full tank screen link
• Screen compartment
• Capillary pumps or channels
• Combinations of above
Acceleration
• Linear thrust
• Rotation (pitch axis)
• Spin (roll axis)
Electrohydrodynamic
Bladders, bellows and diaphragms
	Pump
	Pressure
	• Stored helium
	• Self or thermal pressurization
	  • internal
	  • external
	  • liquid/gas conversion
	Pump plus pressure
		Capillary devices
		Acceleration
		Positive (bladder, etc.)
		Baffles
			Initial chilldown of receiver
			Vent during filling
			• Thermodynamic venting
			• Direct venting with position control
			No vent filling via spray fill and ullage collapse
			Vent to space prior to filling

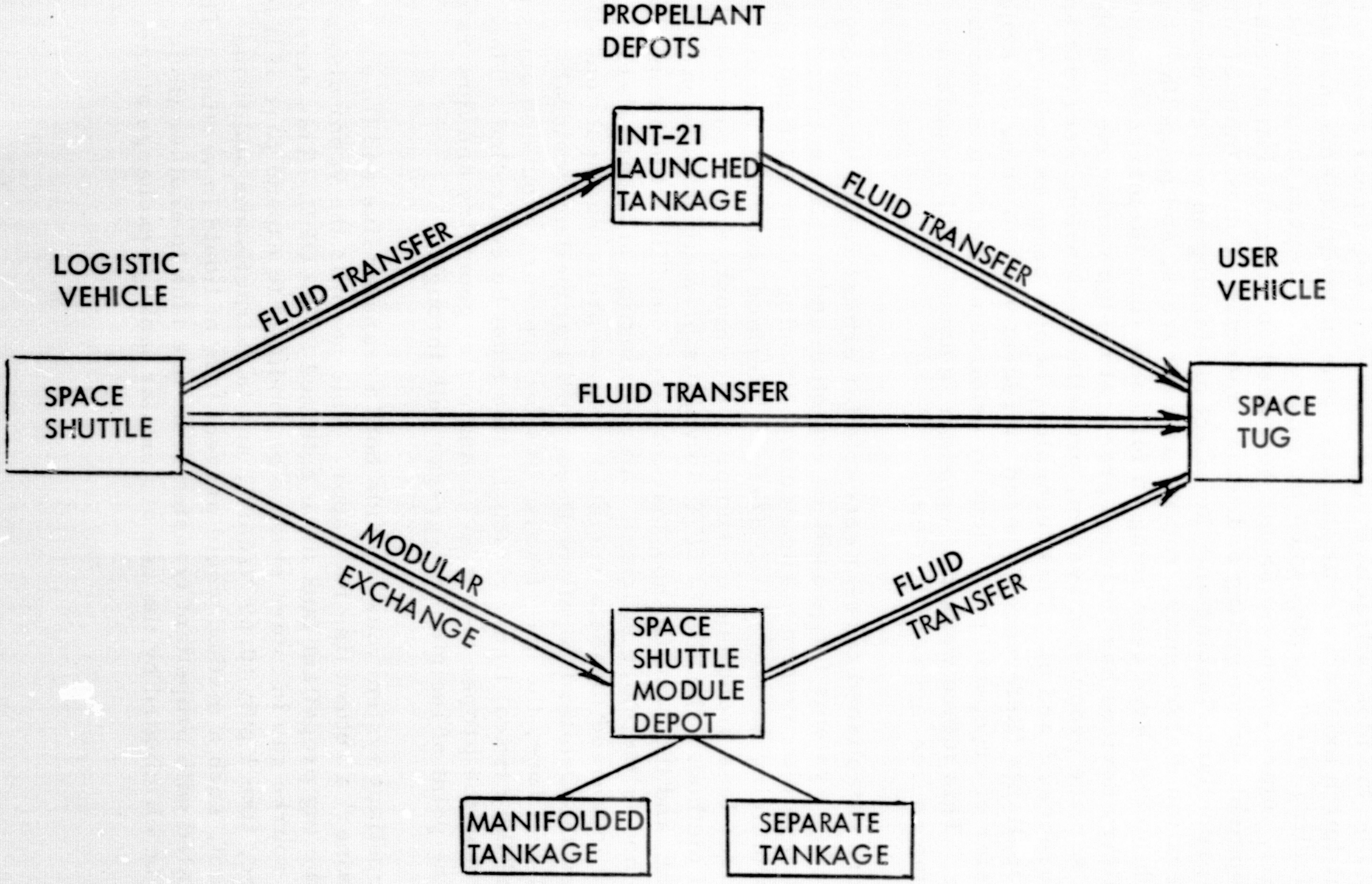


Figure E-17. Logistic Options

shown three to be the most attractive: capillary devices, linear acceleration, and rotational (pitch axis) acceleration. As some of the problems involved in orbital refueling of cryogenics must be regarded as beyond the present state of the art, future emphasis should be placed on using the latest technology developments in conducting additional tradeoffs. An example of this is studies of propellant sloshing and the effect of sloshing on vehicle dynamics during acceleration disturbances, rotational maneuvers, and propellant transfer, for which studies have been going on for some time (References 3 and 4). Work on capillary devices has included practical design work done in support of the Apollo service propulsion system, propellant retension reservoir and screens (References 5 and 6). This technology is now being extended to cryogenic propellants (Reference 7) based on recent work to study the acquisition and transfer of cryogenic propellant by using capillary devices for large scale tankage. This future work will consider in about equal proportion problems of the source and receiver tanks.

The more established current concepts for orbital refueling embody both thrusting "g" systems for propellant retention and a self-contained pumped-fluid expulsion system. The liquid positioning using artificial "g" is the most promising method of liquid/vapor interface control. The use of rotation requires less propellant consumption for thrust, and is currently the preferred method. The choice of pump versus pressure expulsion has been evaluated, and the results show that for large quantities the pump-expulsion method is the more efficient. An examination of Figure E-18 illustrates the increased losses associated with the pressure expulsion method. Since the Orbital Propellant Depot (OPD) propellant residuals represent a direct loss, the use of warm gas for pressure expulsion appears favorable, and the critical liquid residuals also must be controlled. They are minimized by the proper selection of tank outlet geometry and flow rate throttling at the point of incipient ullage-gas pull-through.

The propellant system must include both a transfer line and (in a common container) a return line for vapor. This permits the vapor displaced from the tug propellant tank to be returned to the OPD. This receiver gas return to source can in future studies be modified to expand part of it to a low pressure, like 1 psia (0.69 N/cm^2), and to extract some additional cooling. In fact, such a method would prevent loss of liquid in venting. The losses associated with propellant transfer from one tank to another (fluid transfer) can be eliminated by the use of modular tankage exchange at the

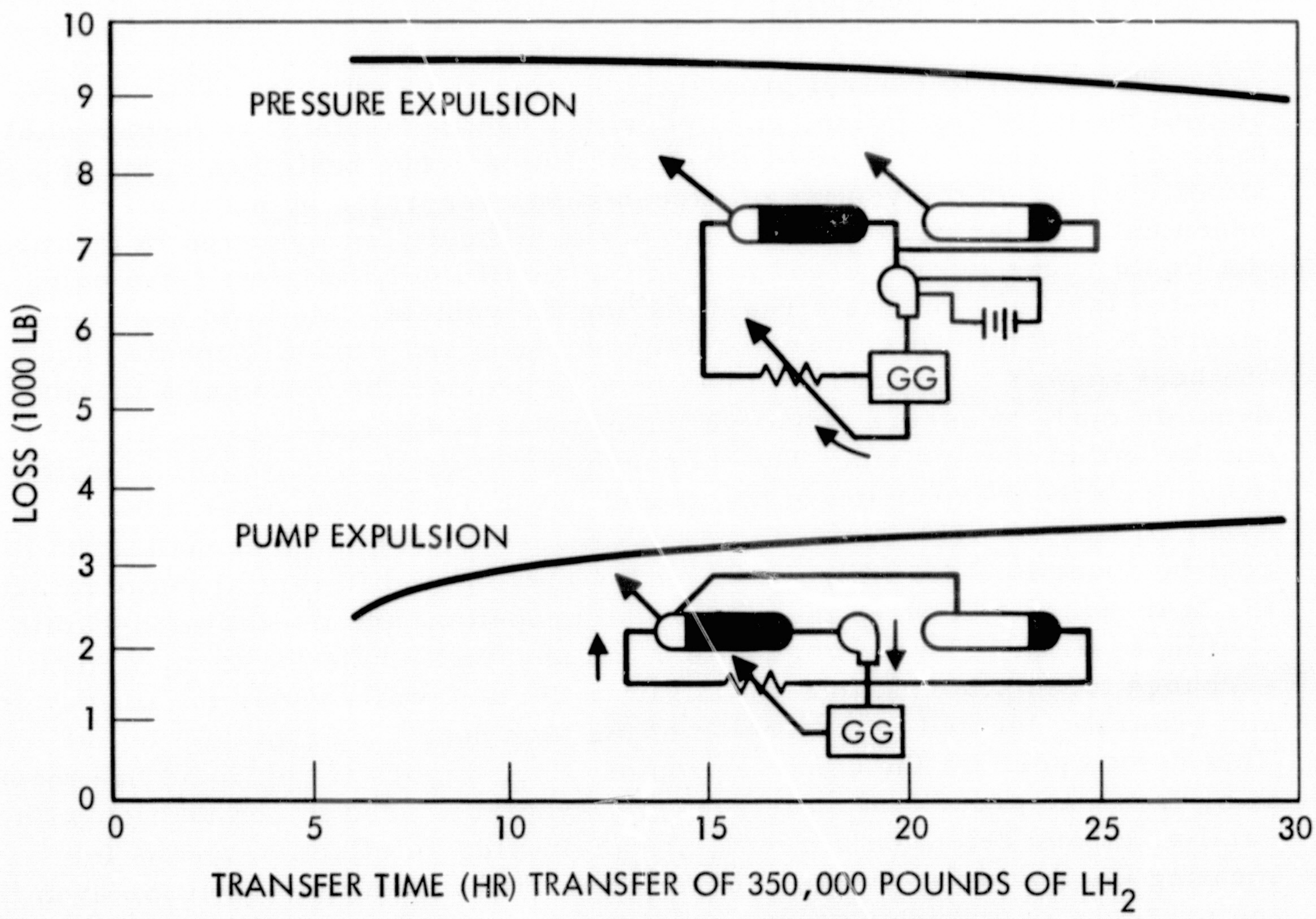


Figure E-18. Increased Losses Associated With Pressure Expulsion Method

OPD. The additional complexity to tug has made this method of little interest as it would probably influence tug configuration disproportionately to its benefit. Another possibility that remains to be studied is a modular transfer between the EOS orbiter and the OPD with a fluid transfer adopted later between tug and the OPD.

THERMODYNAMIC VENTING

The ability to control propellant heating and saturation temperature by well designed insulation, coupled with a venting system, is fundamental to long-term space storage of cryogenic fluids. The basic problems of stratification and the venting of excess vapor generated by heating are accentuated under zero "g" because of the difficulty encountered in locating the liquid vapor interface and preventing small liquid droplets from being ingested into the vent gas stream leaving the vehicle. Methods have been studied to restrain this interface using screens and capillary media, but the best answer to the problem has been to provide the option of a thermodynamic cycle to assure conservation of mass and yet minimize internal energy within the system. The thermodynamic venting system is a device providing direct expansion of the propellant into heat exchanger tubes at a lower pressure. The Joule-Thomson expansion causes the propellant to cool by about 10 F (5.5 C), which means there is a differential temperature for heat transfer between the rest of the propellant and the thermodynamic venting system. The venting system can physically take the form of heat exchange tubes attached intermittantly to the tank walls under the insulation and generally located in the region of the tank near the propellant outlet. This design permits the condensation of ullage vapors if the liquid has moved away from the propellant outlet. If this system is used in conjunction with capillaries and retention screens, it can provide an effective means for locating the liquid vapor interface and retaining it in a fixed position even in the presence of disturbing torques from external influences like ACS thrusters.

Another possible version of the thermodynamic venting system provides a heat exchanger that exchanges heat between the vented fluid (which is expanded to about 1 psia, .69 N/cm^2) and the liquid in the tank. This system has been shown located within the tank and outside the tank as well. In either event, there is a requirement to circulate the liquid through the heat exchanger to provide uniform subcooling and dispersal or mixing of the cooled fluid with that remaining in the tank. It is likely that such a venting system would be located outside the tank in an insulation-protected well so that early development modifications and repairs can be made without resorting to entry to the tankage for each case of rework.

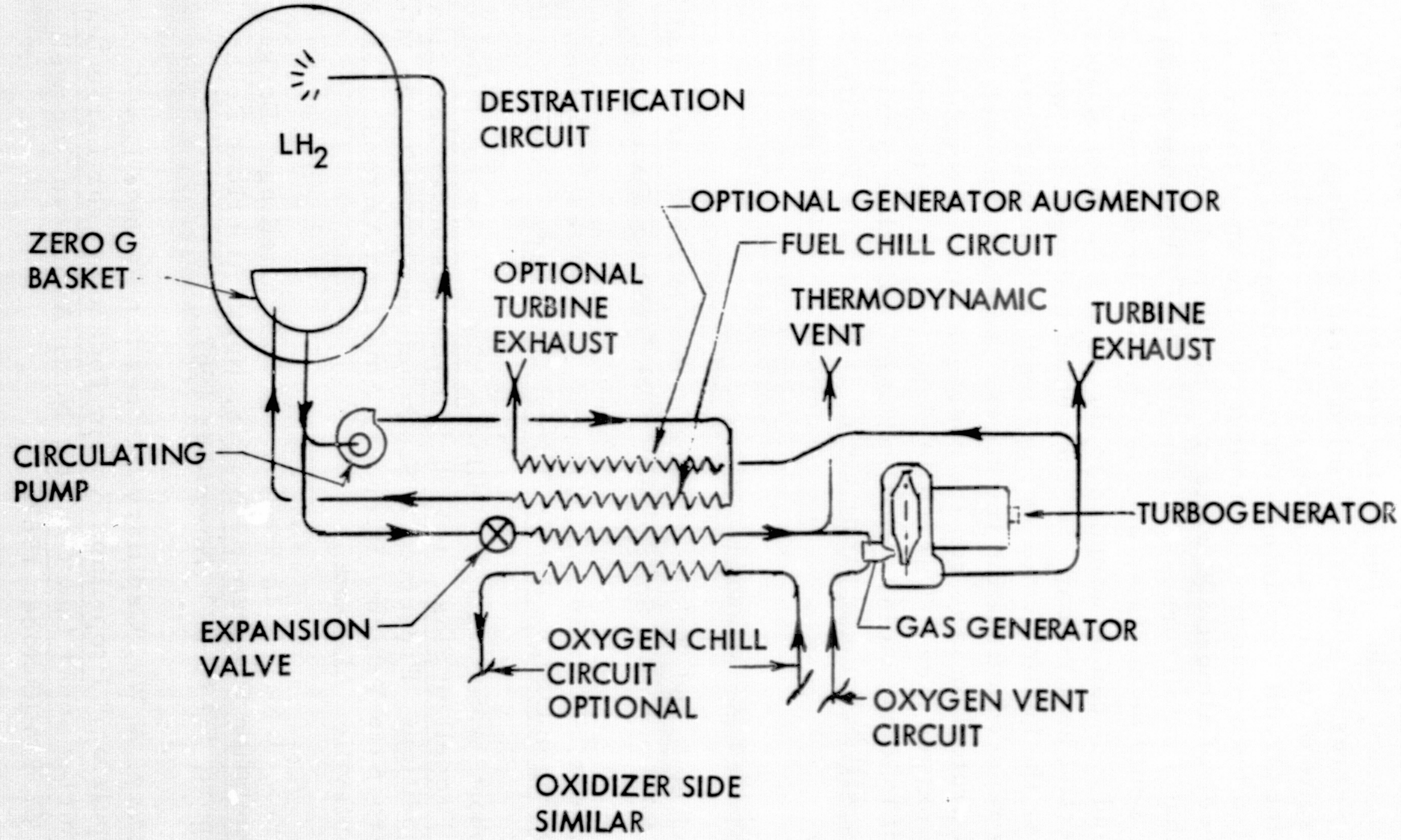


Figure E-19. Integrated Propellant Thermal Control

An unusual form of the thermodynamic vent system is displayed in Figure E-19. This system has the usual expansion valve to produce cooling (refrigerant fluid) and a separate circuit to exchange heat from the propellant with the refrigerant fluid. The exchanger circuit is powered by a small fan that also serves as a destratification fan if the ullage begins to rise in pressure in spite of the action of thermodynamic venting. The addition of an oxygen-vent circuit combining with the hydrogen to produce electrical power has also been examined in modest detail to show that an average delivered power of almost 300 watts may be derived from the boiloff. The problem is that a small turbo-alternator of only 1 kilowatt would be employed and that this would be operated on a 1/3-hr-per-hr duty cycle. The boiloff of oxidizer and fuel can be adjusted to the ideal F/O ratio of about 1:1 by adopting a partial heat exchange of the vented hydrogen gas with the oxygen. In this manner the heating value is improved, and the oxidizer tank will be able to use a reduced insulation thickness. Another approach to oxidizer tank thermal protection is to run the hydrogen vapor through a vapor-cooled shield on the oxygen tank. This would make it possible to reduce the oxygen boiloff to zero, but at a cost in weight. Further consideration of such methods will become appropriate as later studies identify the hardware more clearly.

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APPENDIX F

CRYOGENIC STORAGE

by

Dr. M.B. Hammond

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APPENDIX F. CRYOGENIC STORAGE

SUMMARY

Tug space operations in excess of a day will require insulation much more effective than the plastic foams and honeycombs used on previous cryogenic launch vehicles and injection stages. The government agencies charged with research and industry have been working to solve the problem of long-term cryogenic storage with a result that applicable high-performance or multilayer insulation materials (HPI or MLI) are in an advanced state of development. Current design studies have concentrated on the choice of fabrication, installation, and interstitial gas control as they influence insulation performance, endurance, reuse, and refurbishment.

Development of a high-performance cryogenic propulsion stage requires a high mass fraction structure and low propellant boiloff. Thus a lightweight, low-heat-flow insulation is required. Use of a large vacuum-jacketed double-wall Dewar as applied to the EOS orbiter vehicle would be out of the question for a ground-based or space-based tug because of the extra weight for the structural jacket (about 2000 pounds) (907 kilograms). Since this jacket protects only insulation, the added weight represents a direct reduction of the payload. The study of space-evacuated insulation for the tug has led to design evaluations of the means of purging and venting these insulations. Whether space based or ground based, if the tank contains cryogens during prelaunch, the insulation must be evacuated rapidly in space. The purge gases are used to prevent the intrusion of detrimental atmospheric constituents during the prelaunch phases and may employ either helium or nitrogen purge gas contained in a bag around the insulation. In the case of nitrogen purge, a substrate of closed cell polyurethane foam is required to prevent purge gas condensation on the very cold liquid-hydrogen-tank walls. Furthermore, the ground-based tug may have need to reuse the insulation rather than apply a new insulation every flight. In that case, the insulation must be repressurized and protected during entry. The types of multilayer insulation (MLI)* include the current candidates of NARSAM,

*The descriptive terms "multilayer insulation (MLI). high-performance insulation (HPI), and superinsulation" are synonymous. Since the basic nature of these insulations is multilayer, the abbreviation MIL will be used here.

GAC-9 Superfloc, NRC-2, and doubly aluminized Mylar-nylon mesh. These insulations are all the subject of considerable experience and study with regard to their installation and performance characteristics. Narrowing the selection to a preference for NARSAM is related to NR experience with this material and the ability to predict the desired performance accurately.

During ascent through the atmosphere, the insulation purge bag is vented through a valve of generous dimensions to permit the insulation to be space-evacuated. The vent valve is an opening in the lightweight purge bag or soft shell of the insulation. It can be actuated by pressure or mechanical means. An interesting result of this study is that the boost heat transfer and the duration in space lead to the selection of the type of purge gas. Analysis has shown that for space durations of more than one or two weeks (containing cryogens), it is desirable to use a helium purge gas. Nitrogen gas applies to shorter durations. The boost heat transfer associated with helium gas is traded against higher orbital heat transfer for the nitrogen-purged insulation, based on equal weight and the ineffectiveness of the foam substrate in orbit.

GROUND BASE VERSUS SPACE BASE REQUIREMENTS

The first requirement for a ground-based system is a ground purge gas to avoid condensation of air or other detrimental species in the atmosphere. The space-based installation may have a similar requirement if the first time it is put in orbit it must be fueled and ready to go. Unless the space-based requirement implies that the tankage will be put in orbit empty, both systems, ground based or space based, must be provided with a ground purge system to avoid condensation. Figure F-1 is the basic trade tree that compares the qualitive considerations for selecting the basic design approach.

No great breakthroughs in the preevacuated insulations have been forthcoming, so the main effort has been directed at regular MLI, where the purge system stands out as a dominant requirement. The main differences between the basing concepts are between the more or less complex means required to recover and reuse the MLI in the ground-based case.

In most cases, since the space-based system will be designed to spend a great deal of time in space, it will be provided initially with a helium purge, and no foam substrate will be used. However, for the ground-based system, frequently missions may be as short as two to seven days in orbit, and no more than 14 days in extreme cases. Therefore, it is possible to use

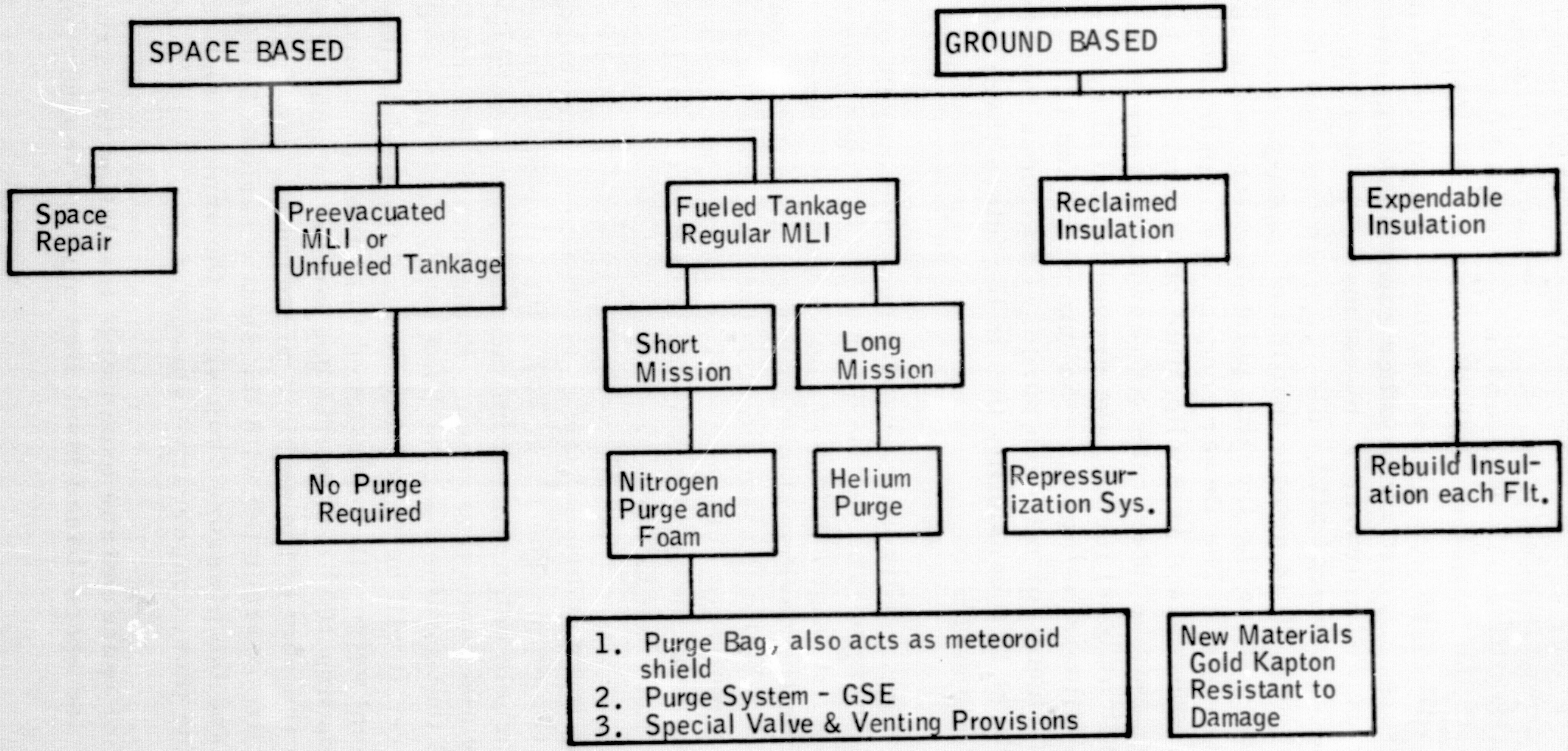


Figure F-1. Basic Trades Tree and Design Approach

nitrogen as the purge gas with a foam substrate. One advantage of using nitrogen with a foam substrate is that the temperature of the outside surface of the insulated tank sitting on the ground in the EOS orbiter payload bay will be warmer with the foam substrate than in the case of the helium-purged insulation. That will mean less possibility of condensation on the outside in the event of exposure to moist air or other condensible constituents in the payload bay. Figure F-2 illustrates, by comparison, how the external surface temperature of typical MLI builds up when installed in the payload bay of the EOS orbiter. It is notable that the nitrogen-purged design surface temperature remains above the freezing point of water, while the helium-purged design is below it. The Dewar concept has been included in this figure for comparison since it is Apollo state of the art. The heat flux curve on the figure may be interpreted as showing that for an MLI thickness of about 1.5 (0.0381 meters) inches, the ground boiloff of hydrogen is 1/2 lb/hr (0.2268 killograms/hour) for helium purge and 1/10 lb/hr (0.0454 killograms/hour) for nitrogen purge. Continuous topping until liftoff is assumed, and there is a possibility that in the future there will be topping from the EOS-OMS tankage.

In view of the possibility of frost forming on the outside of the helium-purged insulation, there will be the need for a dry nitrogen purge in the orbiter payload bay. It is probable that such a purge will be employed regardless to reduce the obvious flammability and explosion hazards of the hydrogen propellant. While the vehicle nitrogen purge will be reasonably effective in keeping moisture out, it is still necessary to have a separate dry-gas purge system on the vehicle insulation to clean it initially and prevent contamination when loaded.

Boost venting is an additional factor related to ground-based versus space-based requirements. The purge bag vent valve must provide sufficient area to handle outgassing from the multilayer insulation into the shuttle payload bay and hence into space. Evaluation of the shuttle environment has indicated that a rather severe back pressure is imposed inside the cargo bay; therefore, it will be necessary to open the payload bay doors. The sizing of the vent valve will be designed to achieve pressures within the insulation approximating 10^{-4} torr within one hour of launch. This pressure in the insulation is the sum of the ambient pressure (external) plus the aperture pressure drop associated with the shuttle payload bay (and with the vent valve itself), and the pressure drop within the annulus and across the insulation to give the most conservative viewpoint.

The question of reuse versus expendable insulation is a ground-based tradeoff. Clearly, the space-based insulation must be reused from mission

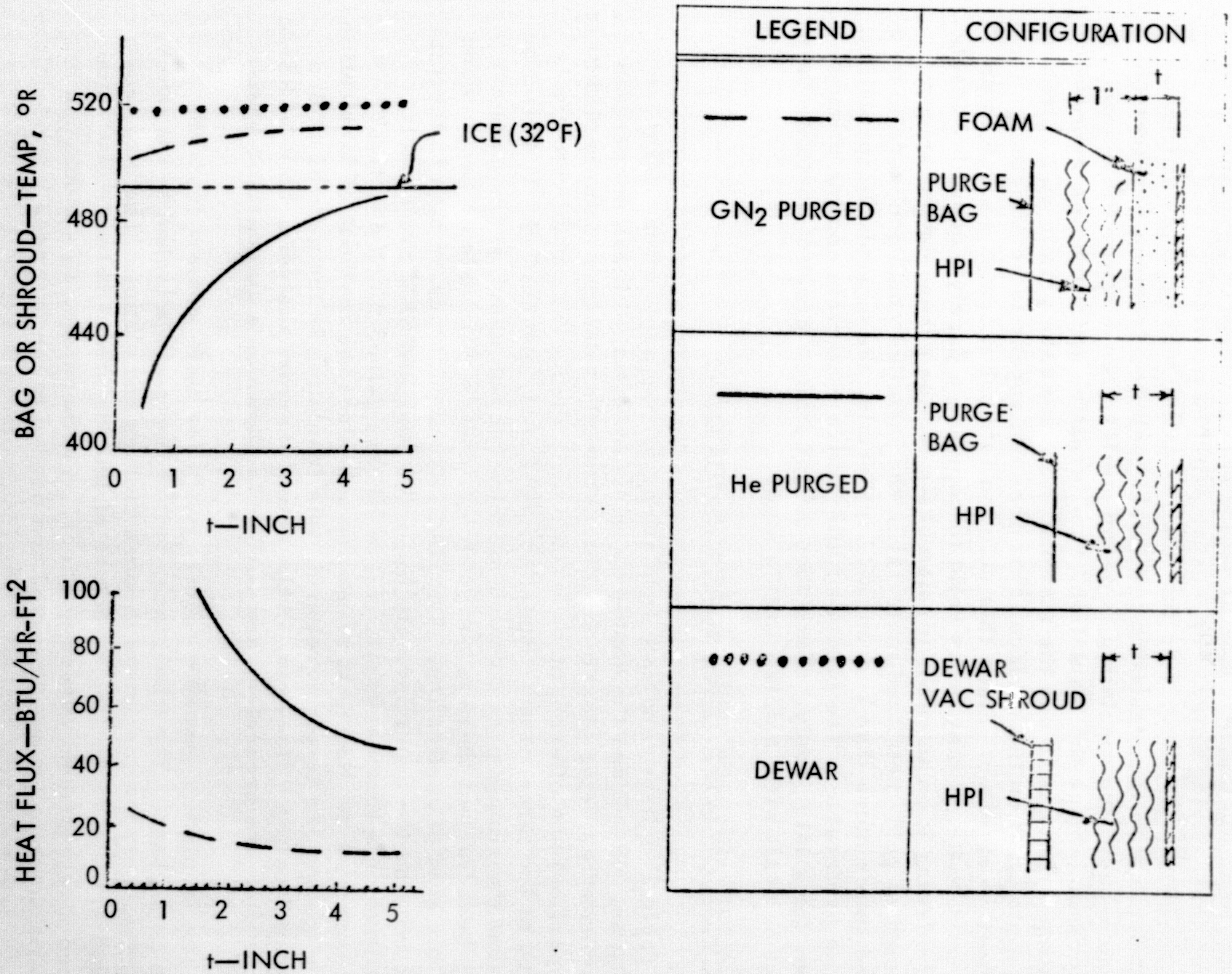


Figure F-2. Ground Hold Thermal Performance Comparisons for Different Multilayer Insulation Concepts

to mission unless provision is made to repair or reinstall it in orbit. Since orbital repair is not anticipated, the tug must be returned to earth for rework. And, whether ground- or space-based, reuse of the insulation requires that, during entry, repressurization be provided through a moisture-eliminating device so that the insulation will be repressurized without damage to the insulation materials (unless infrequent replacement of a space-based tug insulation is permissible). Since water (or humidity) and heat can damage the insulation, an alternative approach to a repressurization system is to use the material Kapton as the plastic substrate and a gold metalizing on the surfaces that does not appear to be affected by condensation of water. Since this solution is much more expensive on an initial cost basis, it may be as valid to compare a purely expendable insulation, replacing it on an each-mission basis for the ground-based system. If the cost of gold-metalized material in the installed condition were to be a hundred times more than current MLI materials, the expendable insulation might be preferable. The choice of alternative materials and metalizing remains for a later study. Since they are of a more practical nature, they will not be dealt with in detail in this study.

SELECTION OF POTENTIAL SUPERINSULATIONS

Table F-1 lists candidate insulations for tug application. Of these insulations, NRC-2 and the doubly aluminized Mylar with nylon mesh can be eliminated as being closely similar to the other designs (namely NARSAM and Superfloc) and offering no advantage over them. The main concern is directed at the three listed, Superfloc, NARSAM, and GAC-9. It should be noted that the simple preevacuated system (vacuum panel) is the heaviest for a given heat flux but there is not sufficient data to eliminate or recommend it for this kind of application. The GAC-9 is about twice as high in thermal conductivity as the Superfloc, and Superfloc and NARSAM are closely competitive on a thermal basis. Since NR has gained much practical experience with the NARSAM insulation and since the methods of fabrication and application are better developed, this insulation is the natural choice for tug application. However, in the actual thermal analysis, both NARSAM and Superfloc are presented, and GAC-9 is shown on a comparison basis.

The insulation concepts require posts for main support from the cryogenic tankage. The fiberglass, hollow posts are attached to the tank wall and support the insulation through lacings and straps. The NARSAM uses an intricate supporting structure of Dacron straps between posts and stiffened straps around curved sections. Furthermore, this insulation has small-diameter nylon pins inserted and bonded to the layers to provide close tolerance layer density control. The insulation is designed to be built in

Table F-1. Candidate Insulations for Tug Application

Type (Supplier)	Description	Strong Points	Potential Penalties
GAC-9 (Goodyear)	Doubly aluminized Mylar sliced foam separators	• Most advanced • Manufacturable, repeatable • Good recoverability from temporary compression	• High conductivity and density • Very thermally sensitive to compression • High outgassing • Thick installation/ complicates design
DAM/NM (Open)	Doubly aluminized Mylar nylon mesh separators	• Low conductivity • Manufacturable, repeatable • Thin installation/ simplifies attaching	• High density • Thermal sensitive to compression • High outgassing
DAM/ Tissuglas (Open)	Doubly aluminized Mylar Tissuglas separators	• Low conductivity • Manufacturable, repeatable • Good recoverability from temporary compression	• High density • Moderately thermal sensitive to compression • Fragile spacer material
Superfloc (Convair)	Doubly aluminized Mylar dacron flock separators	• Low conductivity and density • Manufacturable, repeatable • Good recoverability from temporary compression	• Thermal sensitive to compression • Relatively high materials cost
NRC-2 (National Research Corp.)	Singly aluminized Mylar random crinkle separation	• Low conductivity and density • Relatively small thermal degradation from ±35% deflection • Thin installation/ simplifies attaching • Low materials cost	• Requires extensive layer density control/ complex and heavy installation • Non-repeatable natural lay-up • Compression difficult to inspect
NARSAM (NR-SD)	Singly aluminized Mylar small embossment separators	• Low conductivity and density • Relative thermal insensitivity to deflection • Thin installation/ simplifies attaching • Low materials cost	• Requires layer density control • Aluminized one side only, moisture can permeate Mylar and separate aluminum from Mylar
Vacuum Panel (Linde Division of UCC)	Doubly aluminized Mylar glass fiber separation Lead-Mylar laminate seal	• Does not require evacuation or repressurization • Not degraded by moisture exposure • Can be installed easily	• Highest conductivity and density • Seal sensitive to entry heat • Highly dependent upon prelaunch evacuation • Seal may be damaged easily

50 layer, 1/2-inch (1.27-cm) thicknesses of precut and fitted subassemblies. One-inch and 1-1/2 inch (3.81-cm) thicknesses would be constructed of two and three such subassemblies, respectively. The joints, at the edges, are designed to have an offset of 1 inch (2.54 centimeters) between separate abutting subassemblies. (Similar detail could be provided for other insulations but this NARSAM data are provided to indicate that considerable practical thought and experiment has preceded the application herein considered for the tug.)

Figure F-3 is a picture of the insulation (The NARSAM composite) with a description of the NR cryogenic facility calorimeter used in a recent test with a 40-ft^2 (3.716-M^2) test specimen. This test was one of a continuing series intended to characterize the performance of the candidate MLI systems and the attendant practical and operational considerations involved in application. The test results verified the installed performance as being within 15 percent of the value predicted from a thermal analysis that included allowances for support posts, pins, joints, and vent perforations.*

The multilayer insulations have an effective thermal conductivity that is about three orders of magnitude smaller than other conventional insulations. The multiple radiation barriers are also arranged and supported to minimize the residual solid conduction in the spaces and supports. Also, the insulation must be evacuated to space (a pressure of 10^{-4} torr or less required) to eliminate residual gas conduction effects. All of the candidates use metalized polyester film (such as aluminized Mylar) but effort has been given to other materials such as gold metalized Kapton for resistance to environmental moisture and entry heat degradation. The newer materials may require other concepts of separation since embossing and flocking may be unsuitable and foam or nylon spaces may be inadequate for the thermal environment. The protection offered by the EOS orbiter payload bay is sufficient to prevent excessive temperatures. Thus the current concept is to continue with the lower-temperature conventional materials.

INSULATION PURGE SYSTEM

The insulation purge system encompasses the purge bag required to retain the purge as well as the system components to admit, control, and

*Freeman, B. Test Report, Cryogenic Subsystems Technology High Performance Insulation Research, NARSAM Composite, Liquid Hydrogen Boiloff From a Guarded Calorimeter. North American Rockwell Space Division, SD 70-441 (October 1970).

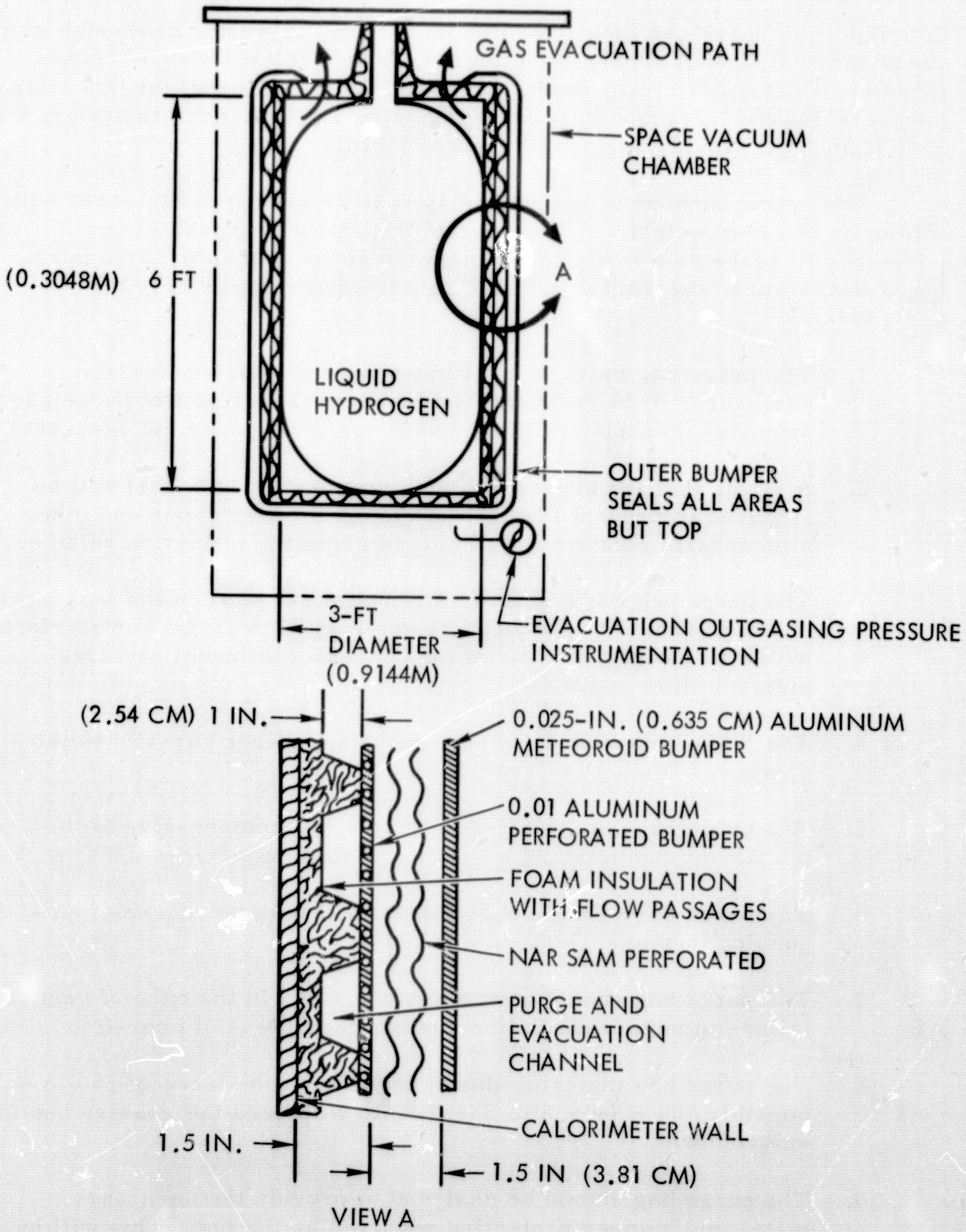


Figure F-3. MLI Test Article/Calorimeter Assembly

distribute the purge gas inside the insulation. In the ground prelaunch case, the purge system will supply dry nitrogen (or helium) at under 10 inches (25.4 centimeters) H_2O differential pressure. Since this will involve a high-pressure source, a double step-down regulator would be used in series, with redundancy, to provide series-parallel reliability.

The prime purpose of the purge bag is to retain the purge gases with a minimum of added weight. A second, yet important, purpose is for the purge bag to serve as a meteoroid bumper in space. In view of the purge bag's importance, the following list of requirements should help the designer:

1. The purge bag must be light in weight and be a low leakage enclosure. It is necessary to conserve the amount of purge gas expended (especially helium, which is a basic natural resource).

2. A slight positive pressure (four to ten inches of water) will be required to prevent the back diffusion of water vapor and other atmospheric contaminants from entering the insulation volume.

3. The purge bag must be maintained erect even when the purge gas has been vented to space. The bag must at least be noninterfering with the vent gas flow paths for both the continuum and free molecule flow regimes.

4. The purge bag must be able to withstand minor abrasion without rupture.

5. The purge bag should not be abrasive nor compressive to the multilayer insulation.

6. The purge bag itself should not be an excessive source of outgassing in space.

7. The purge bag should have a rupture valve in the case of vent valve failure to permit insulation venting regardless of operation.

8. The purge bag material should have heat resistance of 350 to possibly 600 F (449.8 to 588.7 K) to withstand the reentry heating environment.

9. The purge bag should be designed to provide the necessary meteoroid bumper protection required by the tug. This will be the controlling factor if a minimum purge bag proves inadequate for the particular mission.

The purge bag may be constructed of a wide range of materials, some of which will now be discussed. The first purge bag shown in Figure F-4, is an approach based on the trends established for nonrigid, lighter-than-air craft. The material is a rubberized cloth of nylon or similar polymer fabric rayon, polyethylene). It is both flexible and elastic and obtains its strength from the cloth while gaining its seal quality from the rubber (polymer) impregnant (butyl or neoprene) or film. The bag is a lightweight casing that is pressure stiffened and self-erected in the atmosphere, but must be supported when vented. The purge bag is shown stiffened and shaped by pressure-erected hoops. The bag may be tensioned to the tank skirts and insulation posts to keep it off the insulation while the hoops maintain the bag in circular shape. Alternately, the hoops may be constructed of lightweight aluminum framing to provide the same function.

The version of the purge bag shown in Figure F-5 is a rigid cover of glass cloth. The use of a single ply of epoxy impregnated 181 glass cloth is anticipated here, and it may be corrugated to give axial bending strength on a cylindrical tank. The figure shows a typical installation of this material employing bonded fiberglass hat section stiffeners to keep the shape circular in spite of flight load conditions or applied forces. This design shows the purge bag and the support struts for the tank secured to an aluminum ring at both ends of the tank. (The rings are the main load pickup points.) The

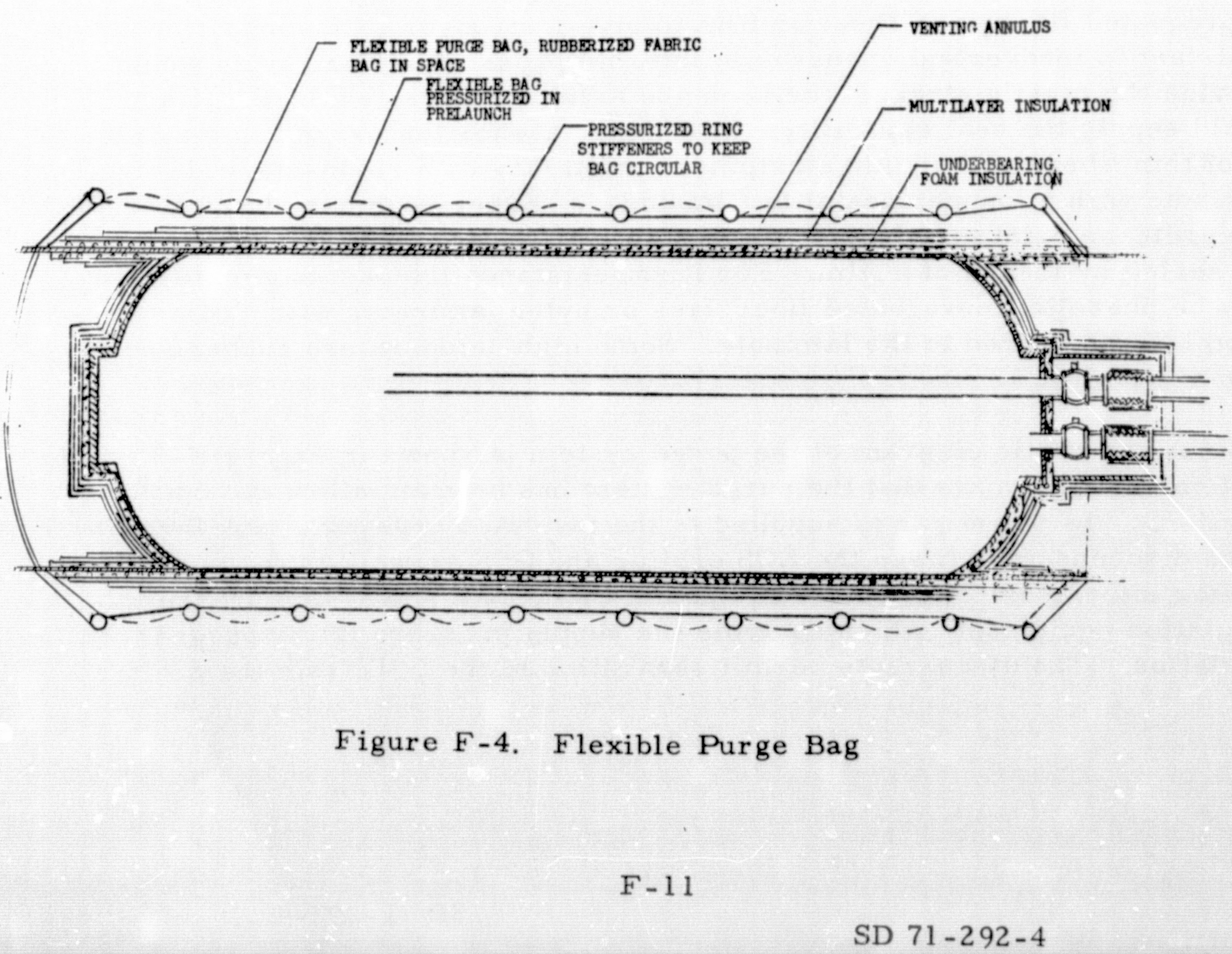


Figure F-4. Flexible Purge Bag

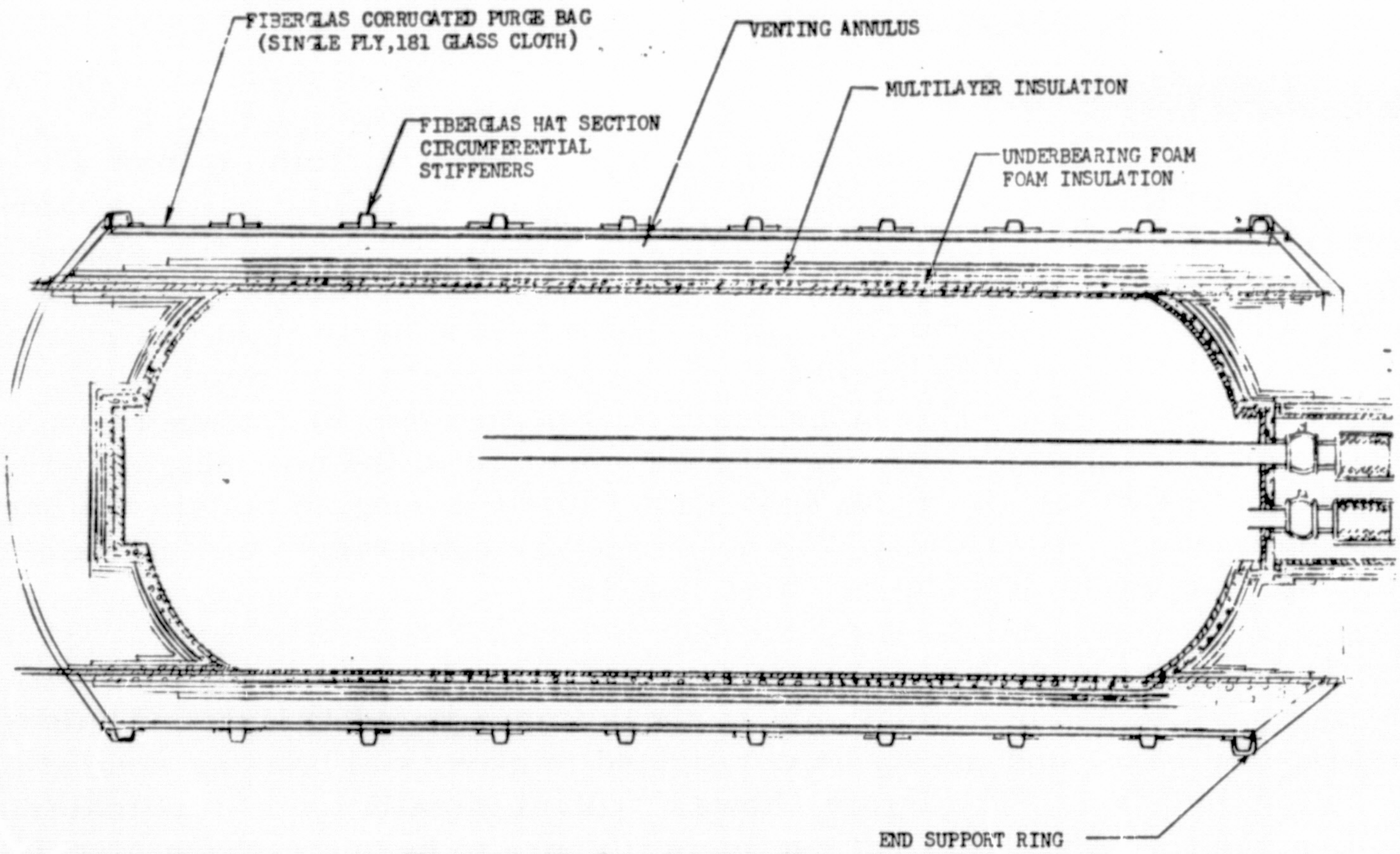


Figure F-5. Rigid Purge Bag

purge bag could also be worked out with a tensilized cover design in which a two-dimensional cross pattern of narrow fiberglass roving is applied over an impregnated fiberglass sheet or film to give a stronger self-supporting cover resistant to inadvertent excesses in internal pressure. The roving would provide the tensile strength for hoop and axial stresses, while the film would supply the seal capability. The materials for the purge bag may also be of thin fiberglass or thin elastomeric cloth having a resin-starved laminate with a low permeability film added. Experience has shown that polyurethane paint may be satisfactory for the joints and surfaces that are not sealed by means of a film. For heat resistance the best bag materials may be phenolic impregnated fiberglass or nylon laminates with Tedlar or Kapton films bonded to the laminate. Some high-temperature rubbers are also available.

A schematic diagram of the purge system is shown in Figure F-6. This system concept shows that the purge system has both an inflow and outflow function. The purge gas is supplied to the insulated cryogenic tank through quick disconnects between the EOS orbiter and GSE as well as the tug orbiter interface. The orbiter equipment includes the two-step pressure regulation equipment, which controls the supply pressure to the purged insulation. The disconnects permit separation of the GSE from the orbiter

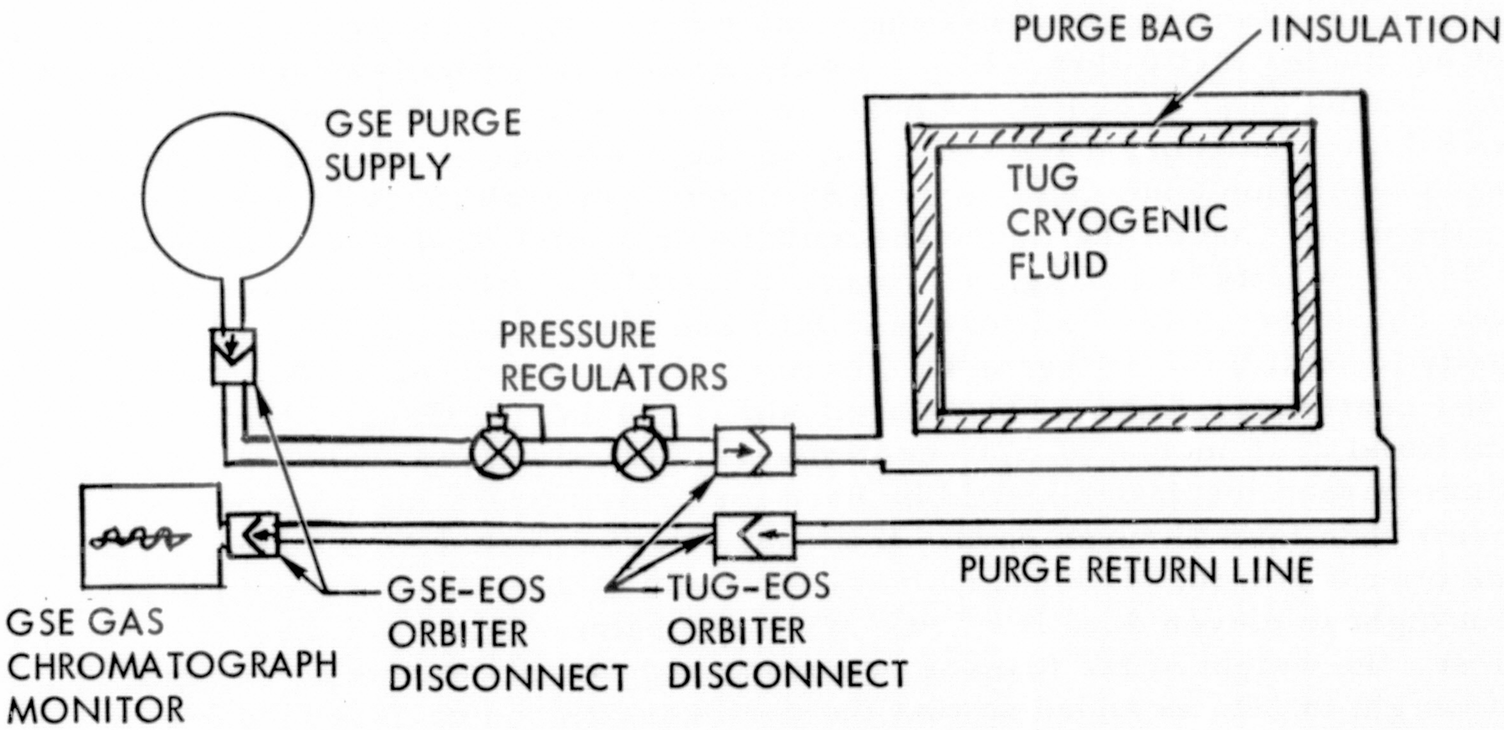


Figure F-6. Purge System Schematic

and the orbiter from the tug in the appropriate sequence. The only other equipment not shown is the additional shutoff valves (and regulator redundancy) to isolate the segments of the system in each vehicle, the vent valve that permits rapid evacuation of the insulation, and the repressurization system for safe entry return for reuse of a ground-based system. These last two assume such an importance that they will be given more thorough consideration in the next two sections. The function of the gas chromatograph on the outlet is primarily that of a gas analyzer. It can be left out without seriously disturbing the system performance. However, it provides a continuous check on hydrogen or oxygen leakage as well as water vapor intrusion. The chromatograph should be envisioned as a continuous checkout tool for the payload bay purge, the degree of insulation preconditioning, and to determine if contamination is taking place; i.e., to help foresee potential trouble spots before they cause trouble. The dominant purpose of keeping the insulation clean by the purge is second only to the primary function of the purge bag as a meteoroid bumper. The meteoroid application requires hardness to break up the fast moving, frangible meteorites. For a discussion of the controlling factors in meteoroid protection. (Refer to Appendix D for more detail.)

The actual selection of the purge gas for the insulation is based on the equal weight and heat transfer crossover shown in Figure F-7. This chart has been based on the boost heating studies presented in conjunction with the venting studies that follow. It is interesting that the helium-purged insulation contributes an accumulated heat input of 11.7 Btu/ft^2 per inch (33.78 J/cm^2 per cm) of thickness during the boost phase, while the nitrogen-purged insulation contributes only 1.85 Btu/ft^2 per inch (5.34 J/cm^2 per cm). For the upper line on the figure the multilayer insulation is assumed to be 2.7 lb/ft^3 density (43.25 kg/m^3) (foam, 2 lb/ft^3) (32.04 kg/m^3) while for the lower curve the closed cell foam and MLI are assumed to have the same density (2 lb/ft^3) (32.04 kg/m^3). The analysis shows that if 1-1/2 inches (0.381 centimeters) of MLI were used with a 3/4-in. (1.905-cm) thickness foam (total 2.25 inches (0.5715 centimeters), fraction foam = 0.333) the helium-purged insulation should be used for orbital operating time periods greater than four to seven days. If the MLI were 3 inches (7.63 centimeters) thick over a 3/4-in. (1.905-cm) foam substrate, the crossover would be about eight to eleven days in orbit. When the helium purge is used, of course, the weight of the foam is removed, and for this analysis, an equivalent weight of MLI is added so that the performance comparison is made on the basis of total equivalent weight. At the lower foam limit (at a foam fraction of 0.07) the foam surface reaches its lower limiting temperature of -300 F (-184 C). Below this it is no longer possible to prevent the nitrogen purge gas from condensing, and this limit occurs at 26 days of equivalent

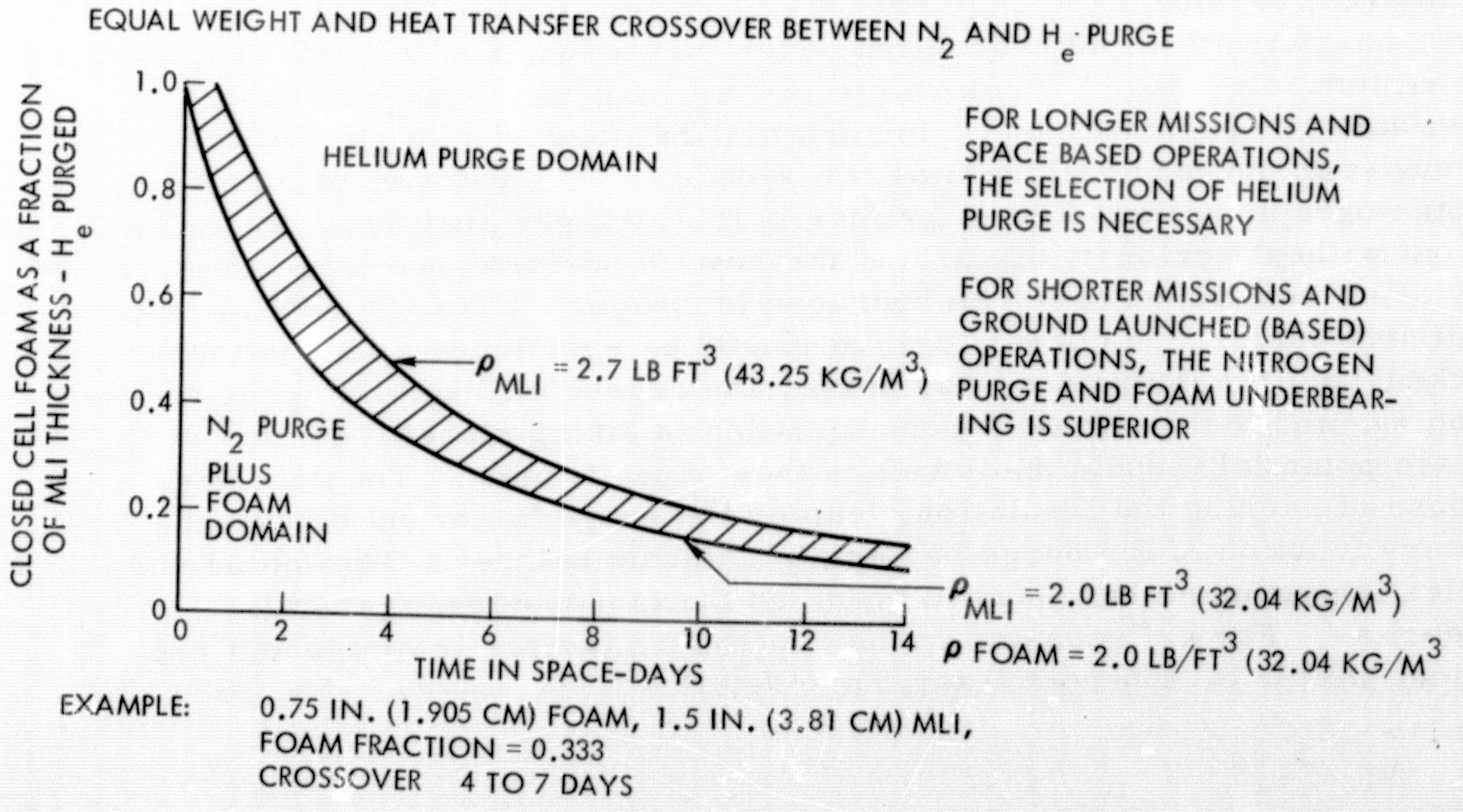


Figure F-7. Selection of Purge Gas for Hydrogen Tank Insulation

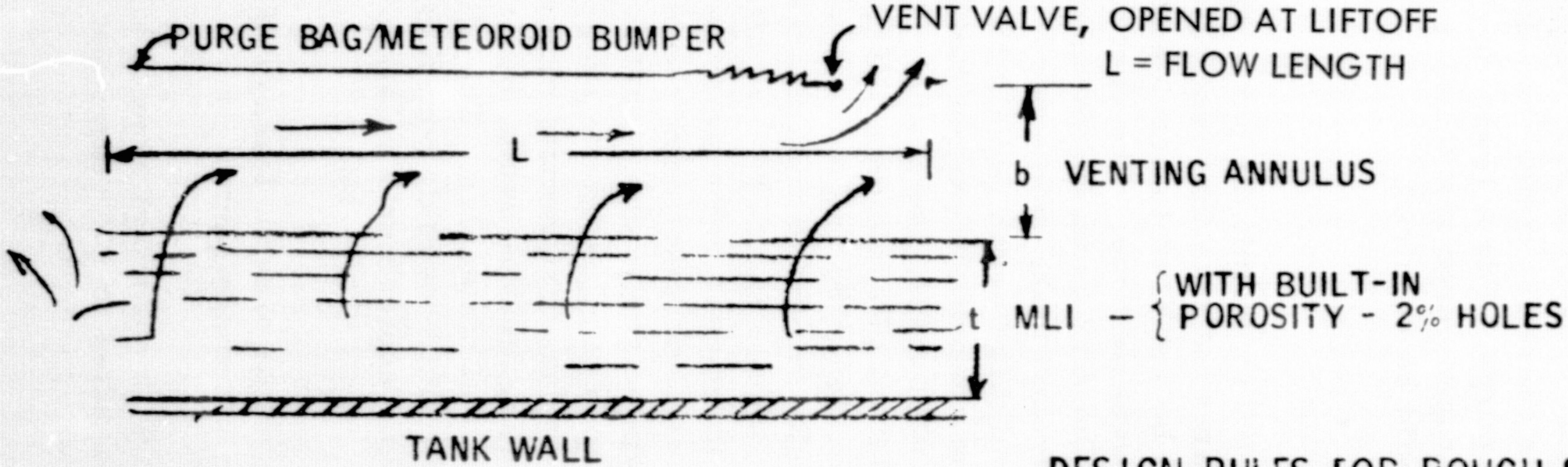
orbital operation (equal density). Optimum insulation thicknesses for 30 days are on the order of 1.5 inches (3.81 centimeters) of MLI; hence, the foam underbearing to reach the lower limit (least possible foam weight) is about 1/8 inch (0.3175 centimeters), which may be extremely difficult to manufacture. Since the 3/4-in. (1.905-cm) foam thickness is more practical to manufacture, the four- to seven-day crossover is more meaningful. Any crogenic spacecraft having longer operating times than these should be designed to incorporate the helium purge. For a seven-day shuttle mission it would be suitable to use nitrogen purge and foam, while for a 30-day tug mission the helium purge would be required.

INSULATION VENTING SYSTEM

During boost a vent valve must be actuated to permit the purge gas to vent from the insulation. Preliminary estimates indicate that vent valves of sizable proportions (a few square feet) will be required to provide the necessary high vacuum condition within a reasonable time. The vent valves will have an effect on vehicle configuration. One will be on the forward skirt, and the other on the aft skirt of the hydrogen tank. Some porosity of the skirts will permit cross flow to vent the bulkhead areas while a third smaller vent valve will be designed to vent the area of the oxygen tanks. It should be noted that, in addition to sizing the vent valve, the sizing of the annular passages and the porosity of the insulation will influence the insulation thermal conductivity. Furthermore, the outgassing and leakage sources from within the EOS orbiter will also tend to slow up the venting process by increased back pressure. The opening of the orbiter payload doors after orbiter main engine cutoff should provide the large area required to help vent and outgas the insulation rapidly. A pictorial description of the venting method is shown in Figure F-8. A bellows-type of vent valve, located circumferentially on the vehicle, is actuated open to allow gas to escape. The flow path length, L, stands for the distance the most remote molecule must travel to escape the vent valve. The venting annulus depth, b, represents the flow path restriction to transport of molecules. The MLI is assumed to have 2-percent porosity to permit rapid broadside pumping of gas into the annulus. The following are general requirements relating to the vent valve:

1. The vent valve must seal the purge bag to prevent leakage of purge gas and back-diffusion of atmospheric contaminants.

2. The vent valve must be able to open with sufficient area to allow the superinsulation to be rapidly evacuated to 10^{-4} torr (and below).



DESIGN RULES FOR ROUGH SIZING

$\Delta P_1 \leq 5 \times 10^{-4}$ TORR AFTER 1 HOUR

$\Delta P_2 \approx \Delta P_3 \leq 10^{-4}$ TORR AFTER 1 HOUR

ANNULUS LENGTH/HEIGHT = $L/b \sim 100$

VENT VALVE AREA $\approx$ ANNULUS AREA

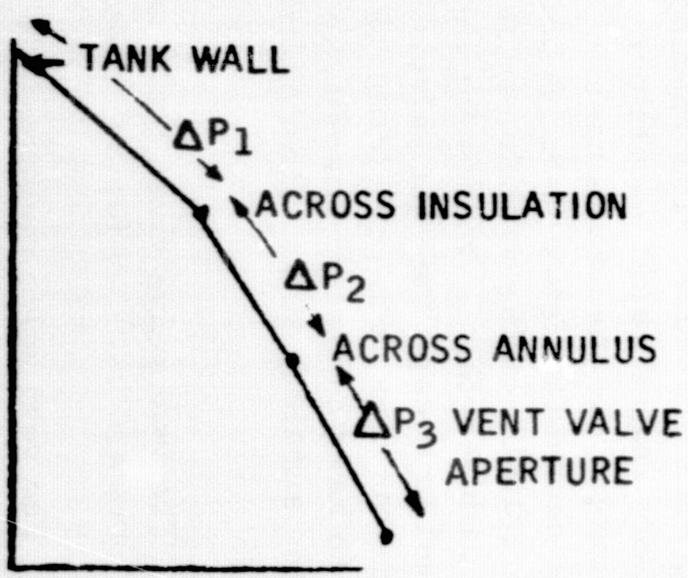


Figure F-8. Venting System for MLI

3. The vent valve actuation system must be capable of opening for boost evacuation and positively closing for repressurization on entry.

4. For the ground-based system, the valve actuation system must provide reliable operation for repeated cycles with capability of being redundantly activated for any single operation.

The next two figures illustrate potential types of vent valves along with support and actuating mechanisms. These illustrations are not to scale or even closely representative of the tug, but they do indicate the need for sealing rims, actuators, etc. Figure F-9 describes a bellows-type vent valve mounted on a support ring of an A-frame mounting system. A piston actuator is used to close the valve when venting is not to take place. At liftoff the pressure on the piston is removed, and the spring return system causes the valve to open. A sealing rim is provided on the edge of the purge bag. A soft gasket material with a metal support backup system is used. The construction of a large-diameter bellows should not be difficult, although the bellows will probably be made in two parts for installation, repair, and replacement.

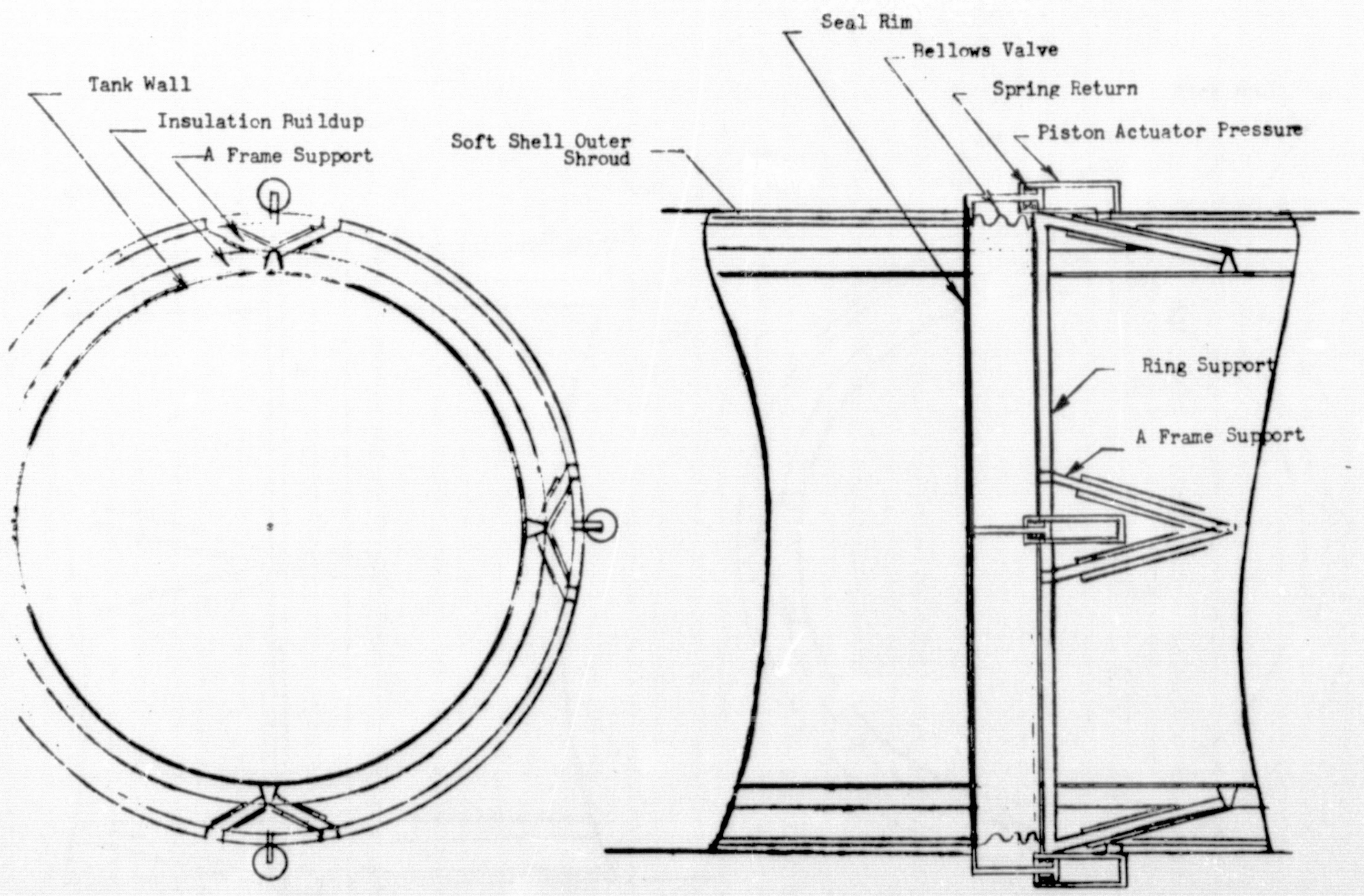


Figure F-9. Bellows Type Vent Valve

Figure F-10 describes a pressurized-tube vent valve not unlike a circular air mattress. The pneumatic expansion of the pressurized tubes is coupled with a spring return system to open the gap when pressure on the tubes is withdrawn. A support screen is provided for alignment of the tubes and for the sliding surface over which the tubes travel in closing the gap. The mounting of the pressurizing apparatus and the vent valve on a support ring from the tank represents a lightweight installation having features worthy of more design study. Many other design approaches can be envisioned, and a number more simple than those shown. However, the important background for any vent valve design decision must be based on analysis of the contributing factors.

The prime purpose of the venting analysis is to design the physical parameters of the venting system so that the boost heat transfer will be held within reasonable limits and so that evacuated performance will be achieved rapidly. As shown in Figure F-8, the venting analysis includes the pressure drop across the insulation, the pressure drop in the venting annulus and across the vent valve aperture, and, if applicable, the pressure drop from inside the vehicle to the back pressure of space. An example of a venting analysis is shown in Figure F-11. It presents altitude pressure versus time for both Saturn V and the EOS orbiter along with estimates of internal compartment pressure within the shuttle. The prime assumptions of the venting

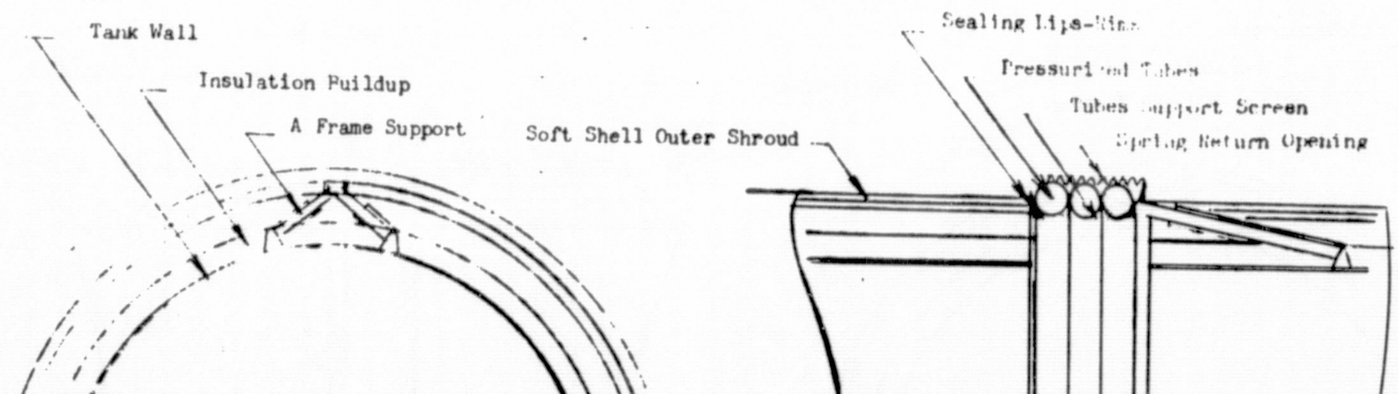


Figure F-10. Pressurized Tubes Vent Valve

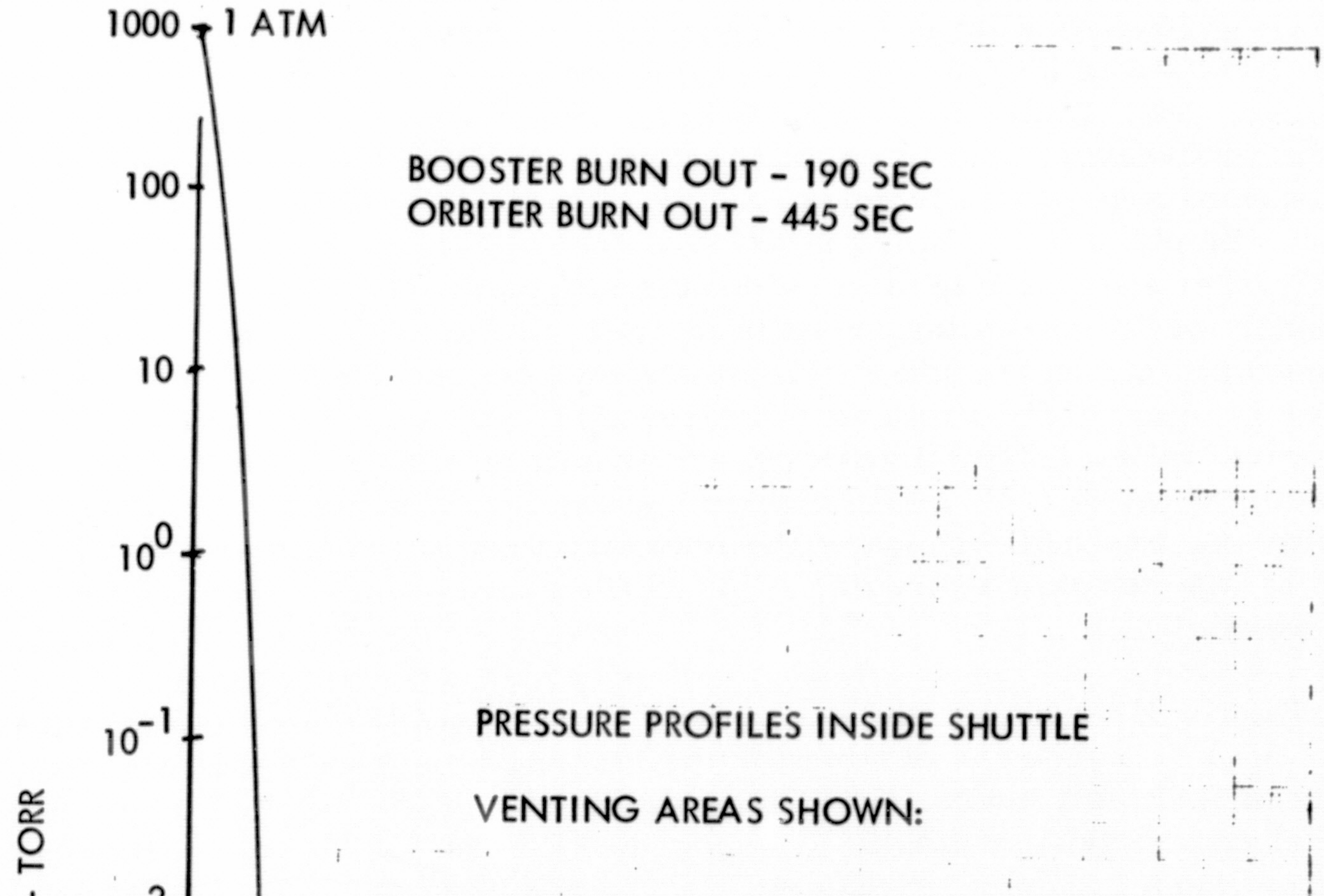


Figure F-11. Ambient Pressure Versus Time After Liftoff for EOS and Saturn V

analysis for the EOS orbiter include (a). a quasi-steady free molecule flow after the transient flow of the first 10 minutes; (b). vehicle internal outgassing sources totaling 4.4 torr ft^3/sec (0.1246 torr m^3/sec) after one hour in space (decreasing inversely with time); (c). internal leakage from all sources of 1.0 torr-ft^3/sec (0.02832 torr-m^3/sec) and (d). the vent orifice size shown is based on an aperture plus a tube of L/D = 1. These assumptions are generally conservative. They point out that a considerable gain can be made by opening the payload doors shortly after orbiter engine cutoff rather than waiting until payload deployment. In this manner the necessity for large venting areas in the base of the orbiter will also be avoided and, hence, an added complication in the entry phase. The EOS orbiter is injected first into an elliptical orbit with a perigee of about 50 nautical miles (92.6 kilometers) and then it coasts to an apogee of 100 nautical miles (185.2 kilometers), where it is circularized. This explains the basic difference of the pressure profile from the Saturn V, which is injected into a circular orbit of 100 nautical miles (185.2 kilometers) right away.

Figure F-12 shows the accumulated boost heat transfer in one square foot of MLI as a function of time for the ideal boost trajectory pressure profile in NARSAM insulation. The analysis employs the pressure-dependent thermal conductivity of NARSAM insulation (Figure F-13), which is based on experimental and analytical criteria of Freeman[1].

Figure F-14 is another accumulated heat transfer curve as a function of time for one square foot of MLI, but this curve is based on extended time duration in orbit and the pressure profiles of Figure F-11 inside the EOS orbiter. The lowest curve, for 300 ft^2 (27.87 m^2) vent area is very close to the ideal performance which would be achieved if the insulation exactly followed the altitude pressure profile for the orbiter. Another way of looking at the boost heating is that for helium purge the heat input is approximately the equivalent of 2.2 days of orbital operation. If the final topping were cut off 90 seconds ahead of launch, about 2.0 more days of equivalent orbital heating would be imposed in addition. For nitrogen purge the boost heating is equivalent to only about one half a day in orbit and a topping cutoff at 90 seconds prior to launch would add less than 1/2 day more. The 20 ft^2 (1.858 m^2) vent area curve increases the first day's heating in orbit by about 50 percent over the ideal evacuated MLI performance and will require more than a week to achieve even a reduction to a 20 percent increase over the ideal MLI performance.

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[1]Freeman, B. Test Report High-Performance Insulation Research, NARSAM Composite, Liquid Hydrogen Boiloff From a Guarded Colorimeter. North American Rockwell Space Division, SD 70-411.

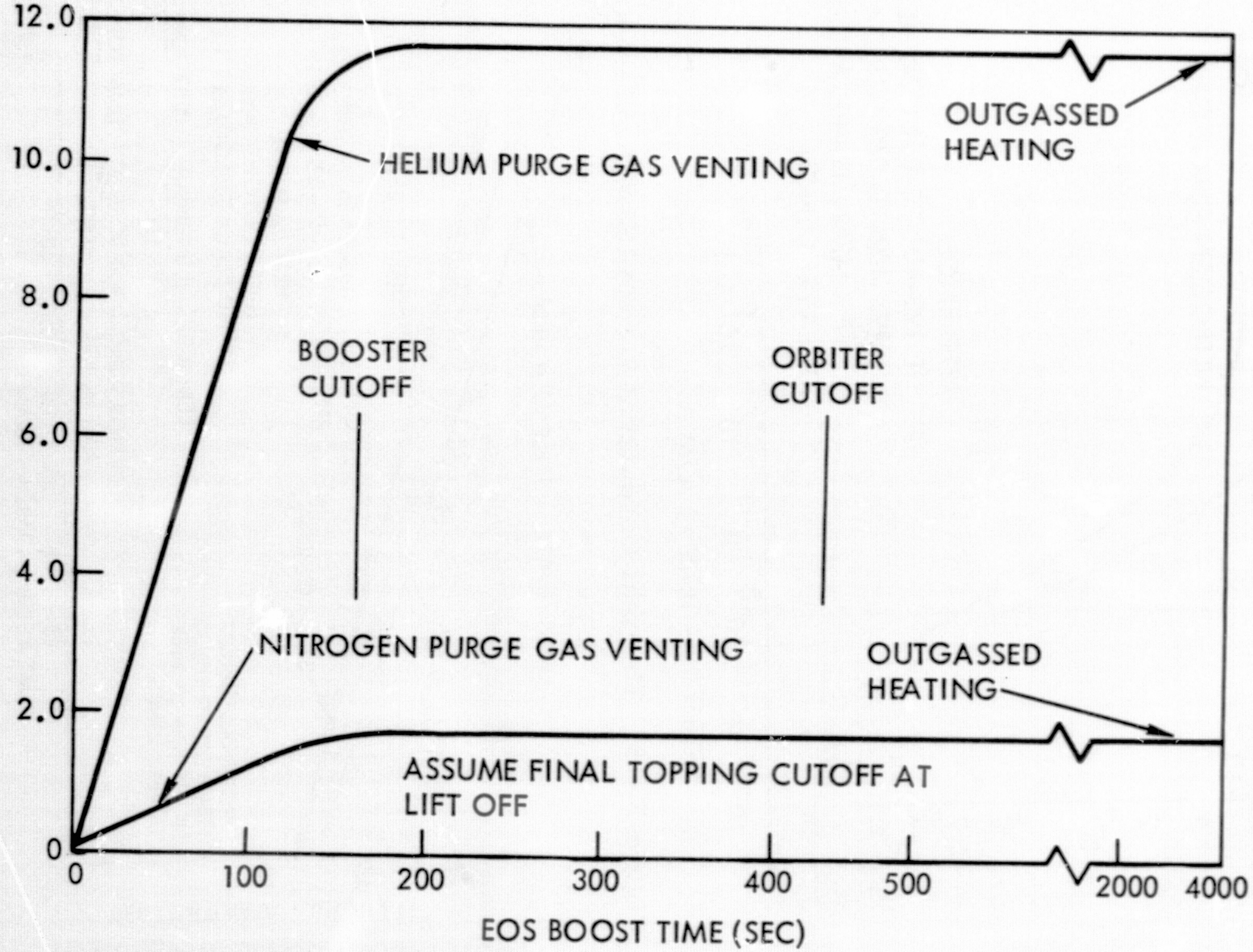


Figure F-12. Accumulated Boost Heat Transfer as Function of Time

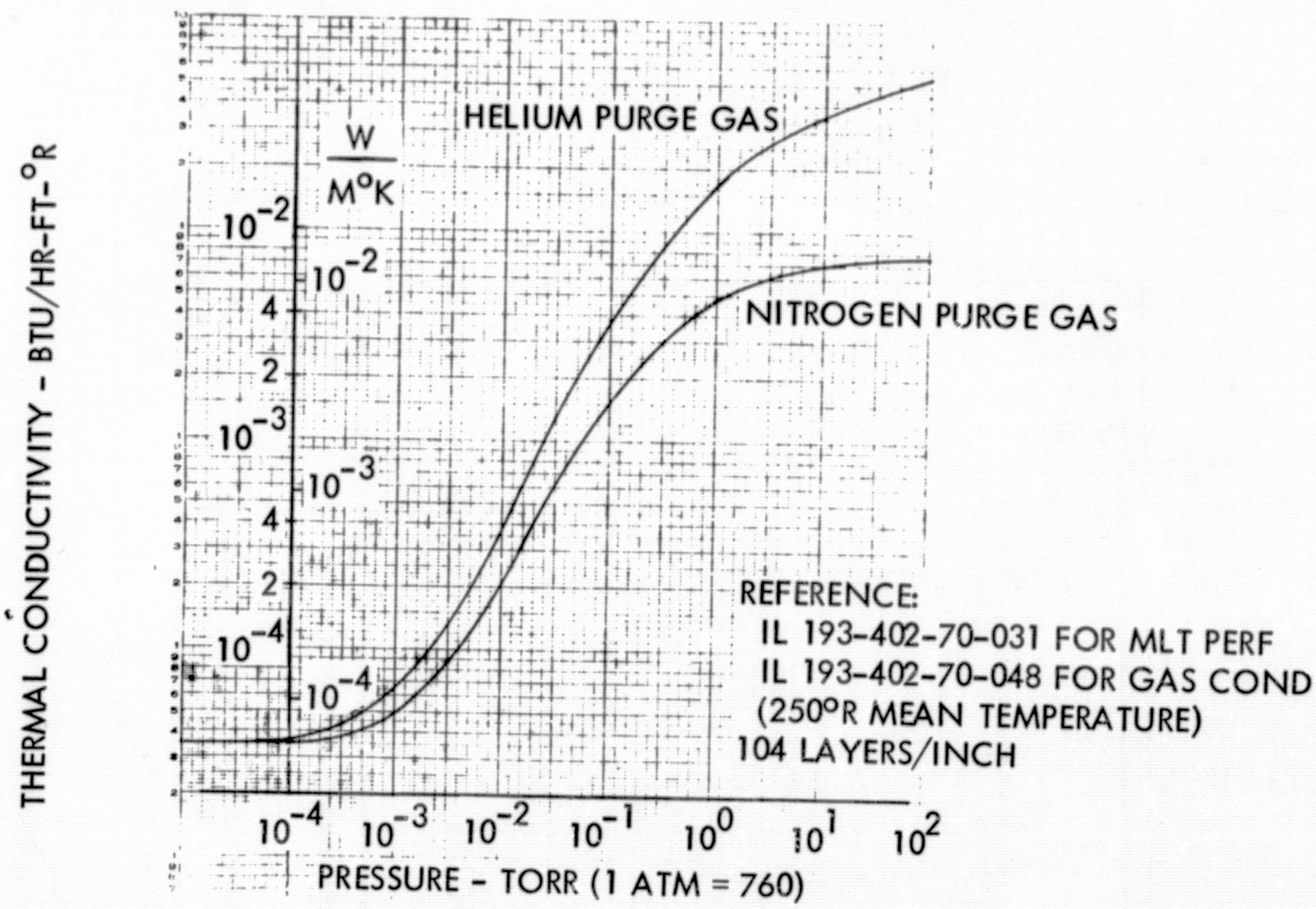


Figure F-13. Thermal Conductivity of NARSAM Insulation

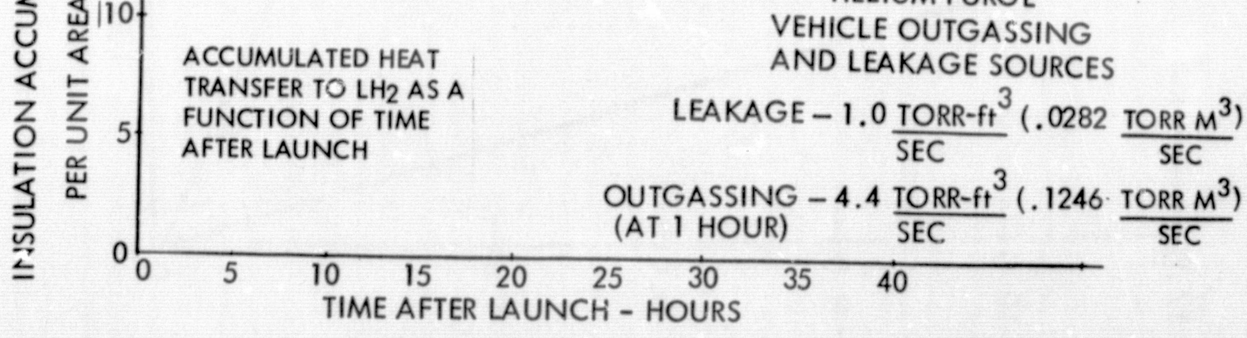


FIGURE BASED ON SHUTTLE VENTING CURVE I.L. 193-402-70-038, FIGURE 3 FOR AFT COMPARTMENT OF SHUTTLE ORBITER INCLUDES OUTGASSING AND LEAKAGE CONTRIBUTIONS

Figure F-14. Accumulated Heat Transfer in LH_2 After Launch

The previous analysis has been shown to point out how critical proper venting is to the performance of space evacuated MLI. It is not an exact analysis because of the extensive assumptions which have been made on the EOS orbiter outgassing, leakage and vent orifice provisions. The important pressure is the average pressure within the insulation and that requires a knowledge of the venting annulus and vent valve pressure drop as well as that for the insulation itself. Figure F-15 shows the differential pressure in the annulus for a typical preconditioned insulation of the NARSAM composite. The outgassing rate is assumed for one hour after launch, and the design objective is to achieve a differential pressure of 10^{-4} torr. Thus the venting annulus depth would be about 1.5 inches (3.81 centimeters), and the vent valve area about 7 square feet (6503 square centimeters) for each vent valve on the hydrogen tank. The pressure drop across the insulation must be included when determining the average pressure within the MLI.

The analytical methods and experimental data are still being developed for the exact treatment of insulation venting. The design approach to NARSAM insulation has been to include vent holes in the embossed layers which will promote outgassing. Based on the foregoing work (Figures F-14 and F-11) it appears necessary to reach a pressure differential of 5×10^{-4} torr across the insulation (shown as the criteria for rough sizing of ΔP on Figure F-8). The current NARSAM design provides for 1 percent porosity (hole area as a percent of total area for each layer) but some

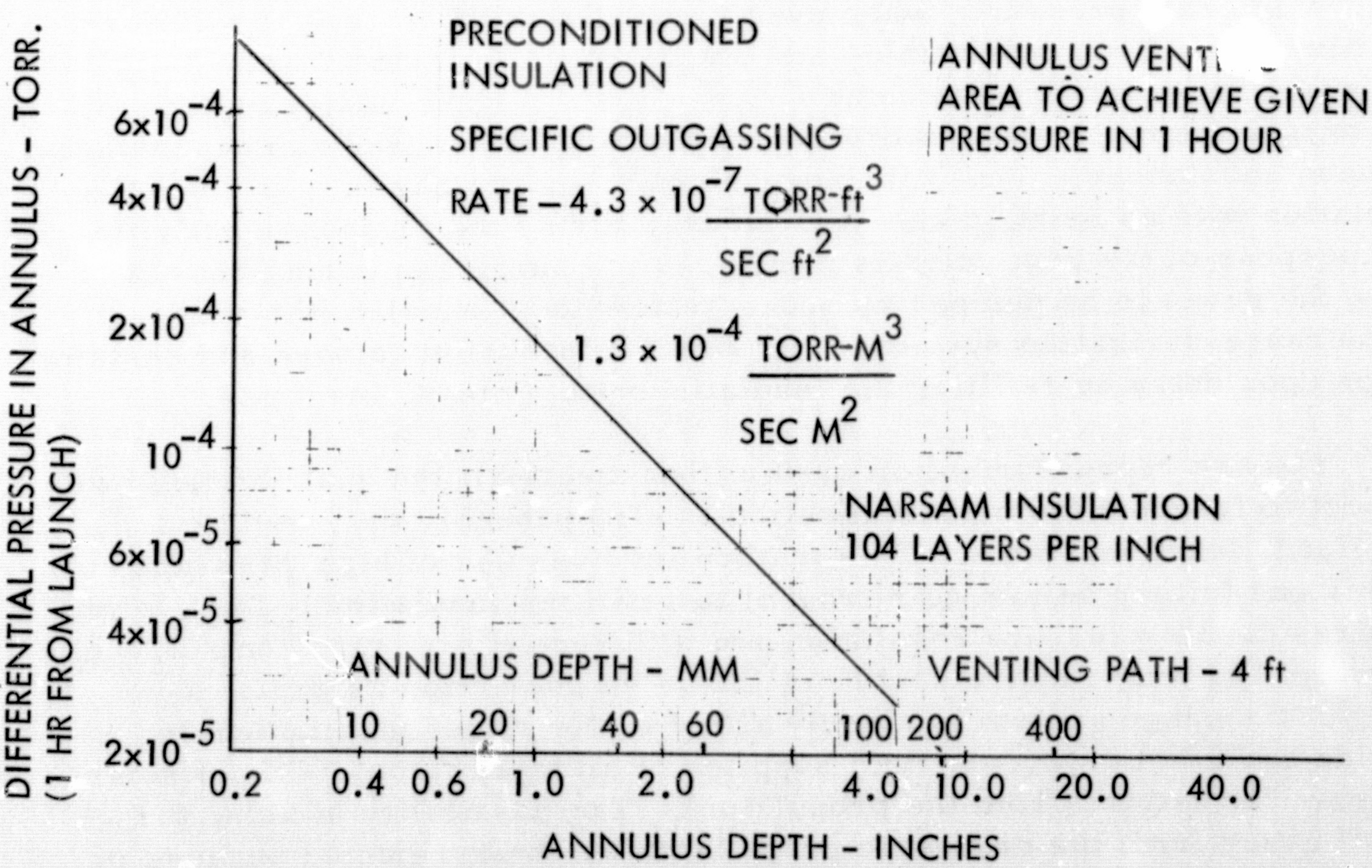


Figure F-15. Differential Pressure

analyses have indicated that 2 percent porosity may be better for insulation that is less well preconditioned. The foam substrate of the nitrogen-purged insulation was actually tested with the NARSAM composite (SD 70-441) and shown not to be a significant contributor to outgassing under cryogenic conditions. In a few hours after evacuation, the differential temperature across the foam is less than 10 F (5.55 K) making it almost LH_2 temperature.

INSULATION REUSE, GROUND BASE

To make a ground-based tug cost effective, the first analyses have indicated that it is desirable to reuse the MLI for each flight. While the insulation may only be 10 to 15 percent of the primary tank structural weight, it can be more than 30 percent of the cost. The alternative of replacing the insulation for each flight will adversely influence insulation cost, but also rapid turnaround time, which could require more vehicles in the end.

The real problem of the soft shell insulation (as compared with the hard shell or Dewar design) is that metallized plastic films appear to be degraded by water and other atmospheric contaminants. The problem for insulation materials on the ground-based tug is made even more severe by entry heating and boost vibration effects. The MLI maintains its low heat conduction by having low metallized surface emittances (high reflectivity) and low internal pressures when evacuated. Any substance that will corrode the films or add to condensables and outgassing is a contaminant. Both dust and water vapor qualify as contaminants. Even carbon dioxide, nitrogen, and oxygen from air may qualify if cryopumping has caused any accumulation of these substances. Thus, the purpose of the purge bag is to protect the insulation with an inert, clean, gas blanket during the prelaunch activities. The purpose of the vent valve is to release the purge gas during boost and allow the gases to be pumped by space evacuation. Finally, the purpose of the repressurization system is to avoid compression damage and contamination upon entry by refilling the insulation with a clean, inert gas.

The one repressurization system that meets all the qualifications with respect to freedom from contamination is also probably the heaviest and, therefore, least desirable. This method involves use of high-pressure bottled gas (either helium or nitrogen) to refill the insulation. The system requires double pressure regulation and differential pressure control, while the purge bag may require double relief-valve protection as well. A cryogenic storage system is closely allied to the stored gas method to provide gassified cryogenic fluid supplies of nitrogen (from ECS perhaps) and possibly helium (from the propulsion). The possibility of using the boiloff gases from the cryogenic propellants has been rejected because of

their hazardous nature. Another repressurization approach is to use air from the atmosphere to fill the cavities. This has the obvious advantage of requiring no expendables from the orbiter, and it makes the differential pressure control a directly solvable problem. For example, to repressurize the insulation only a small blower is required to force the air into the insulation with a small differential pressure. It must be recalled here that an underbearing foam or other device must be incorporated into the tank to prevent cryopumping of the air. Further, it is necessary to separate the components of the air, which might contribute to contamination and degradation of the layers of superinsulation.

Water vapor and dust from air can be separated by filtration methods and a desiccant pack (or refrigerative means). The filtration technique is straightforward and is not a problem, but the following discussion will center on the water vapor. The water vapor must be removed because if it condenses on the aluminized foils, it either causes corrosion or lifts the vapor-deposited aluminum from the surface. The purpose of the water vapor separator is to reduce the dew point of air entering the insulation below -90 F (205 K). The use of a cryogenic heat exchanger to condense the contaminant gases is a direct solution. For example, a liquid-nitrogen-filled heat exchanger, incorporating energy conservation between the inlet and exit streams, should be able to lower the dew point below -200 F (144 K). Furthermore, it is able to separate most of the carbon dioxide and hydrocarbon contained in the pressurizing air. The expenditure of nitrogen (or other coolant) is related to the weight and heat of vaporization of water and other gases condensed. Another solution is to use a desiccant pack to dry the incoming air. This method of drying the air is probably the most ancient but yet the most appropriate approach. The potential desiccants include silica gel, activated alumina, anhydrous calcium sulphate, and magnesium perchlorate, as well as the standard sodium hydroxide and calcium chloride. Current technology has produced some desiccants that absorb almost one-half of their own weight in water. Some have the capability of achieving a dew point of almost -100 F (200 K). For the repressurization method, the desiccant pack should weigh on the order of 10 pounds, including the filter.

Figure F-16 describes a typical repressurization system based on the use of a desiccant pack and a vane axial blower. The desiccant pack is designed to provide the -90 F (205 K) dew point and up to five sea level volume changes of air in the insulation and annulus, allowing a substantial amount for leakage in the vent valve and fittings. The vane axial fan should provide for a differential pressure of about 10 inches (25.4 centimeters) Hg at sea level in a low flow condition. A question was raised, however,

SELECTION:

AIR PRESSURIZATION SYSTEM

POSITIVE CLOSING VENT VALVES
VANE-AXIAL FAN - LOW ΔP, HIGH FLOW
DESICCANT PACK ⟶ -90 F DEW POINT

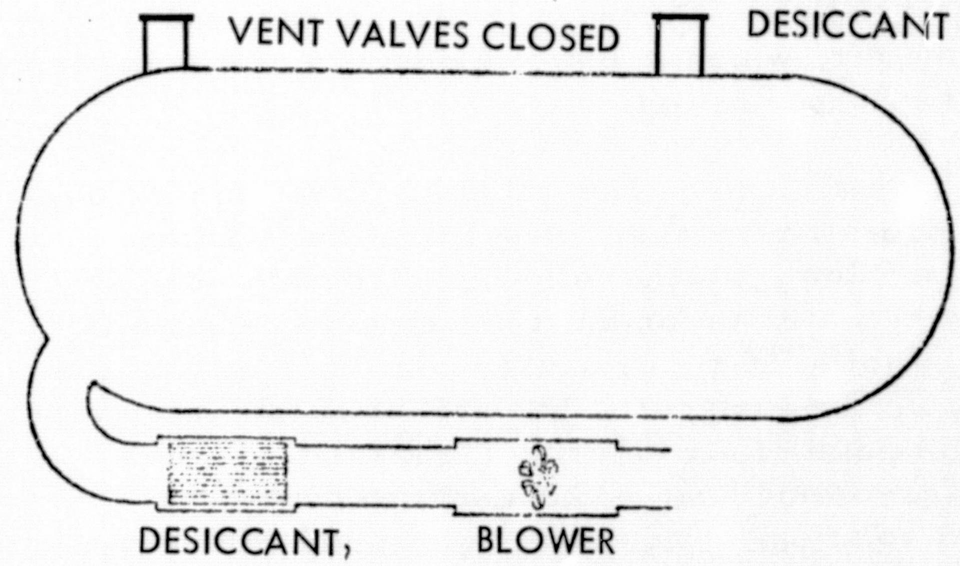


Figure F-16. Ground-Based MLI Reuse

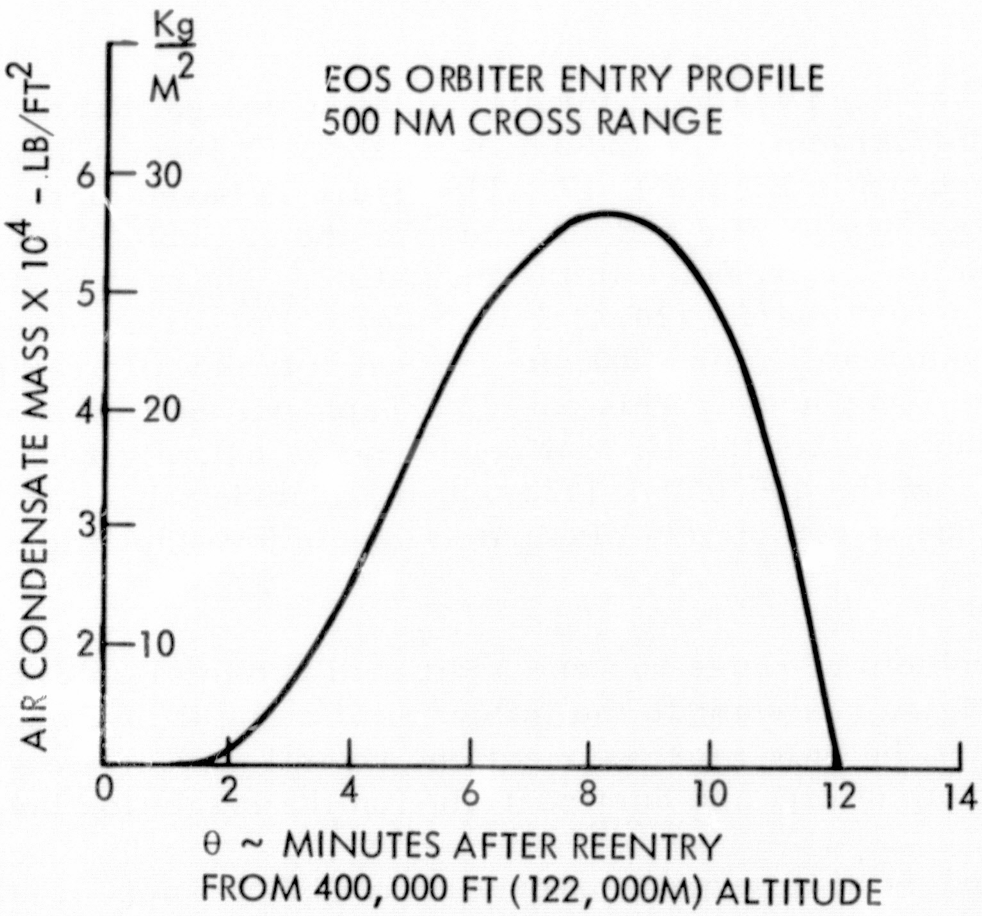
concerning the possibility of air condensation. It was suggested that the use of air repressurization is not suitable for a helium purged insulation unless an effective means can be developed for dumping all cryogenic propellants prior to return to the lower atmosphere (e.g. stratosphere). If the hydrogen is not dumped, air condensation will take place and, as the tank warms up, this will fractionally distill, leaving higher concentrations of oxygen in the residual fluid. The evaporation of one pound of hydrogen is associated with the condensation of about one pound (0.45359 kilograms) of air, and five pounds (2.268 kilograms) of air can produce about 1 pound (0.45359 kilograms) of oxygen. Since insulation materials are not generally compatible with the reactive oxygen, even the impact of a rough landing could provide the energy to start a fire or cause an explosion. The studies of emergency dumping of propellants for abort should be helpful in reducing the risk of such an event and may make it possible to use the air repressurization method for the helium-purged insulation.

For the nitrogen-purged insulation, the air repressurization problem is far less severe. Even though the foam surface is initially close to LH_2, temperature, it rapidly warms and rises above the condensing temperature.

Such a case has been analyzed for an insulation design that had a 1.5-in.-thick (3.81-cm) MLI and a 1.0-in.-thick (2.54-cm) substrate of foam. Results are shown in Figure F-17. The figure is based on the short cross-range orbiter entry profile and accounts for the variable condensing temperature as a function of pressure in the insulation. The peak of the curve shows only about 5×10^{-4} lbs/ft^2 (24.41×10^{-4} kg/m^2) of air condensed on surface of the foam substrate. For a 1000-ft^2 (92.9 m^2) tank wall area this would be only a total of 0.5 pounds. This could perhaps still be important if accumulated at touchdown, but the air that condenses is entirely reevaporated in 12 minutes from the 400,000-ft (122,000-m) altitude initiation of the entry profile and thus is completely dissipated long before the landing phase.

The problem of reuse has one additional potential solution—that of better materials, resistant to the influence of moisture condensation. For such materials the only repressurization protection would be a filter to prevent build-up of dirt and dust particles on the insulation layers.

The studies of improved materials have not yet appeared to exhaust the possibilities. For example, the use of gold metallizing to replace aluminum seems to give better results. Unfortunately the current cost estimates suggest that where 1000 square feet (92.9 square meters) of wall area could be insulated (100 layers) for about \$2000 (material cost alone), the gold metallizing may cost more than \$200,000 for the material, assuming the production quantities run in excess of a few million square feet. Another means of making the metallized coating resistant to moisture has been to incorporate clear vapor-deposited coatings of materials like germanium in controlled thicknesses. This method is complex and more costly than gold, but it may be a way to develop even better abrasion resistance as well. Such a technique has been studied by ADL (Arthur D. Little, Inc.) for use in the thermal blanket of astronauts extra-vehicular activity (EVA) garb. In addition to the reflective surface, the substrate has also been studied in an effort to provide greater temperature tolerance. At present, the payload bay appears to remain below 250 F (283 K) during entry, but the possible adaption of the high cross-range orbiter version and the use of reduced amounts of external insulation (REI) on the orbiter could create higher temperatures, which would require the Kapton plastic film to replace Mylar and achieve temperature resistance up to 500 F (533 K). This discussion is only a prelude to what kind of advancements may be made in the future. For now, however, cost is a dominant consideration, and the use of aluminized Mylar and a suitable repressurization system seems to be the most effective means of achieving reuse of the MLI for the ground-based tug.



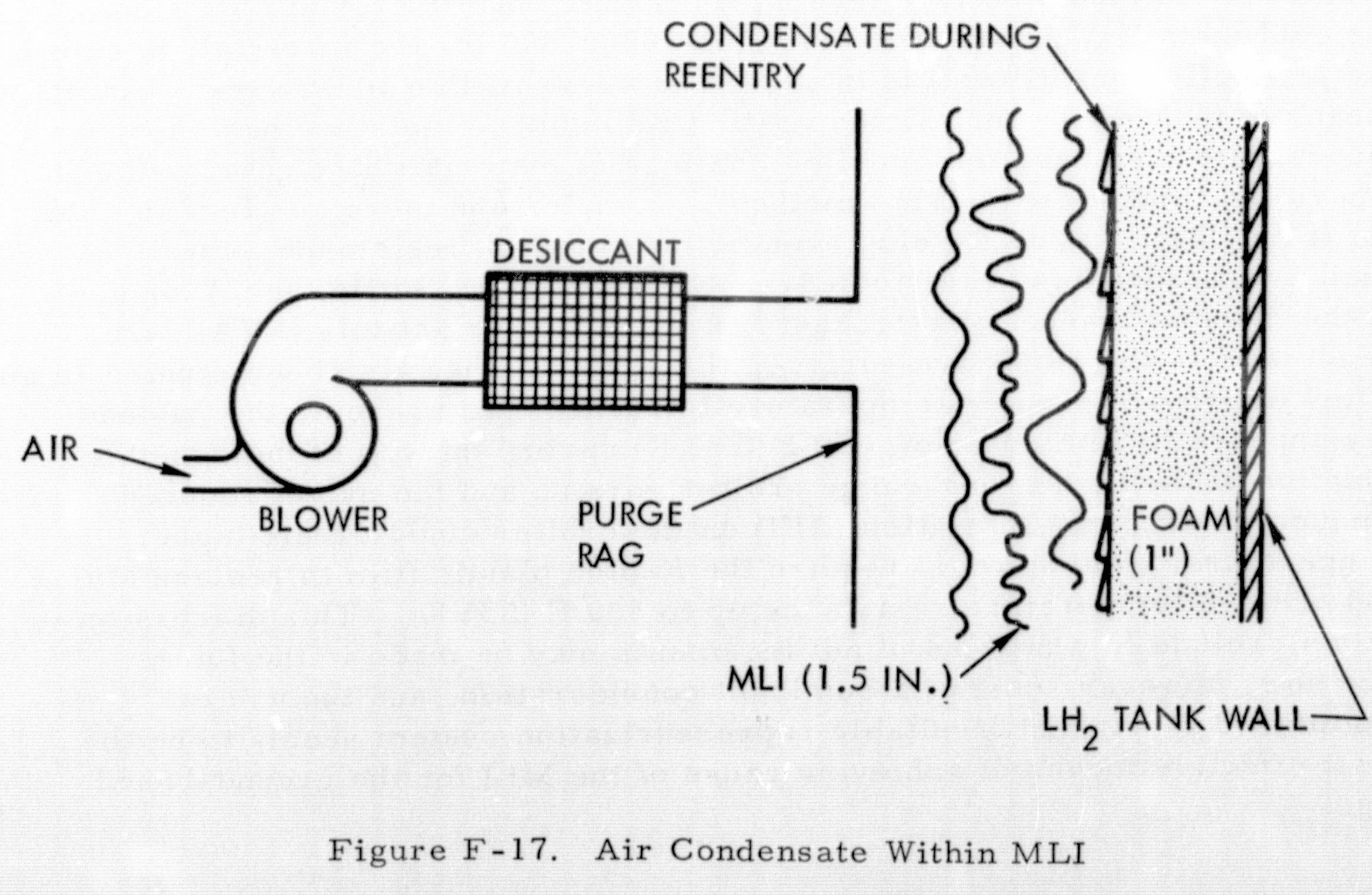


Figure F-17. Air Condensate Within MLI

HEAT TRANSFER ANALYSES

The heat transfer analyses presented in this section include propellant heating contributions from the MLI as well as the structural skirts, which are the main load-carrying members. The following analyses are divided into two main sections: thermal analysis for the 30,000-lb (13,600-kg) propellant module tug and the 60,000-lb (27,200-kg) propellant module tug (representative PM sizes). In each main section the analysis investigates various types of structure represented by the integral and nonintegral configurations, different materials such as titanium, boron epoxy, and glass epoxy for the load-carrying members, and different insulation represented by NARSAM, Superfloc, and GAC-9. The basic thermal analysis assumptions are shown in Table F-2.

The basic structural sizing assumptions for the 30,000-lb (13,600-kg) propellant module tug are shown in Table F-3. This table includes the surface areas and the structural sizing for the heat blocks. The propellant lines are considered to be small additive factors, so they have been omitted from this study.

For the five skirt configurations (Table F-3) defined for the 30,000-lb (13,600-kg) tug, Figure F-18 presents the total heat leak to the cryogen due to the skirt structural short. It can be seen that, for insulation thicknesses of greater than 1.5 inches (3.81 centimeters), the structural heat leak is rather insensitive to further addition of insulation. The skirts are insulated on both sides, with 8 inches (20.3 centimeters) being the insulation thickness per side. For simplicity of solution, it is assumed that the outer layer of insulation at all points on the skirt is at 510 R (283 K) (ambient).

Ranked in order of performance, the skirt structural configurations are:

1. Integral boron-epoxy truss

2. Integral glass-epoxy shell

3. Integral and nonintegral boron-epoxy shell

4. Integral titanium shell

Table F-2. Thermal Analysis Assumptions

A. Assumptions

1. Ambient temprature seen by outer layer of insulation both internal and external of vehicle is constant at 510 R (283 K).
2. Heat blocks are insulated forward and aft 5 feet (1.524 meters) each direction from LH_2 tank.
3. At end of insulation (5 feet) of forward skirt, T_{skirt} = 510 R (283 K).
4. Boundary temperature of aft heat block is 540 R (300 K) because of location of space radiator.
5. Because of lack of specific data for oxygen tanks, rough sizing was done, which indicated 40 Btu/hr (11.7 w) total could be expected through all LO_2 supports.

B. Calculation of Heat Rate Through Heat Block

$$\dot{Q} = K A m A \left[\text{Coth } (mL)\right]$$

$\dot{Q}$ = Total heat rate per continuous skirt due to sidewall and conductive heat transfer

where

h_r = 1/HPI thickness (HPI conductivity)

= Effective "film" coefficient for skirt insulation

$m = \sqrt{h_r \, 2\pi D/K A}$

L = Length of insulated skirt

θ = ΔT between tip and root of fin (skirt)

A = Skirt cross-sectional area (not including HPI)

D = Diameter of skirt shell

K = Conductivity of skirt shell material

C. Thermal-Physical Properties

1. Conductivity of boron-epoxy composite = 0.666 Btu/ft hr R (1.115 w/m K) at 200 R (111 K).
2. Conductivity of titanium alloy - 3.0 Btu/ft hr R (5.2 w/m K) at 200 R.
3. Conductivity of glass-epoxy composite = 0.224 Btu/ft hr R (0.387 w/m K) at 200 R (111 K).
4. K_{NARSAM} = 2.58 x 10^{-5} Btu/ft hr R, (4.48 x 10^{-5} w/m K) — T_H = 510 R (283 K), T_c = 40 R (22.2 K)

   K_{NARSAM} = 3.33 x 10^{-5} Btu/ft hr R, (5.76 x 10^{-5} w/m K) — T_H = 510 R (283 K), T_c = 170 R (94.4 K)

   ρ = 2.47 lb/ft^3 at 104 ℓ/in. (41 ℓ/cm)

Table F-2. Thermal Analysis Assumptions (Cont)

5.	$K_{SUPERFLOC}$	$= 2.80 \times 10^{-5}$ Btu/ft hr°R (4.85×10^{-5} w/m°K)	$T_H = 510$ R (283 K), $T_c = 40$ R (22.2 K)
	$K_{SUPERFLOC}$	$= 4.28 \times 10^{-5}$ (7.42×10^{-5} w/m°K)	
		$= 1.64$ lb/ft³ at 30 /in. (12 ℓ/cm)	$T_H = 510$ R, $T_c = 170$ R (94.4 K)
6.	$K_{DAM/NM}$	$= 3.07 \times 10^{-5}$ Btu/ft hr°R (5.32×10^{-5} w/m°K)	$T_H = 510$ R, $T_c = 40$ R (22.2 K)
	$K_{DAM/NM}$	$= 4.25 \times 10^{-5}$ (7.36×10^{-5} w/m°K)	$T_H = 510$ R (283 K), $T_c = 170$ R (94.4 K)
	ρ	$= 47.1$ kg/m² at 50 ℓ/in. (19.7 ℓ/cm)	

ℓ/in. = layers/inch.

For NARSAM, Superfloc, and DAM/NM, Figure F-19 illustrates the relative benefits of each insulation in terms of heat rate to the LH_2 tank through the HPI only. The heat rate to the tank through structures and HPI were kept on separate curves so the user could readily obtain the heat flux through the different HPI's and apply it to different size tanks.

It may be seen from Figure F-19 that the spread in heat rate between NARSAM (best performing) and DAM/NM varies from 19 to 23 percent over the insulation thicknesses investigated. The point of diminishing returns in the curve of heat rate occurs at roughly two inches (5.08 centimeters) of insulation, which (neglecting the effect of insulation weight) would be the point at which increases in insulation thickness failed to bring substantial reductions in heat rate. It should be noted that the idealized performance of these insulations has been used herein without allowances for joints, posts, and other support members since the actual insulation design could not be determined in detail. The surface temperature used is conservative to account for some of the discrepancy with practical designs since a surface α/ψ of 0.4 would yield orbital average temperatures nearer 440 R (245 K).

Figure F-20 presents the total heat rate into three oxygen tanks through the insulation as a function of HPI thickness. It should be noted that

Table F-3. Basic Structural Sizing Assumptions

Surface area of LH_2 tank = 507.2 ft^2 (54.5 m^2)

Surface area of each LO_2 tank = 123 ft^2 (13.3 m^2)

Total for three LO_2 tanks = 369 ft^2 (39.7 m^2)

Structural Sizing for LH_2 Heat Blocks*

Type of Structure	$\bar{t}$ (cm) (in.)	Diameter Overall (in.) (m)	Diameter Strut (in.) (m)	No. Bays n	No. Bays m	Length of Structure (in.) (m)
Integral boron-epoxy shell						
Fwd adapter	0.076 (0.193)	170.0 (4.32)	-	-	-	58.2 (1.48)
Intertank structure	0.091 (0.238)	170.0 (4.32)	-	-	-	105.7 (2.7)
Non-Integral boron-epoxy shell						
Fwd adapter	0.076 (0.193)	180.0 (4.51)	-	-	-	58.2 (1.48)
Intertank structure	0.094 (0.238)	180.0 (4.51)	-	-	-	118.7 (3.0)
Integral boron-epoxy truss						
Fwd adapter	0.045 (0.114)	170.0 (4.32)	3.56 (0.09	1	12	58.2 (1.48)
Intertank structure	0.058 (0.147)	170.0 (4.32)	3.56 (0.09)	2	12	105.7 (2.7)
Integral titanium shell						
Fwd adapter	0.068 (0.173)	170.0 (4.32)	-	-	-	58.2 (1.48)
Intertank structure	0.081 (0.206)	170.0 (4.32)	-	-	-	105.7 (2.7)
Integral glass-epoxy shell						
Fwd adapter	0.122 (0.31)	170.0 (4.32)	-	-	-	58.2 (1.48)
Intertank structure	0.159 (0.405)	170.0 (4.32)	-	-	-	105.7 (2.7)

*1. $\bar{t}$ = effective shell or strut thickness, accounting for stringers, etc.

2. At present, no similar detailed structural evaluation for oxygen tanks 30,000-lb (13,600-kg) tug.

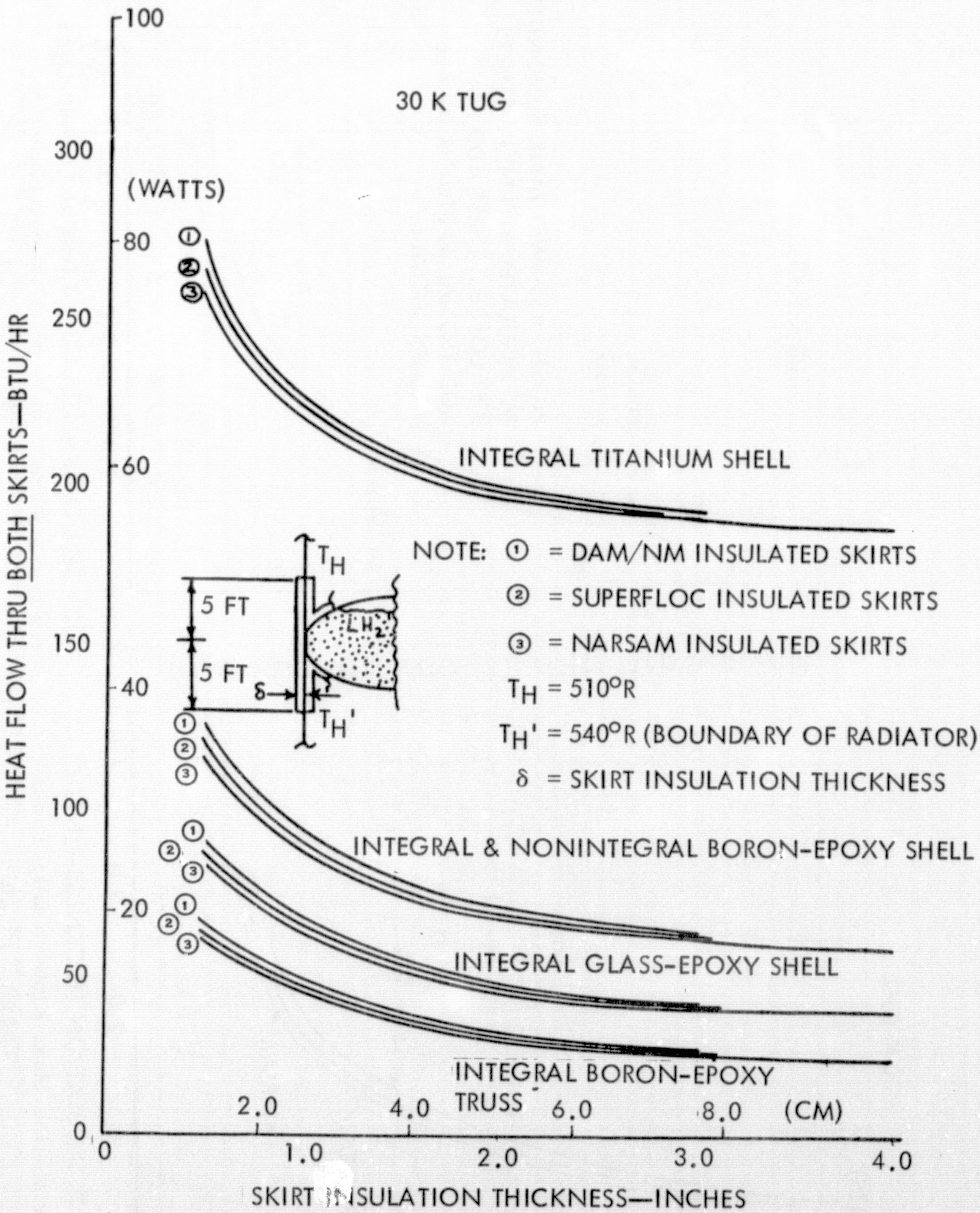


Figure F-18. Effect of Skirt Structure on LH_2 Tank Skirt Heat Leak

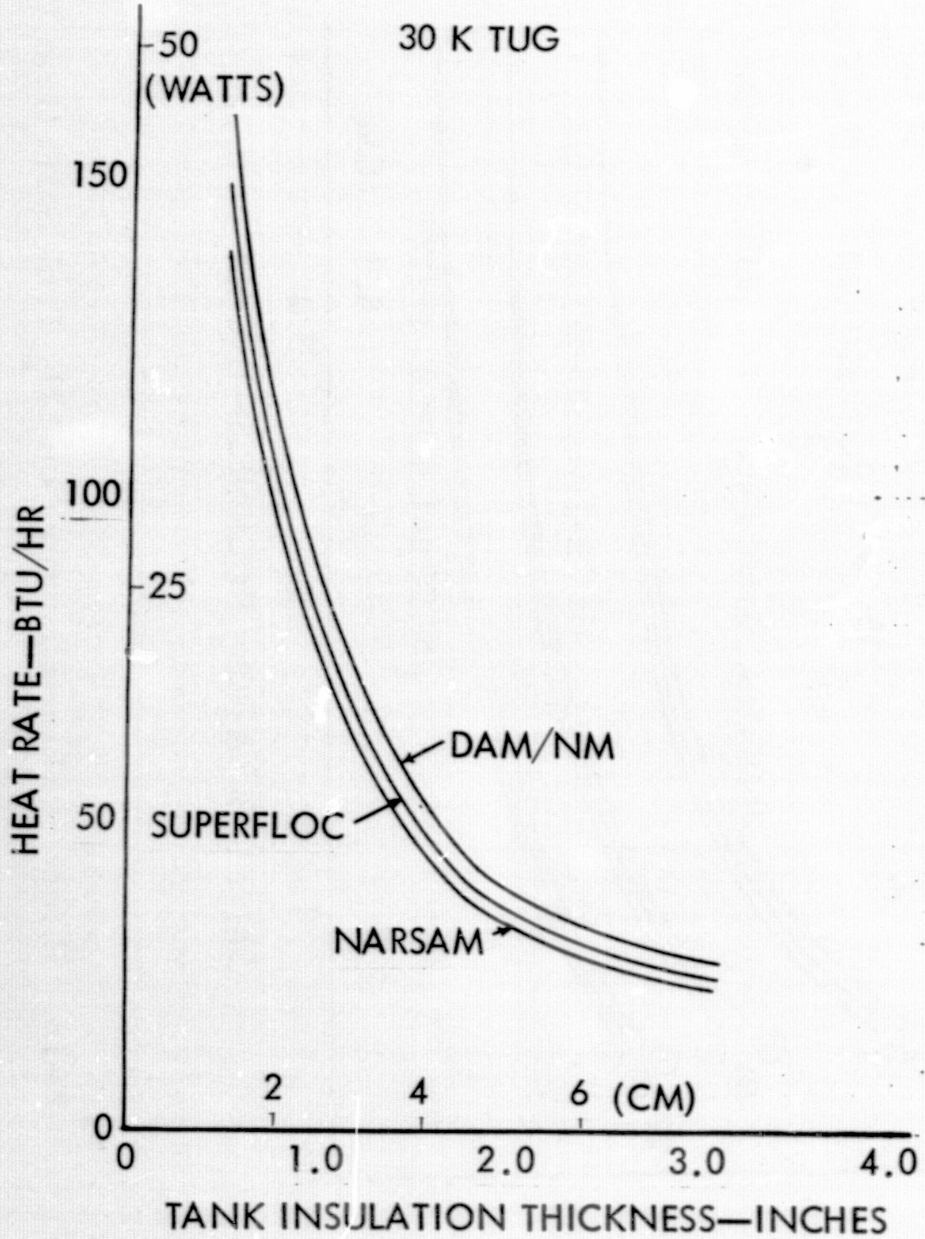


Figure F-19. Heat Flow to LH_2 Tanks Through HPI Only

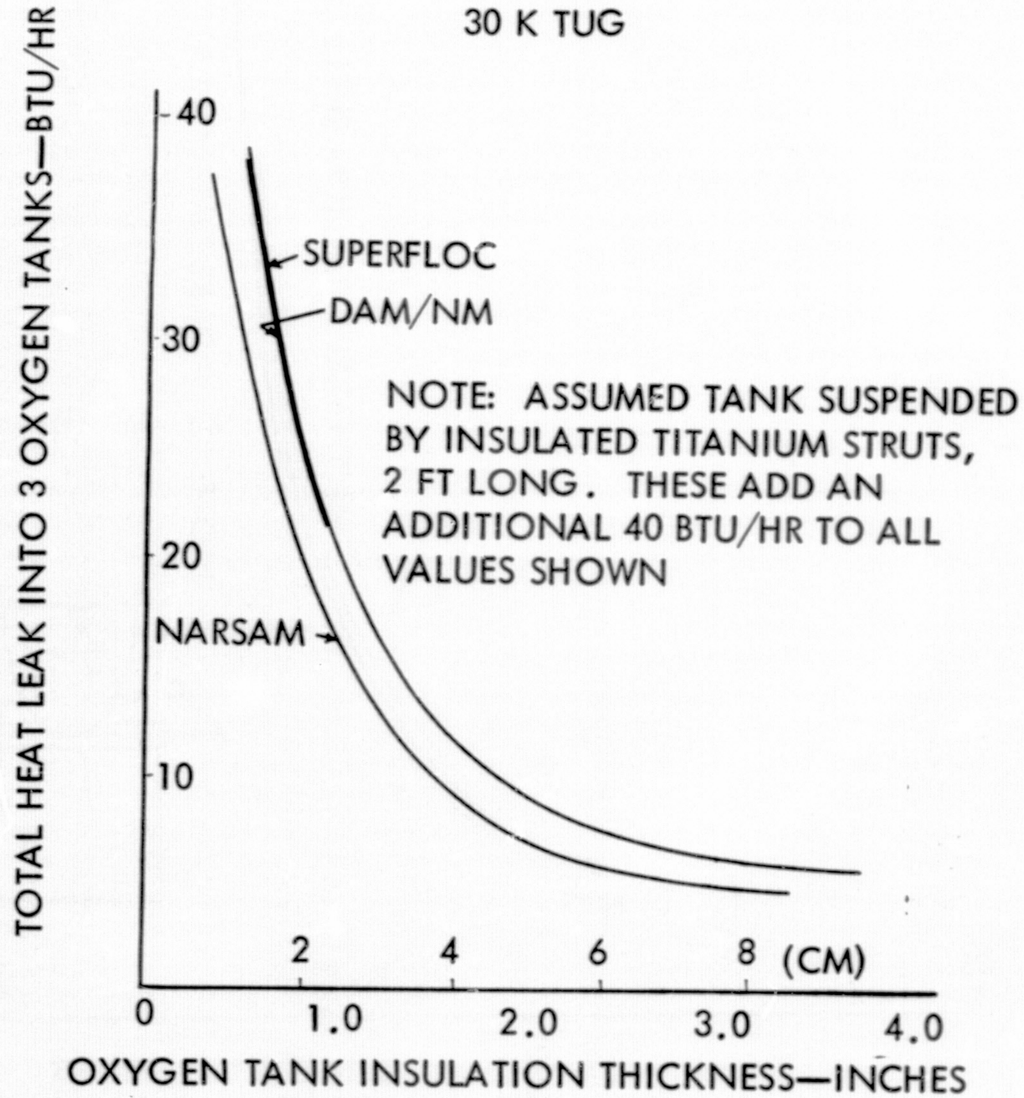


Figure F-20. Heat Flow to Oxygen Tanks Through HPI

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the superiority of NARSAM is much more clearly defined for the case of a higher cold boundary temperature (T_c = 170 R (94.5 K) for LO_2 tank). At the time of this study, no data were available on the structural support sizing for the LO_2 tanks. A rough calculation indicated that for three 2-foot-long insulated titanium struts per tank, a total of 40 Btu/hr (11.7 W) should be added to the heat rate through HPI to account for structural penetrations.

The plot of weight penalty versus insulation thickness (shown in Figure F-21) gives a feeling for the relative benefit of adding insulation weight to decrease boiloff weight. The term weight penalty defined here simply means the arithmetic sum of total insulation weight and boiloff weight over a given mission duration.

Points to note are that one may use less insulation than the 1.5 inches (3.8 centimeters) specified from Figures F-18 and F-19 because the slight increase in boiloff is overcompensated by a decrease in total insulation weight. As little as one-half inch of insulation may be used without incurring a significant weight penalty for the seven-day mission; however, thicknesses this small are usually avoided because of installation difficulty. Furthermore, this curve does not include plumbing heat leaks.

It should also be noted that although one-half inch (1.24 centimeters) of insulation may be fine for short duration missions, it may incur a significant weight penalty for long duration missions. (See curve for 45-day case.) Hence, it is best to pick a median (1.5 inches, 3.8 centimeters), which incurs a minimum penalty for both maximum and minimum mission durations. Another point to note is that this is not a true optimization since a very high delta-V mission, such as the synchronous equatorial mission, may have a payload penalty of up to 10 pounds (4.451 kilograms) for each pound of fixed weight, which would indicate a much smaller thickness is still assumed to be a good compromise for a multimission tug capability.

Figure F-22 gives the same information for the oxygen tanks as Figure F-21 did for hydrogen. A compromise insulation thickness is indicated for the seven- and 45-day missions, which overlap in an area that might be considered the design point (2.0 to 2.5 inches, 5.08 to 6.35 centimeters).

It should be noted that the curve for Superfloc is "flatter" than that of NARSAM. In fact, the two curves actually intersect. This is because the density of Superfloc is 50 percent less than that of NARSAM. Hence, for a given insulation thickness, the HPI weight contribution to the weight penalty will be 50 percent less for Superfloc. NARSAM's superior conductivity is what makes it competitive at lower insulation thicknesses.

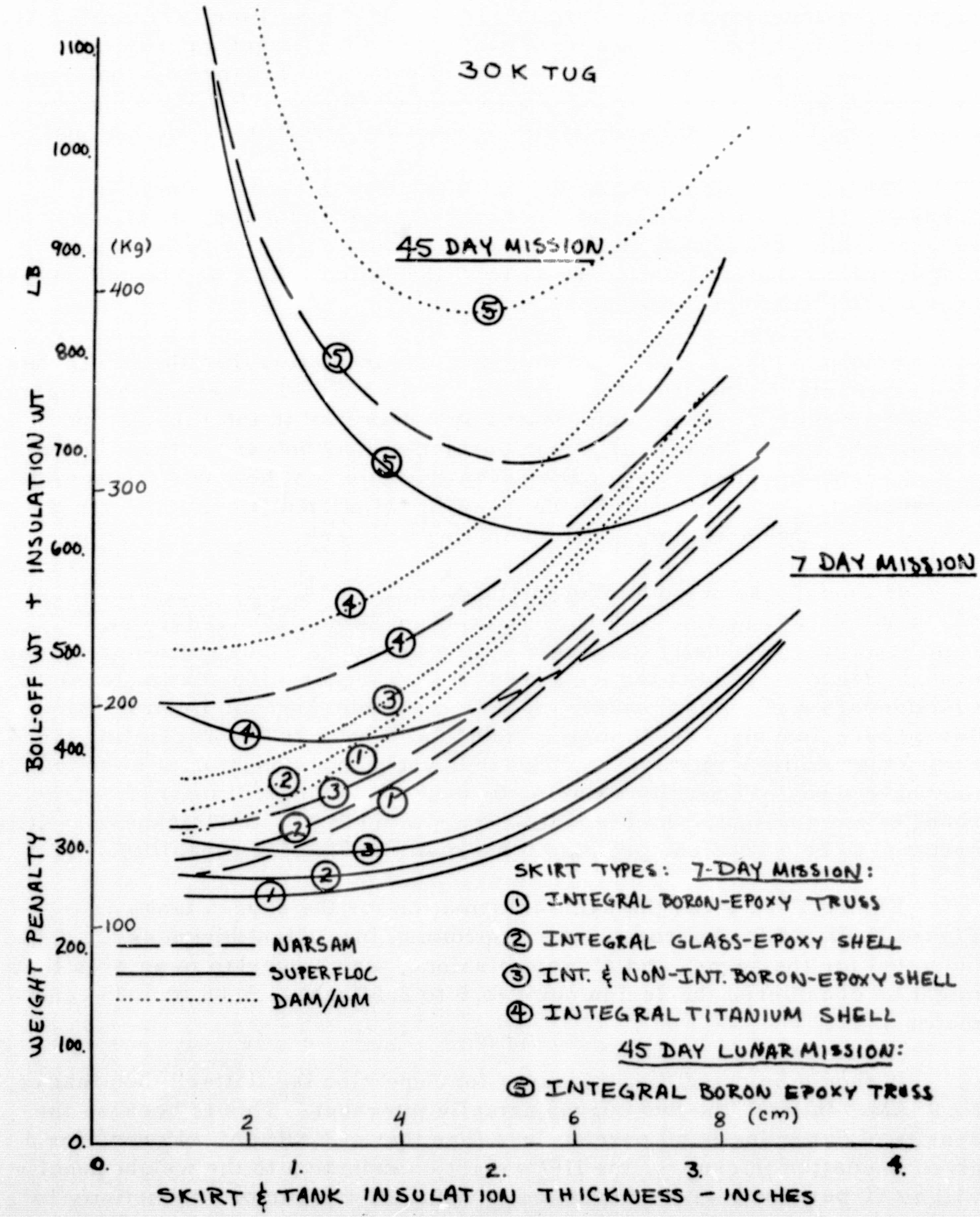


Figure F-21. Hydrogen Tank Weight Penalty Versus Insulation Thickness

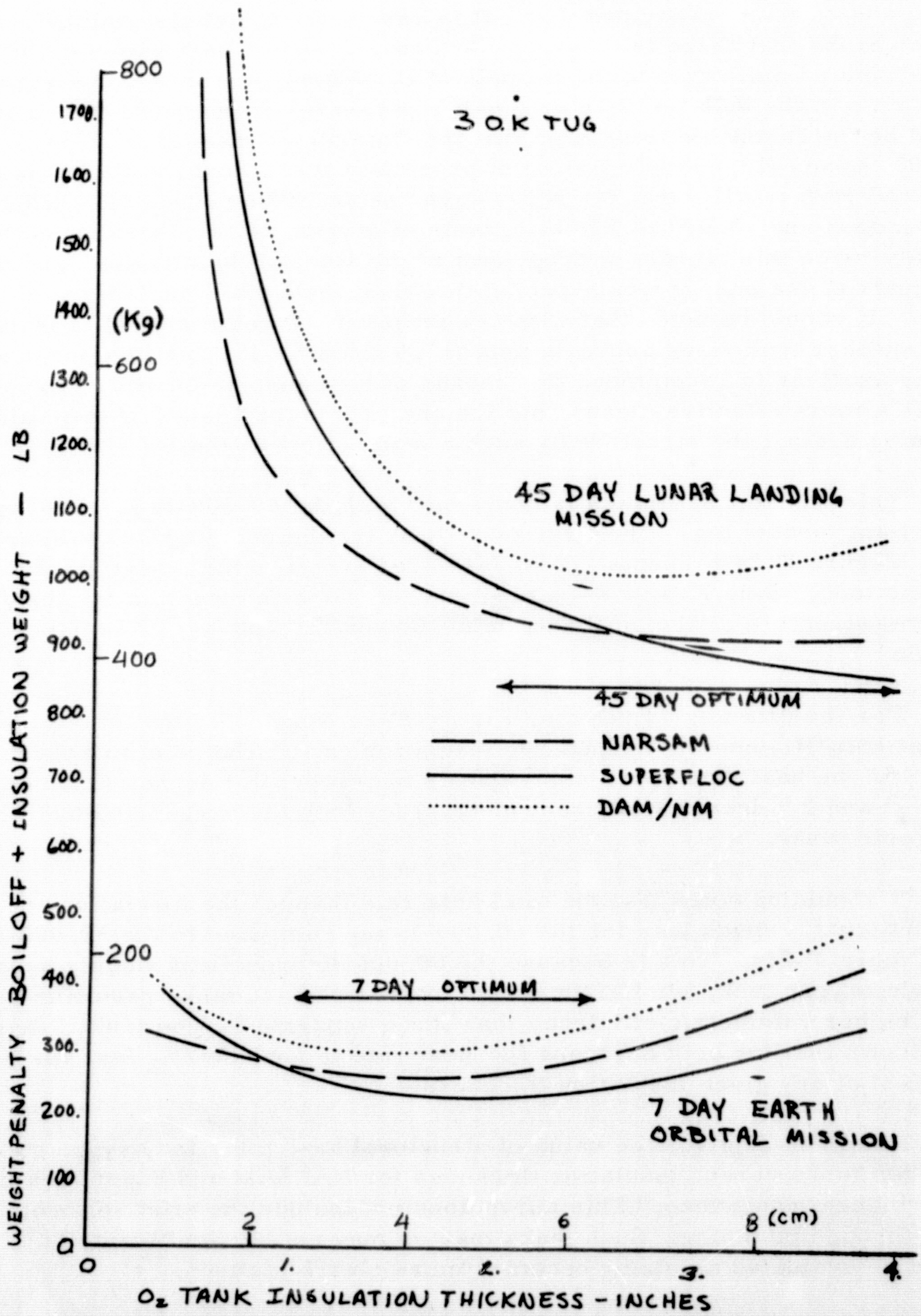


Figure F-22. Oxygen Tank Weight Penalty Versus Insulation Thickness

If the cryogen tanks were initially at a saturated vapor condition at 20 psia, Figure F-23 illustrates what the maximum tank pressure would be at the end of a seven-day mission. It is assumed for both the hydrogen and oxygen tanks that there is no venting and that all of the heat entering the tank is distributed uniformly to the propellant by destratification and reflected in a net rise in the bulk temperature (vapor pressure) of the fluid. Since it would be unreasonable to assume that the fluid sink consisted of 30, 000 pounds (13, 600 kilograms) of propellant throughout the flight, it was assumed (arbitrarily) that for seven days one had 2000 pounds (910 kilograms) of hydrogen and 10, 000 (4, 550 kilograms of oxygen. Since there would be roughly twice this amount of propellant at the start of the mission, and only residuals at the end, it was expected that this would yield an "averaged" value. It should be noted that with two inches (5. 08 centimeters) of insulation, neither tank rises above 26 psia (17. 9 N/cm^2. It is not the purpose of this analysis to recommend this means of heat absorption since evaporation is a more effective means, but the analysis helped establish penalties accruing against the structure by such a mode of operation.

This next set of studies is concerned with the 60, 000-lb (27, 200-kg) propellant module tug. The first set will be for the nonintegral configuration. Figure F-24 presents the surface area and structural heat block assumptions, while Figure F-25 presents the same information for the integral structure design of the same 60, 000-lb (27, 200-kg) propellant module.

For the nonintegral vehicle, Figure F-26 represents the total amount of heat (Btu/Hr) coming through the insulation only. The point of reduced return in the heat rate curve corresponds to between 1. 5 inches (3. 82 centimeters) and 2. 0 inches (5. 08 centimeters) for both the oxygen and hydrogen cryogenic tanks.

It should be noted that the total heat rate through the insulation is proportionately much less for the 60, 000-lb tug compared to the 30, 000-lb tug (Figure F-20). This is because the 60, 000-lb propellant module uses a single oxygen tank, similar in design to the hydrogen tank, resulting in a much higher volumetric efficiency than three separate oxygen tanks. Also, it is shown that for both cryogens the heat rate is consistently less for NARSAM at any given insulation thickness.

Figure F-27 gives the value of structural heat rates for oxygen and hydrogen tanks versus insulation thickness for NARSAM and Superflock high-performance insulations. This curve does not include the effect of heat leaks through plumbing. Such leaks may or may not be significant and should be evaluated as sizing becomes more clearly defined.

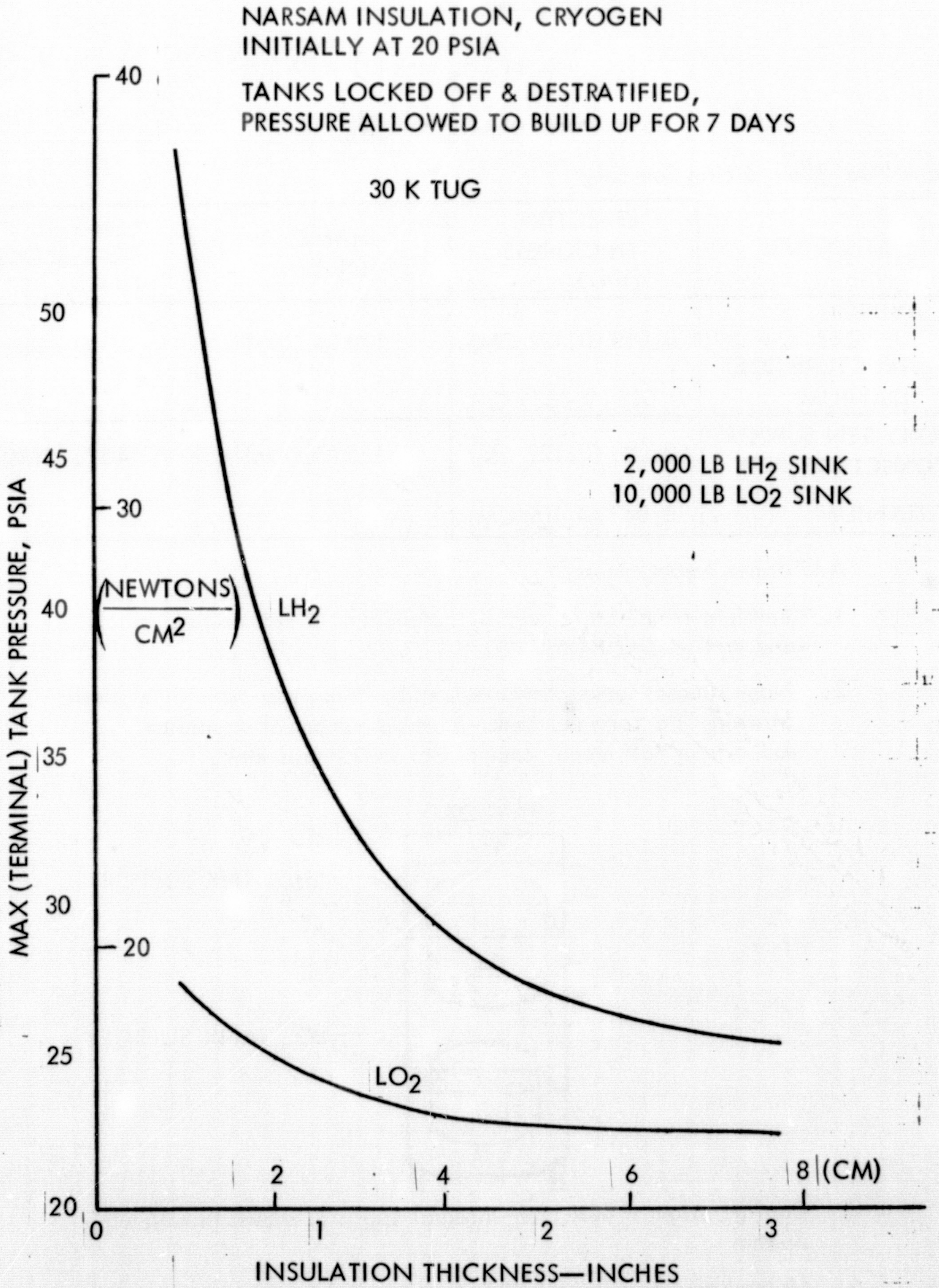


Figure F-23. Pressure Rise for 7-Day, Zero Vent Mission

Structural Sizing

Surface area of LH_2 tank (1) = 1075 ft^2

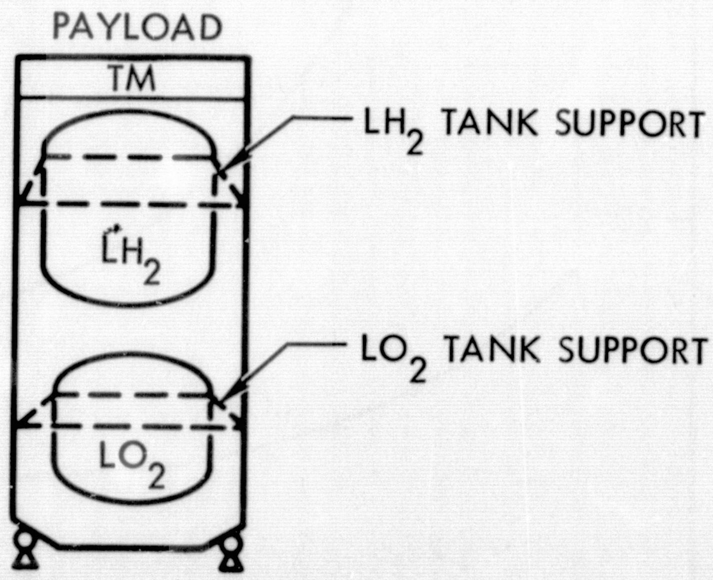
Surface area of LO_2 tank (1) = 476 ft^2

Tank Heat Block Sizing Summary

STRUCTURE	EFFECTIVE THICKNESS (IN.)	DIAMETER (IN.)	LENGTH OF HEAT BLOCK (IN.)
HYDROGEN SUPPORT STRUCTURE (B/E)	0.048 (0.122 CM)	170 (4.32M)	25 (0.636M)
TITANIUM	0.025 (0.0636 CM)		
OXYGEN SUPPORT STRUCTURE (B/E)	0.048 (0.122 CM)	155 (3.94M)	40 (1.01M)
TITANIUM	0.043 (0.109 CM)		

Additional Assumptions:

1. Ambient temperature seen by outer layer of HPI on heat block and tank is 510°R (283°K).
2. Non-integral tanks characterized by being supported by a single frustrum of a cone per tank - cutting number of structural supports in half when compared to an integral tank.



3. Configuration of 60K Non-Integral Tug is as shown in following sketch:
4. Method of determining heat rate through skirts and heat blocks, as well as thermal physical properties of materials remain unchanged over those referenced for 30K Tug (Figure F-18).

Figure F-24. Structural Sizing for the 60K Pound Propellant Module Tug Nonintegral Tank

Structural Sizing

Surface area of LH_2 tank (1) = 1065 ft^2 (115M^2)

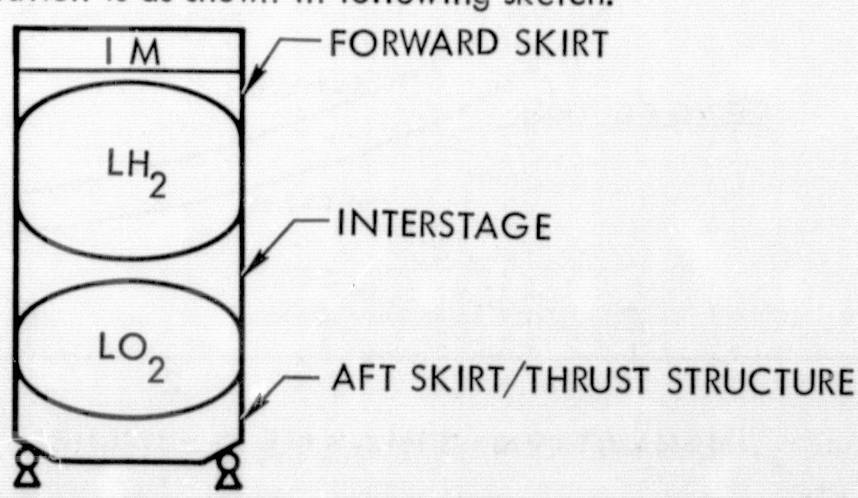
Surface area of LO_2 tank (1) = 597 ft^2 (646M^2)

Tank Heat Block Sizing Summary

STRUCTURE	EFFECTIVE THICKNESS (IN.)	DIAMETER (IN.)	LENGTH OF HEAT BLOCK (IN.)
FORWARD HYDROGEN SKIRT (B/E) TITANIUM	0.048 (0.122 CM) 0.034 (0.086 CM)	170 (4.31M)	58 (1.47M)
HYDROGEN-OXYGEN INTERSTAGE (B/E) TITANIUM	0.048 (0.122 CM) 0.028 (0.071 CM)	165 (4.18M)	133 (3.48M)
OXYGEN TANK AFT HEAT BLOCK (B/E) TITANIUM	0.048 (0.122 CM) 0.025 (0.0635 CM)	160 (4.06M)	40 (1.01M)

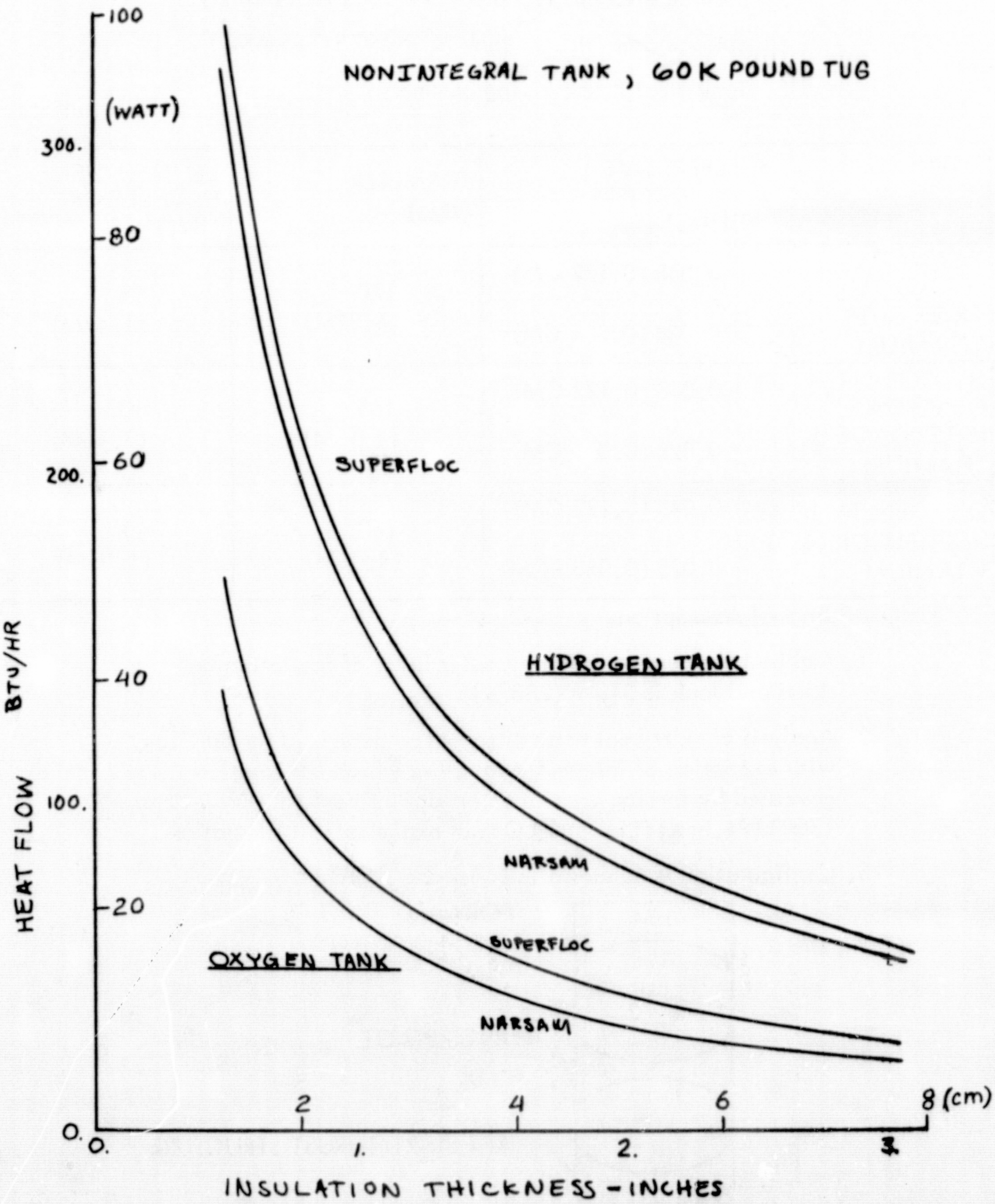
Additional Assumptions

1. Ambient temperature seen by outer layer of insulation on skirts and tank is 510°R (283°K)
2. Analysis of skirt heat rates proceeds as described for 30K Tug, with exception of interstage structure which has solution governed by forcing ends of interstage to be 40°R (22.2°K) and 170°R (94.5°K) (LH_2 and LO_2) in mathematical simulation.
3. Configuration is as shown in following sketch:



4. Thermal physical properties of materials remain unchanged from those referenced for 30K Tug (Figure F-18).

Figure F-25. Structural Sizing for the 60K Pound Propellant Module Tug Integral Tank



* DOES NOT INCLUDE SUPPORTS, PLUMBING, ETC.

Figure F-26. Heat Flow to Cryo Tanks Through HPI Only

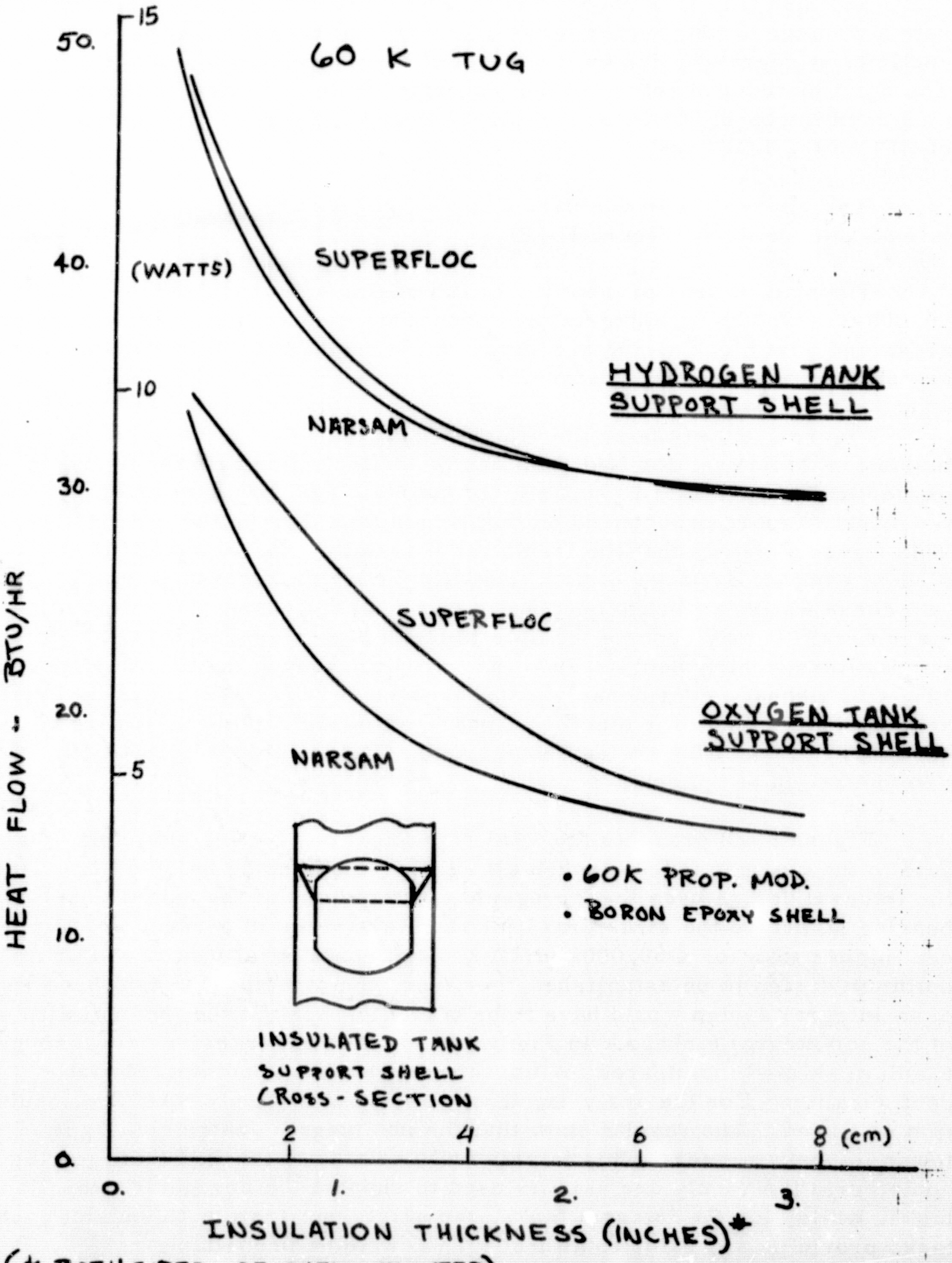


Figure F-27. Heat Flow Through Insulated, Nonintegral Support Structure

As expected, it is shown that NARSAM gives a relatively larger heat rate advantage on the oxygen structure. This is because NARSAM's conductivity is degraded less by the elevated oxygen cold-side temperature. The point of reduced return of the heat rate curves appears to be one inch (2.54 centimeters) of HPI on the LH_2 tank and 1.5 inches (3.8 centimeters) of HPI on the LOX tank.

It is interesting to compare Figure F-27 with the structural heat leak into an integral tank (Figure F-28).

The next figures present the heat transfer analyses for the 60,000-lb (27.200-kg) integral tank propellant module tug. The structural areas and heat block sizing are shown in Figure F-25. The other assumptions are similar to those used before.

Figure F-29 presents the effect on tank heat rate of varying the insulation thickness, for NARSAM and Superfloc. For both the hydrogen and oxygen tanks, NARSAM is superior in maintaining a low heat rate. Again the effect is more pronounced on the oxygen tank than on the hydrogen tank. This figure assumes that the insulation hot-side temperature is 510 R (283 K) over all surfaces of both tanks (including the interstage void). This is a conservative estimate because of a lack of definition of the actual interstage area. It may become feasible to let this area approach LOX temperature, which would reduce the heat flux through the LOX forward and LH_2 aft bulkheads considerably. However, if it is desired to use this rather considerable volume for electronic gear, or instrumentation, it may be necessary to condition the environment to a higher temperature to allow safe operation of the components.

Figure F-28 presents the total structural (neglecting plumbing) heat rate to the cryogen tanks through all supports. It should be noted that since the integral design uses the cryogenic tanks as part of the vehicle load-bearing structure the structural heat rates are from 50 percent to 100 percent higher than for a 60,000-lb (27.200-kg) nonintegral vehicle. For an optimum design to be established, however, the structural weight benefits of an integral design would have to be traded off against the thermal efficiency of the nonintegral tankage. In Figure F-30, the heat transfer rate through titanium skirts is compared for both the integral and nonintegral tank configuration. For the analysis, it was assumed that only NARSAM insulation was used. The results show that the nonintegral design clearly is lower in heat transfer. This is primarily related to the fact that, for the nonintegral tank, only one skirt is used to support the cryogenic propellant tanks, while, for the integral tanks, the skirts penetrate both ends of each tank, providing a greater total heat transfer to the propellant.

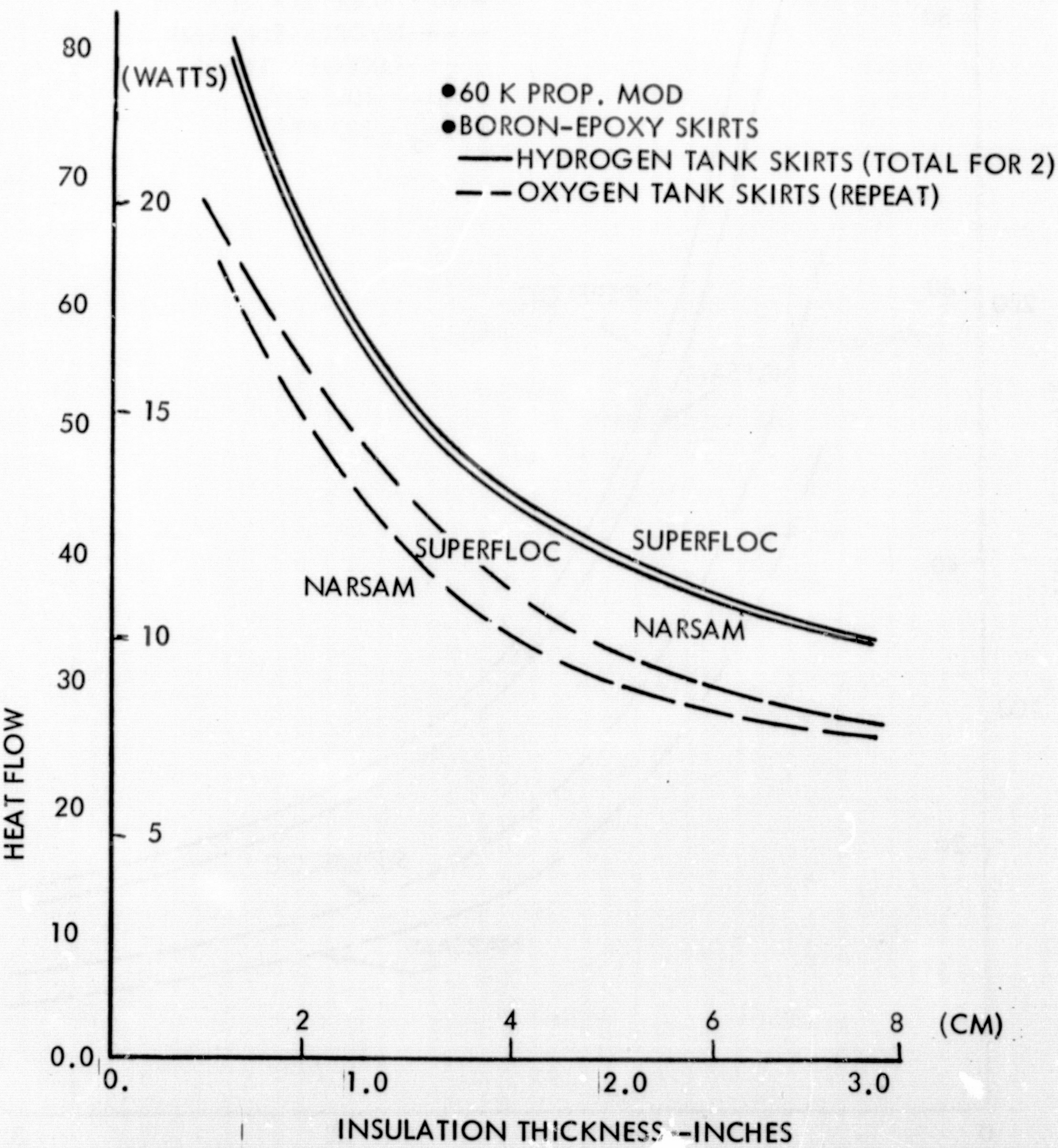


Figure F-28. Heat Flow Through Insulation Integral Skirts

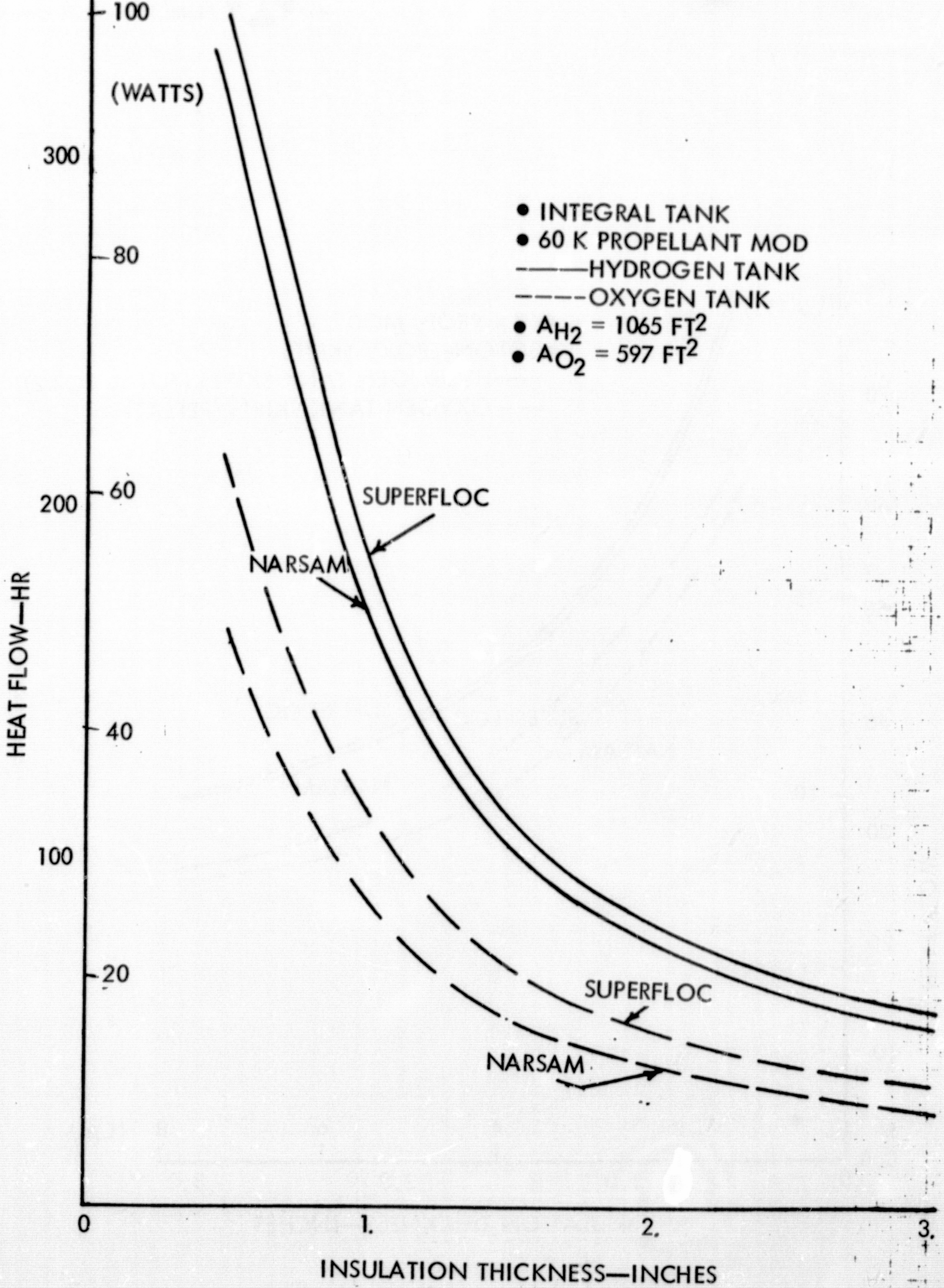


Figure F-29. Heat Flow to Cryo Tanks Through HPI Only

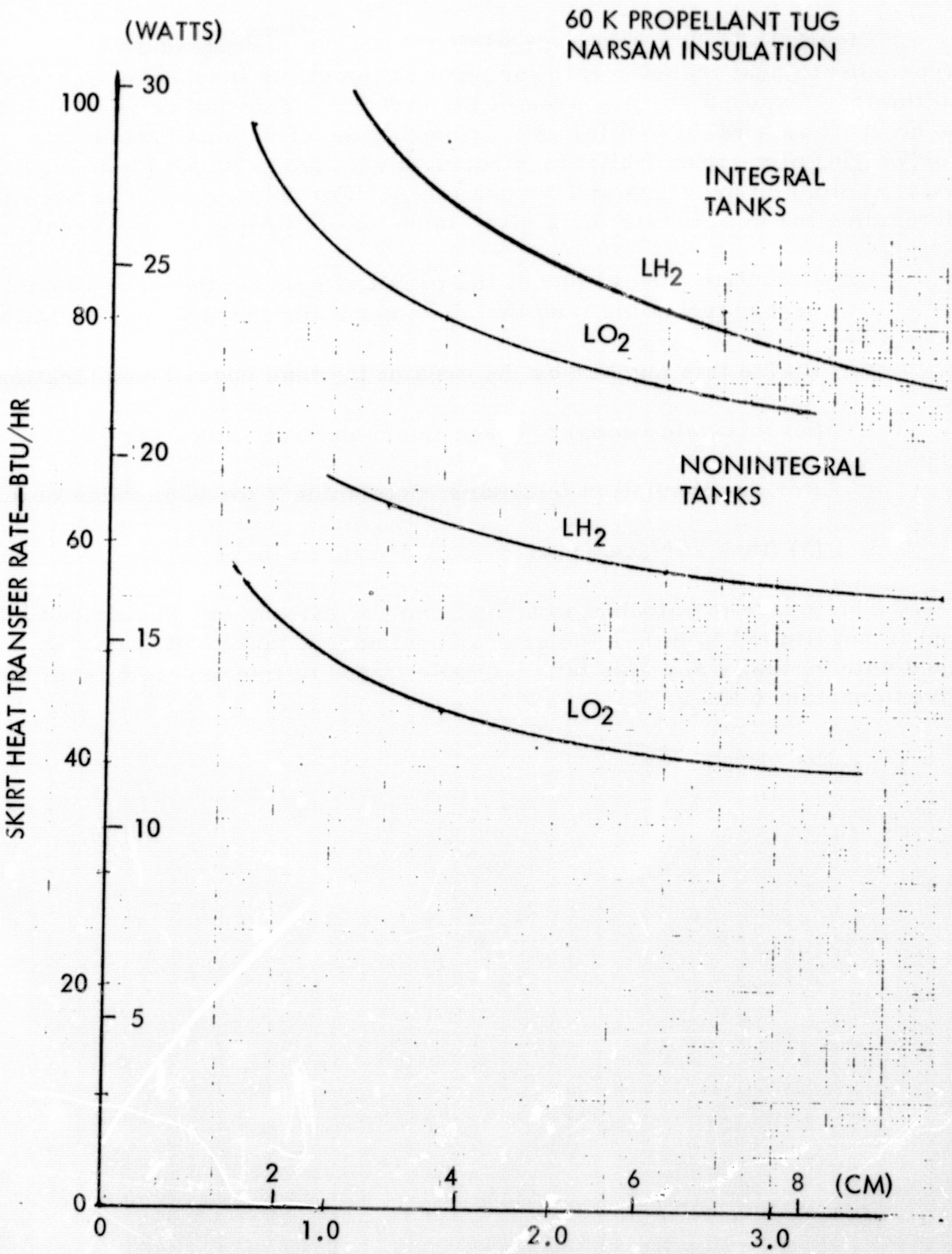


Figure F-30. Heat Transfer Rate Through Titanium Skirts Into Propellant Tanks

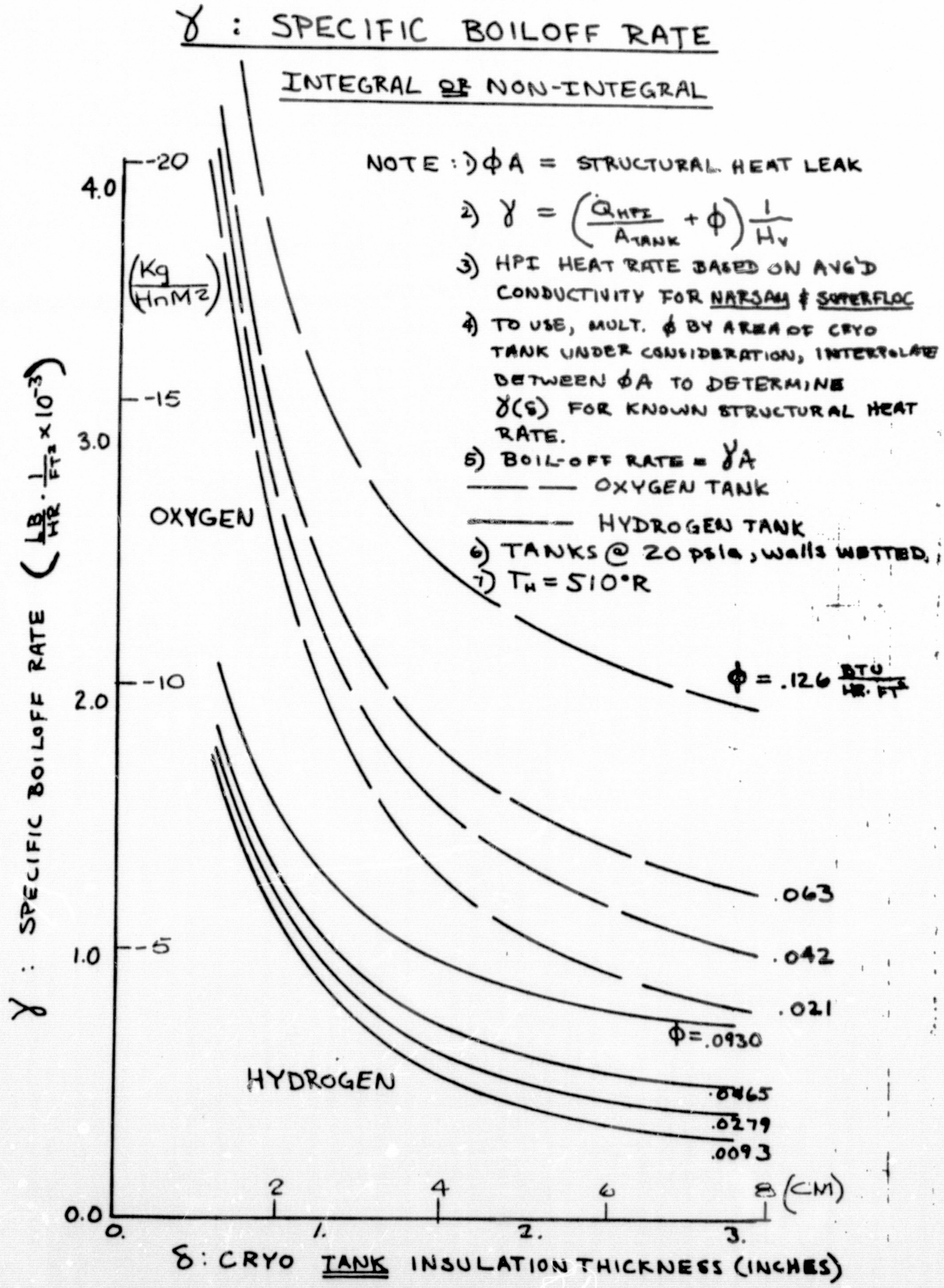


Figure F-31. Specific Boiloff Rate

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APPENDIX G

EXPENDABLE TUG CONCEPTS

by

D.W. Peebles, J. Schollenberger

and

R. Cook

APPENDIX G. EXPENDABLE TUG CONCEPTS

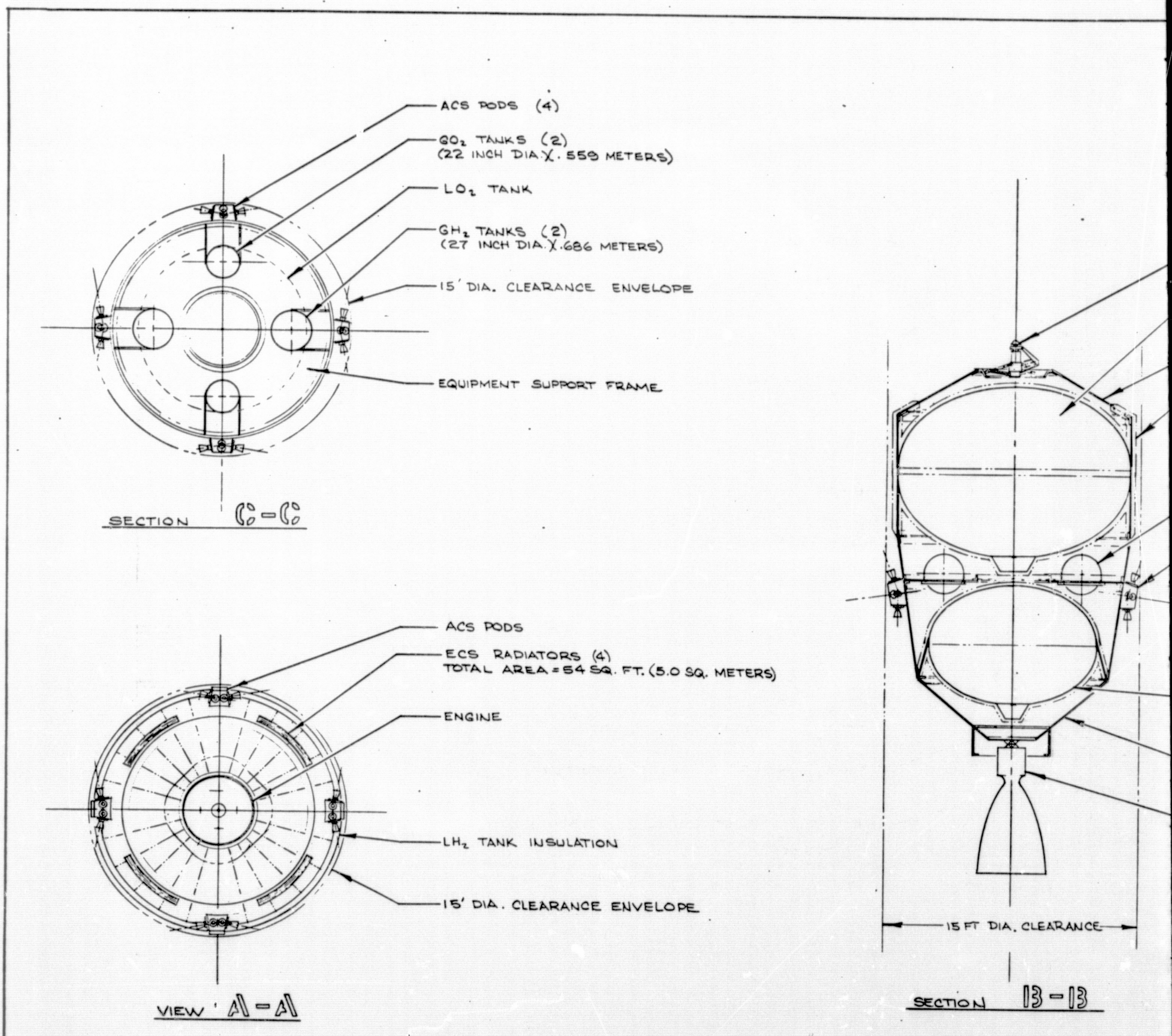
OBJECTIVE AND SCOPE

The objective of this effort was to develop preliminary configuration layouts for expendable tugs that could be costed and compared to reusable tug concepts.

Preliminary parametric performance analysis indicated that an O_2/H_2 expendable stage for the geosynchronous 10,000-lb (4535-Kg) payload design mission would require 27,000-lb (12,250 Kg) of propellants. An earth—storable stage (N_2O_4/A-50) would require 46,000 pounds (20,860 kilograms) of propellants. These two concepts were considered representative of good design for the cryogenic and the storable classes of expendable vehicles. Since the effort was accomplished at the very end of the study period, the effort was brief and was not accomplished in time to be included in the main body of this report. Minimum cost and weight were the prime objectives of the design studies.

EXPENDABLE O_2/H_2 CONCEPT

The cryogenic single-stage vehicle is conceptually shown in Figure G-1. It has been sized to accommodate 27,000 pounds (12,250 Kg) of O_2/H_2 propellants each contained in a single elliptical tank. The hydrogen tank is integral with the forward cylindrical structure, while the LOX tank is supported within a tapered aft body section. The hydrogen tank has a volume of 1040 cubic feet (29.43 cubic meters) and a diameter of 166 inches (4.22 meters) with 1.4:1 elliptical bulkheads. The forward and aft LH_2 tank skirts form the primary cylindrical structure of the vehicle. This section is fully insulated to aid the reduction of the thermal losses in the LH_2 tank. The primary structure between the liquid hydrogen and liquid oxygen tanks is skin-stringer aluminum conical construction. At the center of this section is an internal frame for supporting the gas tanks, ACS equipment, the avionics equipment, EPS equipment, and the environmental control system equipment. This frame also provides stability for the rigid, externally mounted attitude-control pod. The liquid oxygen tank is supported by an integral conical skirt attached to the primary structure at the lower end of the intertank section frame. The engine thrust structure also intersects the aft body structure at the same location.



FOLDOUT FRAME 1

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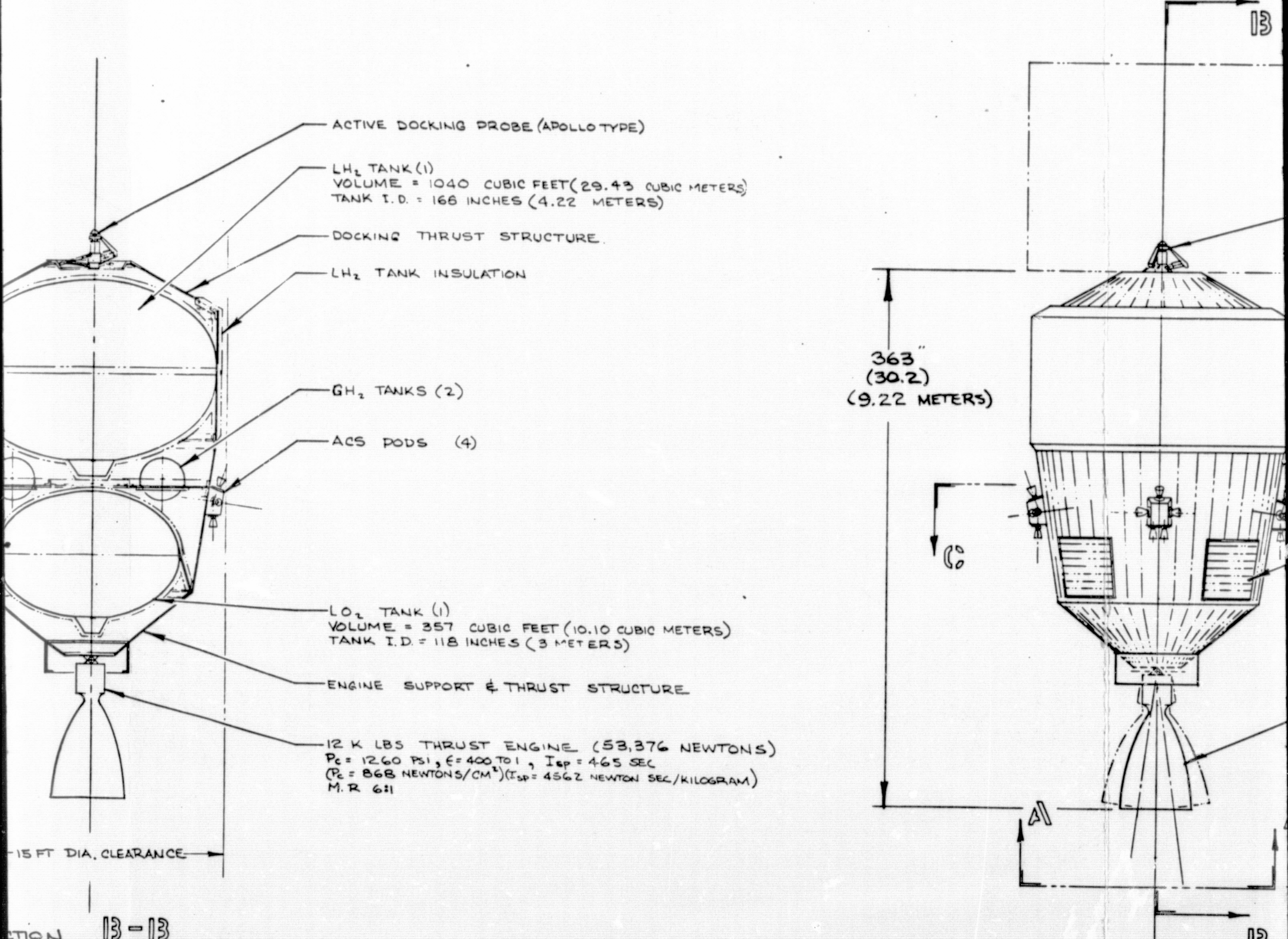


Figure G-1. Gro

FOLDOUT FRAME 2

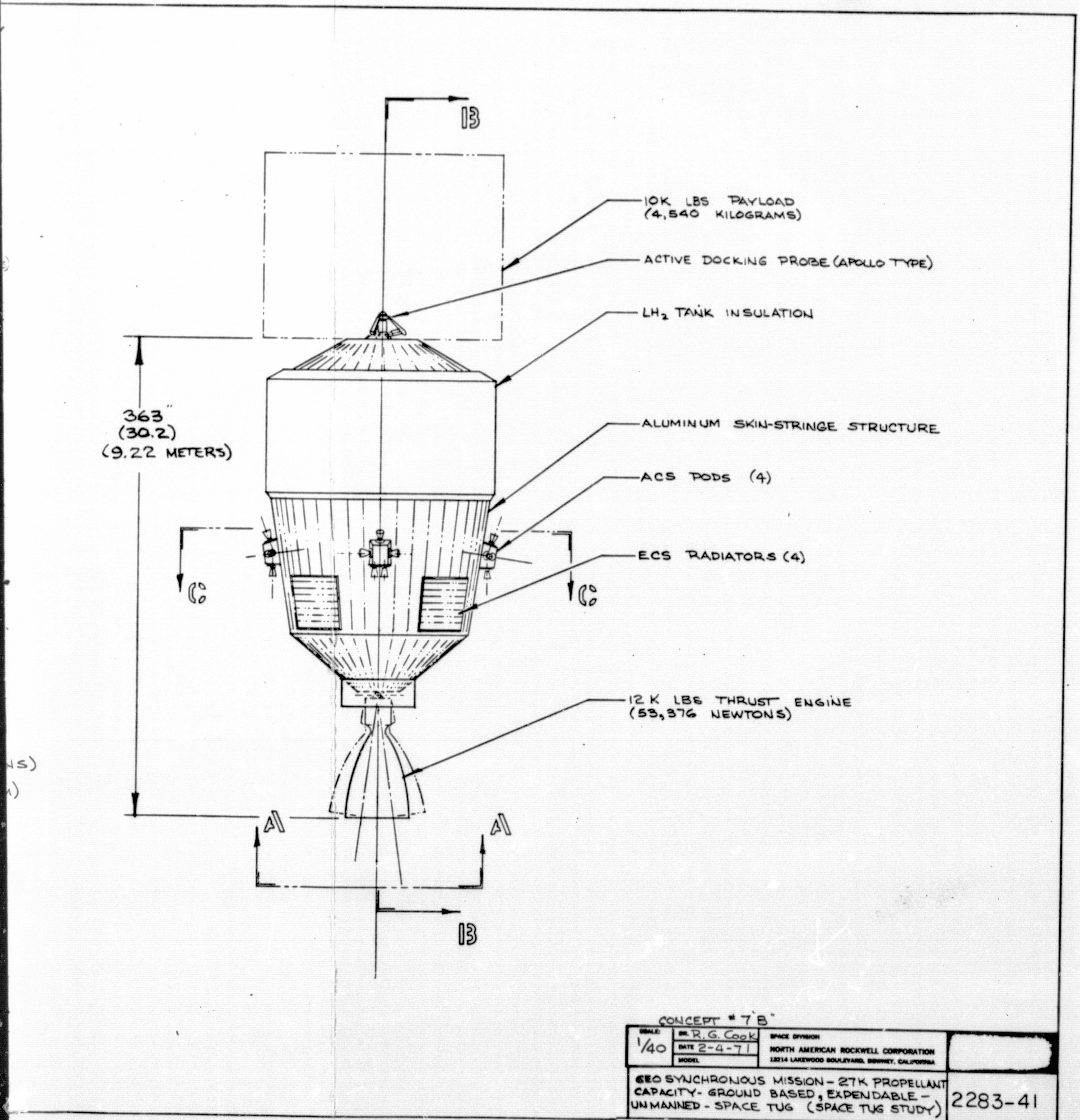


Figure G-1. Ground Based, Expendable-Unmanned Space Tug Geosynchronous Mission

FOLDOUT FRAME 3

The environmental control system radiators with an area of 54 square feet (5 square meters) are externally mounted on the intertank support structure. Access panels in the intertank support structures, permits the installation, servicing, and checkout of the internal subsystems of the vehicle.

The vehicle structure that supports the active docking probe (Apollo-type) is skin-stringer conical thrust structure attached to the forward liquid hydrogen tank support skirt. This structural intersection forms a ring frame at the top of the LH_2 tank support skirt.

The advanced single main engine has a thrust of 12,000 pounds (53,376 newtons), chamber pressure of 1260 psi (868 newtons per square centimeters), specific impulse of 465 seconds (4662 newton seconds per kilogram), an expansion ratio of 400 to 1, and a mixture ratio of 6 to 1. The main engine is attached to the aft end of the conical thrust structure.

This concept has no provisions for aft docking. Therefore, the EOS cargo bay support system attaches to the tug at the lower intertank primary structural frame, which receives all axial loads and approximately half of the lateral loads. The remaining lateral loads and supports are located on the upper LH_2 skirt frame. The overall length of the vehicle is 363 inches (9.22 meters), and the maximum diameter is 180 inches (4.57 meters).

PARAMETRIC STORABLE PROPELLANT TUG STUDY

Within the expendable geosynchronous mission vehicle optimization study it was determined that a storable propellant configuration might be cost effective and competitive with the cryogenic propellant vehicles. As it happened, the storable-propellant vehicle's initial cost was lower than the cryogenic propellant vehicle, and the costs per vehicle based on several vehicles over a long period of time were very nearly the same. Several initial ground rules and constraints were established in configuring the storable propellant vehicle. A single 15,000-pound (66,723-N) thrust engine was selected that was based on an uprated Agena-type engine. The chamber pressure was increased to 500 psia (345 N/cm^2), the area expansion ratio was increased to 150, and the specific impulse was increased to 325 seconds (3187 N-S/kg). Aerozine -50 (A-50) fuel and nitrogen tetroxide (N_2O_4) oxidizer were the selected propellants. Based on these propellants, a usable propellant weight of 46,020 pounds (20,874 kilograms) was required to deliver 10,000 pounds (4536 kilogram) of payload to geosynchronous orbit.

The study was initiated through consideration of several tankage arrangements. These included common bulkhead tanks, single fuel and oxidizer tanks of spherical and elliptical shape, and arrangements of

four cylindrical tanks. An initial spectrum of possible candidate vehicle configurations was prepared and is shown in Figure G-2. Eight concepts were configured of various tankage arrangements, and two concepts that offered distinct advantages were chosen for further design definition.

All of the concepts used a single engine and an integrated subsystem equipment approach. They also were all configured to stay within the 15-ft-diameter (4. 57-m) EOS cargo bay payload envelope without retraction or stowage of appendages being required such as the ACS engine modules. Wherever possible, an integrated tank structure was used to minimize the inert weight.

Concept "A" of Figure G-2 incorporated single fuel and oxidizer tanks with elliptical bulkheads of a 1. 4 to 1 ratio. The structure is an integrated approach, and the subsystem equipment and ACS propellant tanks are located in the annular area between the main propellant tanks. The vehicle is 312 inches (7. 9 meters) long and is 120 inches (3. 0 meters) in diameter. Concept "B" is identical to "A" except that spherical tanks were used rather than ellipsoids. This stage is 355 inches (9. 0 meters) long and 108 inches (2. 7 meters) in diameter. Concepts "C" and "D" investigated common bulkheads. Concept "C" used an elliptical bulkhead of 1. 4 to 1 ratio, and "D" used spherical bulkheads. The vehicles were 298 inches (7. 6 meters) and 328 inches (8. 3 meters) long and 120 inches (3. 0 meters) and 108 inches (2. 7 meters) in diameter consecutively for "C" and "D." Subsystem equipment was located in the annular area at the forward end of the vehicles.

The last single-tank arrangement, Concept "E," used a spherical oxidizer and elliptical fuel tank. This stage measured 334 inches (8. 5 meters) long and was 120 inches (3. 0 meters) in diameter at the forward end and tapered inboard toward the aft end. The subsystem equipment was located as in Concepts "A" and "B." The remaining three concepts investigated the use of multiple tanks. The first concept, "F," used four tanks of equal volume. The ACS pods were located on the aft tapered section of the structure to keep them within the 15-ft-diameter (4. 57-meters) envelope. The vehicle maximum diameter is 180 inches (4. 57 meters), which allows the propellant tanks to be as large as possible. The tanks have spherical bulkheads with a cylindrical midsection. Four radial beams support for the engine and the bulkheads, which supports the tanks. Subsystem equipment is located in the annular area at the aft end of the structure.

The remaining two configurations used tanks of the same diameter and bulkheads as the Apollo service module service propulsion system storage and sump tanks. Concept "G" used four tanks of the same diameter as the sump tanks, 51 inches (1. 3 meters) inside diameter. The two fuel and

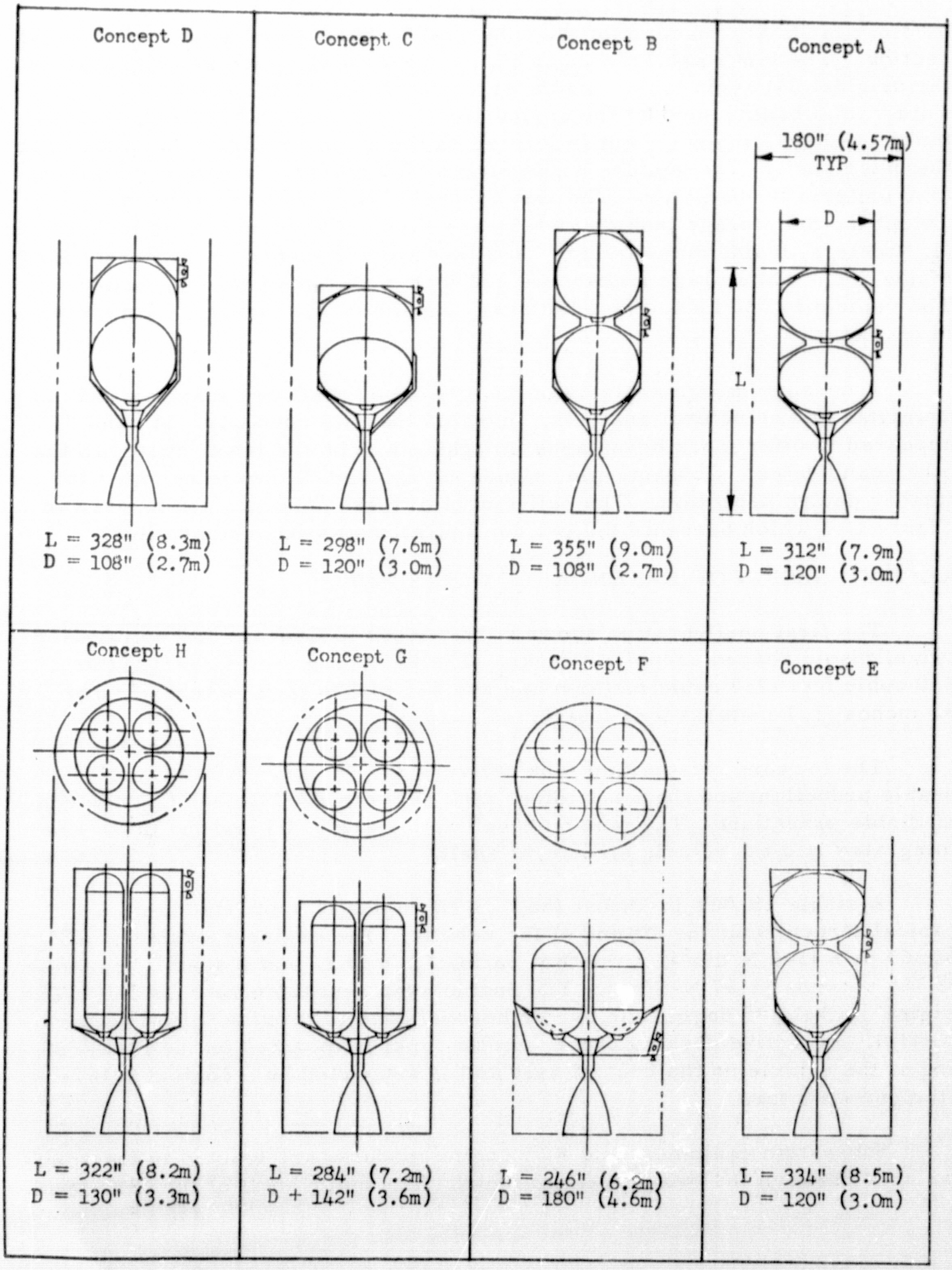


Figure G-2. Candidate Vehicle Configurations

oxidizer tanks are made up of the spherical bulkheads and a long cylindrical section. The tanks are 157 inches (3.9 meters) long inside as compared to the original SM sump tanks, which were 152 inches (3.8 meters). Four radial beams support the engine and the bulkhead, which supports the tanks. Subsystem equipment is located on the beams between the tanks near the outer shell. The vehicle is 284 inches (7.2 meters) long and 142 inches (3.6 meters) in diameter. The last concept, "H," is identical to "G," except the SM storage tank diameters are used. Four tanks of 45 inches (1.1 meters) inside diameter are used that are 195 inches (4.9 meters) long inside. The SM storage tanks were 152 inches (3.8 meters) long inside. The vehicle is 322 inches (8.2 meters) long and is 130 inches (3.3 meters) in diameter.

A cursory design analysis of the system of configurations, as well as a weights and structural analysis, revealed that two concepts, "A" and "F," appeared to offer more advantages in lighter weight and lower cost than the other candidates. Consequently, Concepts "A" and "F" were selected for further design definition. The refinement of these concepts can be seen in Figure G-3 which presents the two configurations as inboard profiles.

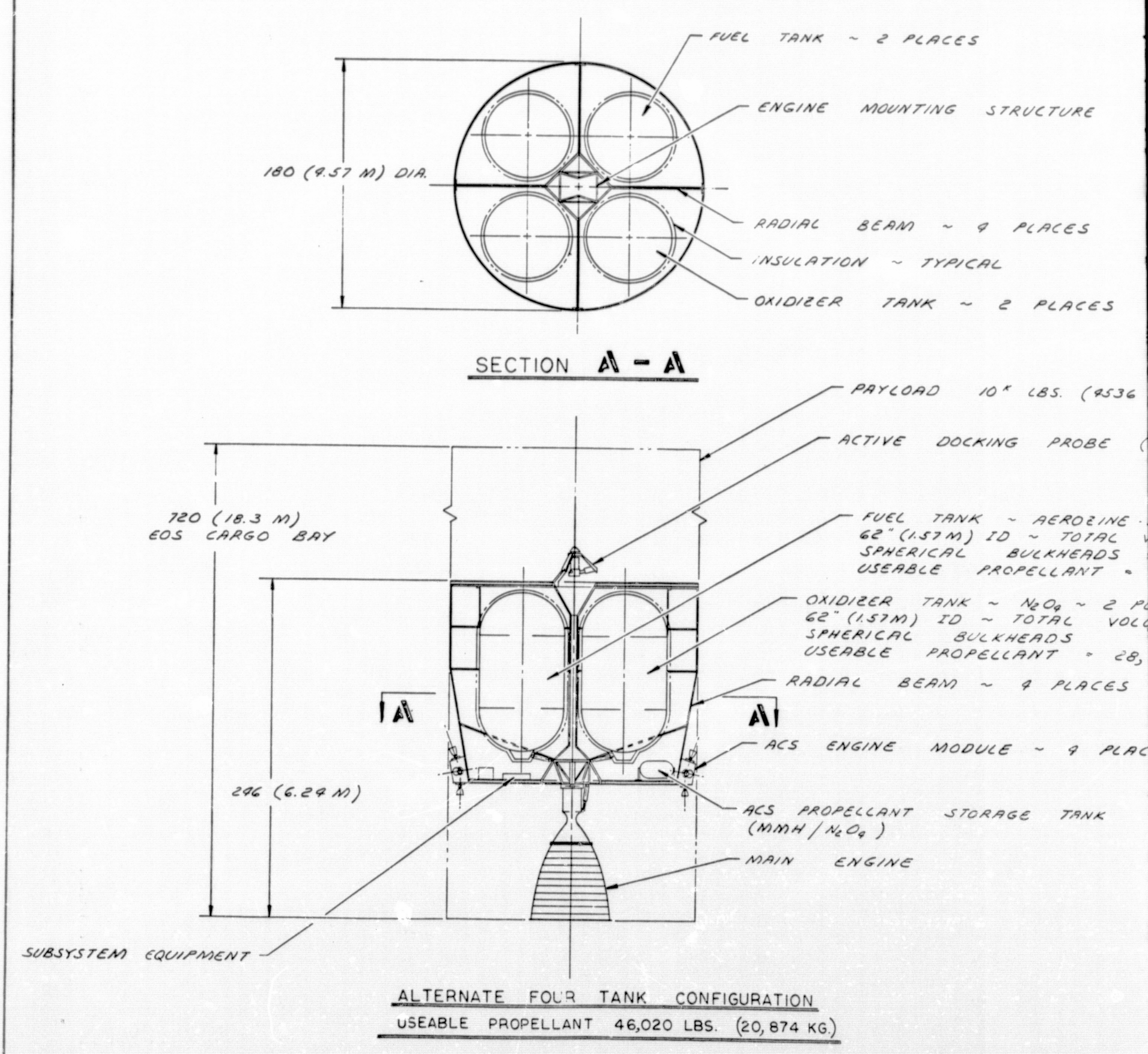
REFINED DESIGN OF EXPENDABLE CONCEPTS

The first configuration shown corresponds to Concept "A" which used two ellipsoid shaped propellant tanks. The two tanks are of equal volume, 330 cubic feet (934 cubic meters) and are 115 inches (2.9 meters) ID and 82 inches (2.1 meters) long inside.

The forward located fuel tank delivers 17,700 lbs. (8,029 Kg) of usable propellant and the aft oxidizer tank delivers 28,320 lbs. (12,846 Kg) of usable propellant. The tank bulkhead ratios are 1.4:1 and the tanks are integrated into the vehicle structural shell.

A single 15,000-lb-thrust (66,723-N) engine is supported from a conical structure at the aft end of the vehicle. The engine operates at a P_c of 500 psia (345 N/cm^2), a mixture ratio of 1.6 to 1, and a specific impulse of 325 seconds (3187 N-S/kg). The nozzle area expansion ratio is 150. The engine gimbals ±5 degrees in two orthogonal axes to provide thrust vector control. An active docking probe (Apollo-type) is mounted on the forward end of the vehicle on the central axis and is supported by a conical skin-stringer structure.

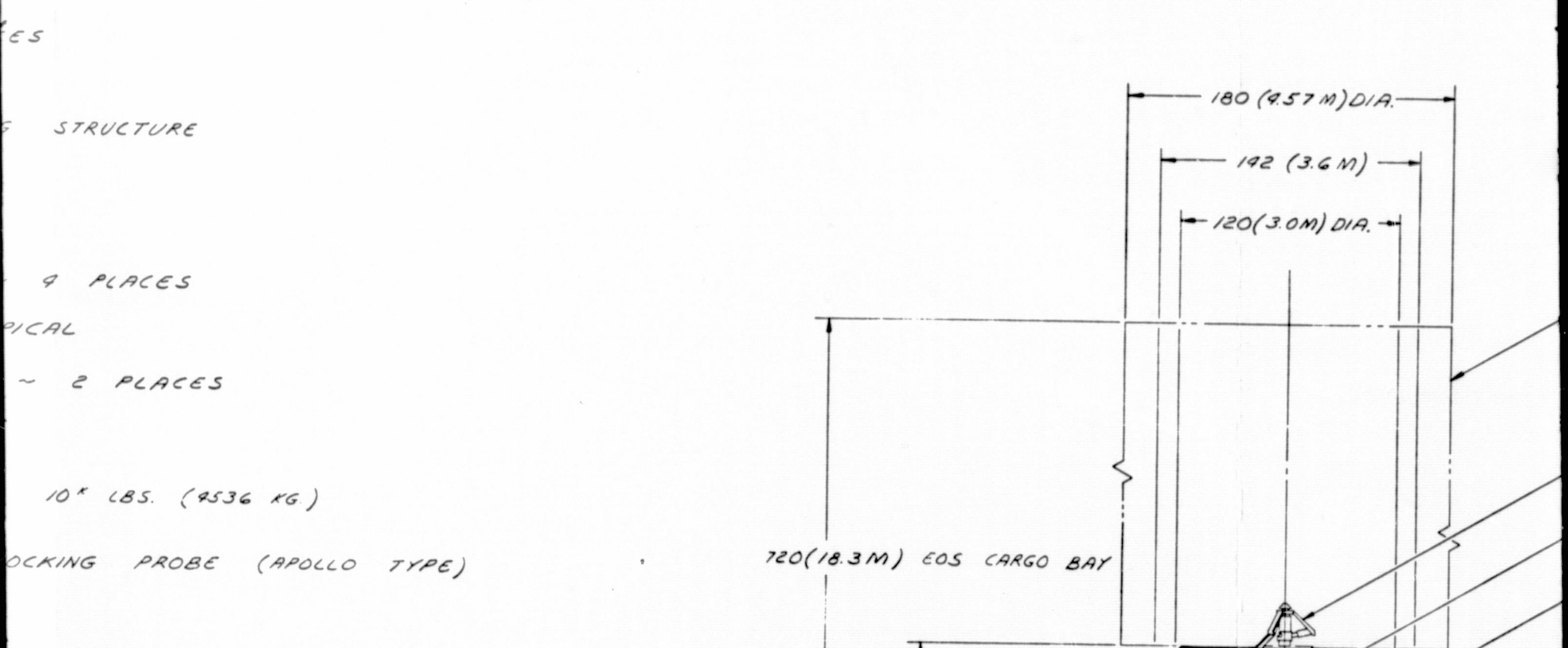
Subsystem equipment and ACS engine modules are loacted in the annular area between the two propellant tanks. The outer shell of the vehicle is



ALTERNATE FOUR TANK CONFIGURATION

USEABLE PROPELLANT 46,020 LBS. (20,874 KG.)

FOLDOUT FRAME 1



INTEGRATED DUAL TANK EXPENDABLE CON

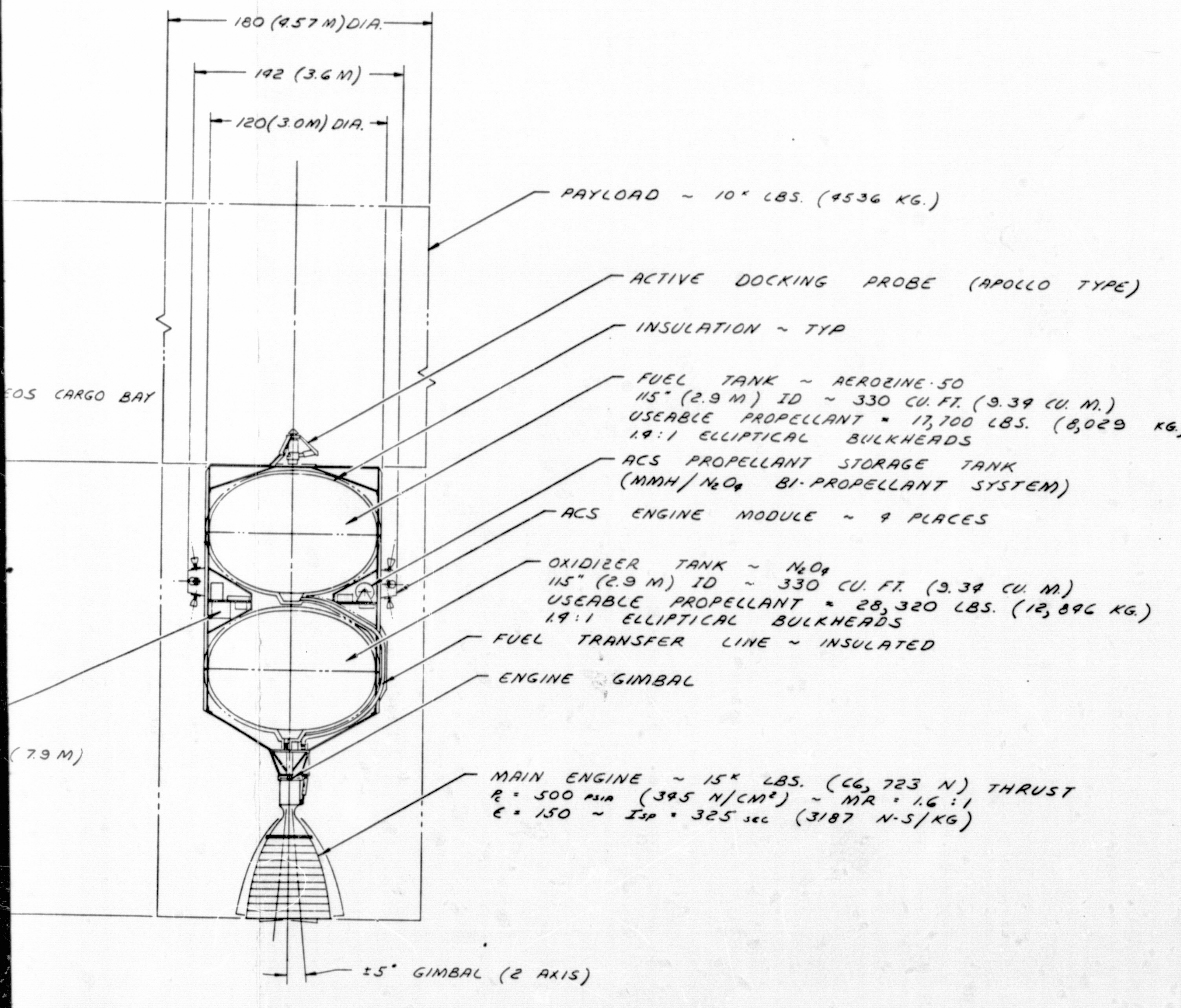
USEABLE PROPELLANT 46,020 LBS. (20,874

STORABLE PROPELLANTS A-50/N_2O_4

Figure G-3. Expend

FOLDOUT FRAME 2

(12)



INTEGRATED DUAL TANK EXPENDABLE CONFIGURATION

USEABLE PROPELLANT 46,020 LBS. (20,874 KG.)

STORABLE PROPELLANTS A-50/N_2O_4

SCALE 1/40	DR. J. Shallenberger	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
	DATE 2-3-71		
	MODEL TUG		
EXPENDABLE GEOSYNCHRONOUS MISSION STORABLE PROPELLANT 46,020 LBS. (20,874 KG.) SPACE TUG STUDY			2283-42

Figure G-3. Expendable Geosynchronous Mission Storable Propellant 46,020 Pounds (20,874 kg)

FOLDOUT FRAME 3

of skin-stringer construction and is integrated with the two propellant tanks and the forward and aft support cones. The vehicle is 312 inches (7.9 meters) long, excluding the docking probe, and is 120 inches (3.0 meters) in diameter. The overall distance between the flats of the ACS engine modules, which extend beyond the 120-in. (3.0 m) diameter, is 142 inches (3.6 meters).

The alternate configuration shown is based on Concept "F." This vehicle uses the maximum available diameter within the EOS cargo bay, 180 inches (4.57 meters), to obtain the shortest possible configuration. Four tanks with inside diameters of 62 inches (1.57 meters) and 115 inches (2.9 meters) long were used with an internal volume of 165 cubic feet (4.67 cubic meters) each. The two fuel tanks deliver 17,700 pounds (8,029 kilograms) of usable oxidizer, and the two oxidizers deliver 28,320 pounds (12,846 kilograms) of usable propellant. The structural concept employed in this configuration is not an integrated structure approach. Four radial beams are used with a skin-stringer shell. The beams support the single engine, which is identical to that used in the first configuration, in addition to a forward and aft bulkhead. The tanks are supported on skirts from the forward bulkhead. The aft section of the outer shell is tapered inboard, and the ACS engine modules are attached to the aft section of this structure. This location keeps the modules within the 180-in-diameter (4.57-m) EOS cargo bay payload envelope.

An active docking probe is located at the forward end and is supported from the beams and bulkhead. Subsystem equipment is located in the annular area at the aft end of the vehicle. The equipment and ACS bipropellant storage tanks are mounted on the aft bulkhead. The bipropellant system selected for the ACS system is MMH/N_2O_4. This system was also used in the first configuration. This vehicle is 246 inches (6.24 meters) long, excluding the probe, and 180 inches (4.57 meters) in diameter.

Analysis of each configuration indicated that little difference in weight exists between the two configurations. Consequently, both require the same amount of propellant. Both configurations use new tankage. The second configuration is the shortest possible, with the exception of using elliptical bulkheads on the same tanks, which would decrease the overall length by only 6 inches (0.15 meters). The first configuration is considered as the simplest with the least number of components.